AW109SP

ROTORCRAFT FLIGHT MANUAL

Agusta Publication Code 502700600

AgustaWestland
A Finmeccanica Company

ROTORCRAFT FLIGHT MANUAL

AW109SP

Document N°109G0040A018

Approved by EASA

Copy assigned to rotorcraft: S/N . . . .

Registration Marks . . . .

ISSUE B: 11 DECEMBER 2009
REVISION 4: 11 FEBRUARY 2011

THIS DOCUMENT MUST BE CARRIED IN AIRCRAFT AT ALL TIMES

Issued by Agusta S.p.A.

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LIST OF REVISIONS

REVISION No.	SUBJECT	APPROVAL
—	Issue A	EASA Approval with Type Certificate Data Sheet (TCDS) R.005 dated 25 May 2009
—	Issue B	EASA Approvals N° 10028242, 10028244, 10028246, 10028248 dated 11 December 2009
1	Revised Title page, A-1, B-1 and 3-147.	EASA Approval N° 10031062 dated 26 July 2010
2	Revised Title page, A-1, B-1, B-2, 1-25, 2-iv, 2-64, 2-66, 2-67 thru 2-70, 3-iv thru 3-viii, 3-74 thru 3-76, 3-103 thru 3-108 and 3-169 thru 3-171. Added pages 2-71 and 2-72.	EASA Approvals N°10030940,10031063 dated 26 July 2010
3	Revised Title page, A-1, B-1, B-2, 1-iii, 1-25, 1-26, 2-20 thru 2-25 and 2-30.	EASA Approval N° 10032356 dated 8 November 2010
4	Revised Title page, A-1, B-1, B-2, 1-25, 1-30, 2-20, 2-30, 3-167, 3-168. Added pages B-3 and B-4.	EASA Approval N° 10033829 dated 11 February 2011
	Revised pages 1-31, 1-37, 1-39, 1-41, 4-24 and 4-25.	Approved under the Authority of DOA N° EASA 21J.005

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INTRODUCTION

GENERAL

It is responsibility of the flight crew to be familiar with the contents of this manual.

REVISIONS (REISSUES)

This manual is subject to revisions (reissues) which will be automatically distributed to all holders of the manual. It is the responsibility of the operator to assure that the revisions (reissues) are incorporated into the manual upon receipt.
At the beginning of the manual there is the “List of Revisions” table that shows all pages of the manual which have been revised as well as number and approval reference of each revision.

REVISION SYMBOL

Revised text is indicated by a black vertical line on the outer margin of the page, adjacent to the affected text and the revision is printed in the lower inner margin. The revision symbol identifies the addition of new information, a change of procedure, the correction of an error, or a rewording of the previous information.

TEMPORARY REVISIONS

Temporary Revisions are issued when immediate data is to be included in the manual. The Temporary Revision data can add to or cancel the initial data in the manual. They are numbered progressively for each section of the manual. Temporary Revision pages are not written in the “List of Effective Pages”. A complete list of **active** and **inactive** Temporary Revision are written in the “List of Temporary Revisions” page.

TERMINOLOGY

WARNINGS, CAUTIONS AND NOTES

WARNING

An operating procedure, practice, etc., which, if not correctly followed, could result in personal injury or loss of life.

CAUTION

An operating procedure, practice, etc., which, if not strictly observed, could result in damage to, or destruction of, equipment.

Note

An operating procedure, condition, etc., which is essential to highlight.

USE OF PROCEDURAL WORDS

The concept of procedural word usage and intended meaning which has been adhered to in preparing this RFM is as follows:

"**Shall**" or "**Must**" have been used only when application of a procedure is mandatory.

"**Should**" has been used only when application of a procedure is recommended.

"**May**" has been used only when application of a procedure is optional.

"**Will**" has been used only to indicate futurity, never to indicate a mandatory procedure.

"**Condition**" has been used to determine if the item under examination presents external damage which could jeopardize its safe operation.

"**Secured**" has been used to determine if the item under examination is correctly locked; mainly referred to doors and disconnectable items.

"**Security**" has been used to determine if the item under examination is correctly positioned and installed.

"Fly Attentive" has been used when the pilot needs to maintain close control of the flight path using hands-on when required.

"Fly Manually" has been used when the pilot needs to directly control the flight path using hands-on.

ABBREVIATION

The use of capitol letters in the text, apart from normal grammatical usage indicates the actual wording or marking of indicators, controls or control positions on the helicopters.
Abbreviations and acronyms used throughout this RFM are defined as follows:

— a.c : Alternating current

— ADC : Air Data Computer

— ADU : Air Data Unit

— ADF : Automatic Direction Finder

— ADI : Attitude Data Indicator

— ADL : Airborne Data Loader

— ADLCP : Airborne Data Loader Control Panel

— ADS : Air Data System

— AEO : All Engines Operating

— A/F : Airframe

— AFCS : Automatic Flight Control System

— AFDS : Automatic Flight Director System

— AGL : Above Ground Level

— AHRS : Attitude and Heading Reference System

— ALS : Above Landing Surface

— ALT : Baro - ALTitude mode

— AM : Amplitude Modulation
— ANP : Actual Navigation Performance
— AP : Automatic Pilot
— APCP : AutoPilot Control Panel
— API : Actuator Position Indicator
— APD : Actuator Position Display
— ATH : Approach To Hover
— ATTD : ATTituDe
— AWG : Aural Warning Generator
— BC : Back Course mode
— BCD : Binary Coded Data
— BIT : Built In Test
— BRG : BeaRinG
— BRT : BRighTness
— CAS : Calibrated AirSpeed
— CCW : CounterClockWise
— CDS : Cockpit Display System
— CG : Center of Gravity
— CHS : Core Helivionics System
— CLB : CLimB mode
— CLR : CLeaR
— CMS : CoMmunication System
— CMU : Communication Management Unit
— COLL : COLLective mode
— COM : COMmunication
— CPL : CouPLing
— CRS : CouRSe

— CS	:	Certification Specifications
— CST	:	ContraST
— CVDR	:	Cockpit Voice Data Recorder
— CVR	:	Cockpit Voice Recorder
— CW	:	ClockWise
— d.c	:	Direct current
— DF	:	Direction Finder
— DH	:	Decision Height
— DIS	:	DIStance
— DME	:	Distance Measuring Equipment
— DMG	:	Digital Map Generator
— DP IFR	:	Dual Pilot IFR
— DR	:	Dead Reckoning mode
— DTK	:	Desired TracK
— DTM	:	Data Transfer Module
— DTO	:	Direct TO
— DU	:	Display Unit
— DVS	:	Doppler Velocity System
— EADI	:	Electronic Attitude Direction Indicator
— ECU	:	Engine Control Unit
— EDU	:	Electronic Display Unit
— EECU	:	Engine Electronic Control Unit
— EEPROM	:	Electrically Erasable Programmable Read Only Memory
— ELEC	:	ELECtrical
— ELT	:	Electronic Locator Transmitter

— EMM	:	Engine Maintenance Manual
— ENT	:	ENTer
— EOP	:	Engine Oil Pressure
— EOT	:	Engine Oil Temperature
— ESIS	:	Electronic Standby Instrument System
— ETA	:	Estimated Time of Arrival
— F()	:	Function of
— F^{-1}()	:	Inverse Function of
— FADEC	:	Full Authority Digital Engine Control
— FAF	:	Final Approach Fix
— FAR	:	Federal Airworthiness Regulation
— FCC	:	Flight Control Computer
— FCU	:	Fuel Computer Unit
— FD	:	Flight Director
— FDCP	:	Flight Director and Control Panel
— FDE	:	Flight Deck Effect
— FDR	:	Flight Data Recorder
— FLI	:	First Limit Indicator
— FLIR	:	Forward Looking Infra-Red
— FM	:	Frequency Modulation
— FMEA	:	Failure Modes and Effects Analysis
— FMS	:	Flight Management System
— FP	:	Fuel Pressure
— FPLN	:	Flight PLaN
— FPS	:	Flight Planning System
— FTR	:	Force Trim Release

- G/S : Glide Slope
- Hd : Density altitude
- HDG : HeaDinG
- HDG SEL : HeaDinG SELect
- HF : High Frequency
- HOV : HOVer
- Hp : Pressure Altitude
- HTAWS : Helicopter Terrain Awareness and Warning System
- HUMS : Health and Usage Monitoring System
- HYD : HYDraulic
- IAS : Indicated AirSpeed
- ICS : InterCommunication System
- IGE : In Ground Effect
- IFR : Instrument Flight Rules
- IGRF : International Geomagnetic Reference Field
- ILS : Instrument Landing System
- IMC : Instrument Meteorological Condition
- INS : Inertial Navigation System
- ISA : International Standard Atmosphere
- ITT : Inter-Turbine Temperature
- IVSI : Instantaneous Vertical Speed Indicator
- JAR : Joint Aviation Regulations
- KCAS : Knots Calibrated AirSpeed

— LCD	:	Liquid Crystal Display
— LCF	:	Low Cycle Fatigue
— LH	:	Left Hand
— LNAV	:	Lateral NAVigation
— LOC	:	LOCalizer
— LX	:	Stormscope / Lightning detector
— MAINT	:	MAINTenance
— MCDU	:	Multifunction Control and Display Unit
— MAP	:	Missing Approach Point
— MCL	:	Master Caution Light
— MCP	:	Maximum Continuous Power
— MDA	:	Minimum Descent Altitude
— MER	:	Modular Electronic Rack
— MFD	:	Multi Function Display
— MFR	:	ManuFactureR
— MGB	:	Main Gear Box
— MISS	:	MISSion
— MON	:	MONitoring
— MSG	:	MeSsaGe
— MSM	:	Mission System Management module
— MTR	:	MoniToR
— MWL	:	Master Warning Light
— NAVM	:	NAVigation Module
— ND	:	Navigation Display
— NF	:	Free turbine speed (= N2)

— NG	: Gas generator speed (= N1)
— NM	: Nautical Miles
— NR	: Rotor speed
— NVM	: Non Volatile Memory
— NVG	: Night Vision Goggles
— OAT	: Outside Air Temperature
— OEI	: One Engine Inoperative
— OGE	: Out of Ground Effect
— OTS	: Observation Targeting System
— PAM	: Power Assurance Margin
— PAX	: Passenger(s)
— PCP	: PFD Control Panel
— PERF	: PERFormance
— PFD	: Primary Flight Display
— PFT	: Pre Flight Test
— PLA	: Power Lever Angle (alias engine power lever)
— PMS	: Power Management Switch (alias engine mode switch)
— PROG	: PROGress
— PSSA	: Preliminary System Safety Analysis
— QTY	: QuantiTY
— R ALT	: Radar Altimeter mode
— R.A	: Radar Altimeter
— RAM	: Random Access Memory
— RCP	: Reconfiguration Control Panel
— REC	: RECording

— RF : Radio Frequency
— RFI : Radio Frequency Indicator
— RFM : Rotorcraft Flight Manual
— RH : Right Hand
— RNAV : Radio NAVigation
— RNG : RaNGe
— RNP : Required Navigation Performance
— ROC : Rate Of Climb
— RPM : Revolutions Per Minute
— SAS : Stabilization Augmentation System
— SCP : System Control Panel
— SCRL : SCRolL
— SID : Standard International Departure
— SOE : Set Of Sensors
— SPIFR : Single Pilot IFR
— STAR : STandard ARrival
— STD : STandarD
— SYS : SYStem
— TAS : True AirSpeed
— TAWS : Terrain Awareness and Warning System
— TCA : Track Change Alert
— TKE : TracK angle Error
— TOC : Top Of Climb
— TOP : Take Off Power
— TQ : TorQue
— TRQ : ToRQue

— TNG : TraiNinG
— TST : TeST
— TTD : Time To the end of Destination
— TTG : Time To Go to the way point
— UHF : Ultra High Frequency
— UTC : Coordinated Universal Time
— UTM : Universal Transverse Mercator
— VFR : Visual Flight Rules
— VHF : Very High Frequency
— VID : Video
— V_{LE} : Maximum landing gear extended speed
— V_{LO} : Maximum landing gear operating speed
— VMM : Vehicle Monitoring Module
— VMS : Vehicle Monitoring Systems
— V_{NE} : Velocity Never Exceed
— V_{NEI} : V_{NE} for IFR operations
— V_{MINI} : Minimum airspeed for flight under Instrument Flight Rules
— VOR : VHF Omni-directional Relative bearing
— VS : Vertical Speed
— Vy : Best Rates of Climb Speed
— WCA : Warning Caution Advisory
— WOW : Weight On Wheels
— WX : Weather Radar
— WxR : Weather Radar
— XTK : Cross TracK Distance

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PART I - EASA APPROVED

SECTION 1

LIMITATIONS

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SECTION 1

LIMITATIONS

GENERAL

Compliance with Section 1 of this manual is mandatory.

The helicopter must also be operated in accordance with the appropriate operating rules.

BASIS OF CERTIFICATION

The helicopter is certified by the European Aviation Safety Agency (EASA) in accordance with CS / JAR / FAR 27 for Small Rotorcraft as detailed in the Type Certificate Data Sheet EASA R.005.

TYPE OF OPERATIONS

The AW109SP helicopter, in its basic configuration, is approved for Day and Night VFR and IFR operations in non-icing conditions.

Aerobatic manoeuvres are prohibited.

MINIMUM FLIGHT CREW

The minimum flight crew consists of one pilot who shall operate the helicopter from the right crew seat.

The left crew seat may be used for an additional pilot when the approved dual controls are installed.

NUMBER OF SEATS

Eight, including the pilot’s seat.

WEIGHT AND CENTER OF GRAVITY LIMITATIONS

WEIGHT

Maximum gross weight for
ground taxing and towing .... 3175 kg (7000 lb)

Maximum gross weight for take-off and landing........ 3175 kg (7000 lb)

Minimum gross weight for flight................................. 2050 kg (4519 lb)

CENTER OF GRAVITY

Longitudinal limits... See Figures 1-1and 1-2

Lateral limits .... See Figures 1-1and 1-2

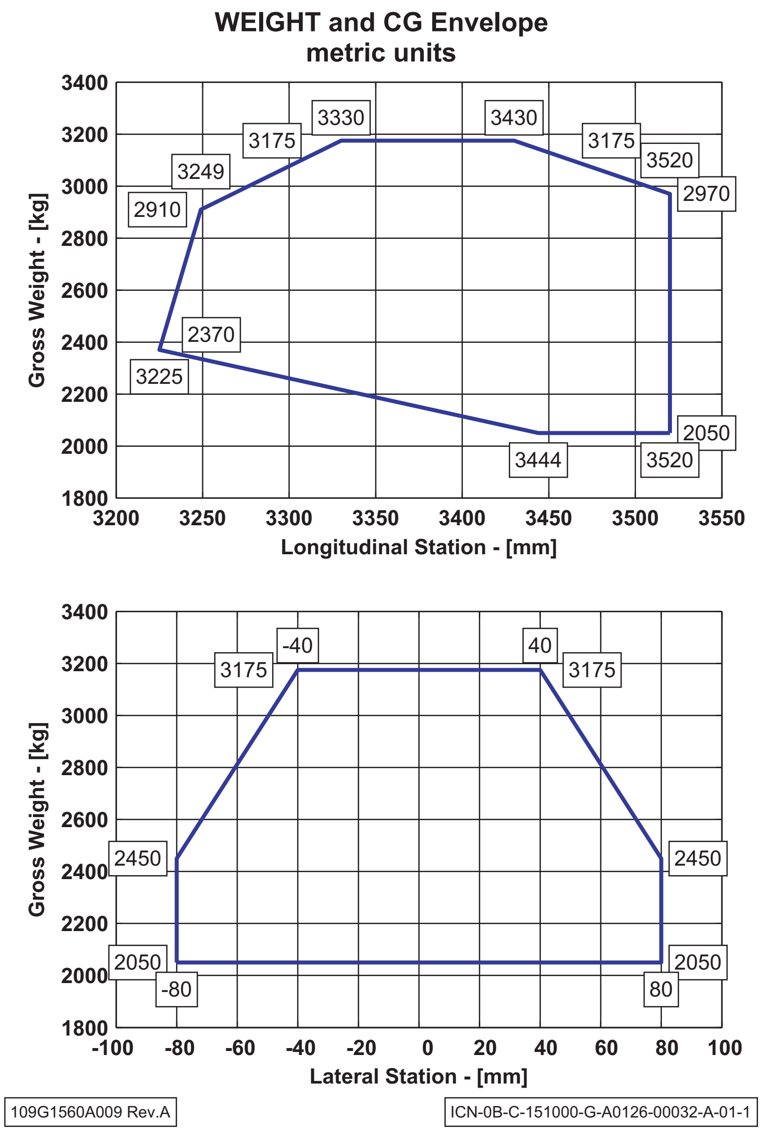

Figure 1-1 Weight , Longitudinal and Lateral CG Envelope (metric units)

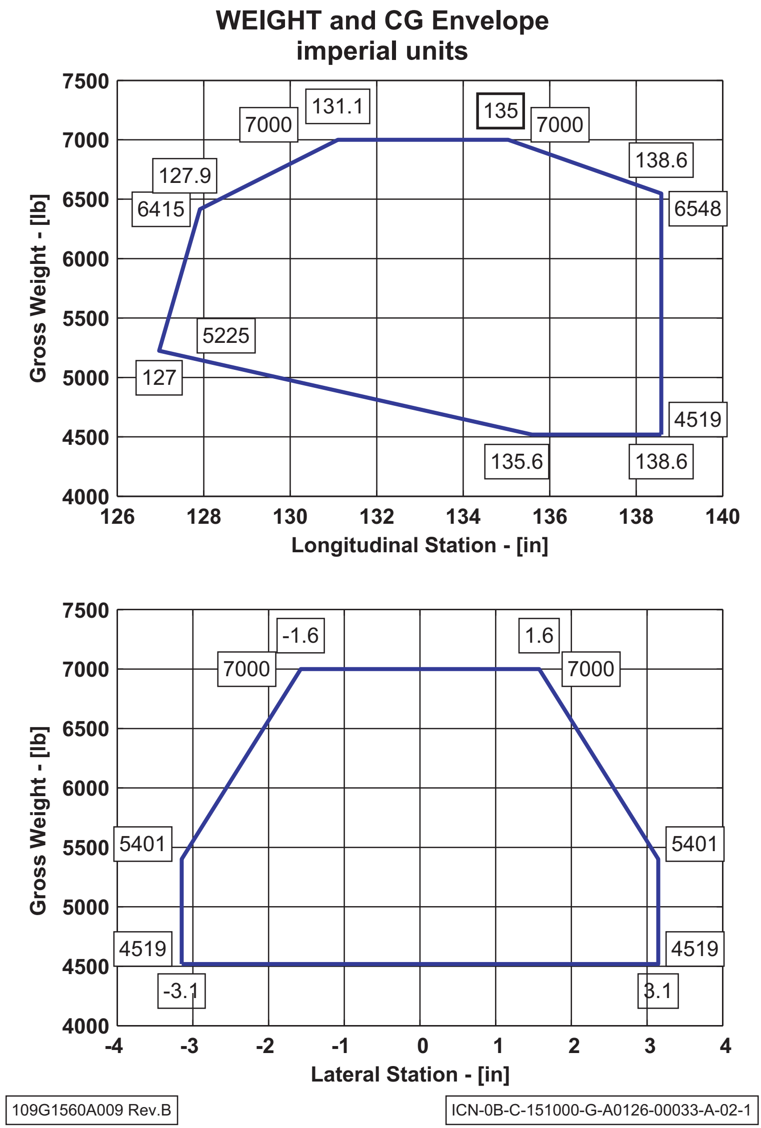

Figure 1-2 Weight, Longitudinal and Lateral CG Envelope (imperial units)

AIRSPEED LIMITATIONS

V_{NE} (Power-ON) .... See Figure 1-3

V_{NE} (OEI/Power-OFF): .... V_{NE} (Power-ON) - 40 KIAS

V_{NEI} (maximum IFR airspeed) .... V_{NE} (Power-ON) - 20 KIAS

Vmini (minimum IFR airspeed) ....55 KIAS

Maximum airspeed during IFR/VFR approaches...................140 KIAS

Maximum landing gear operating airspeed (V_{LO}) ....140 KIAS

Maximum landing gear extended airspeed (V_{LE})...................140 KIAS

Minimum airspeed in autorotation (without
close external references)...60 KIAS

Maximum airspeed with single AP operational:
— normal flight .... V_{NE} (Power-ON) - 40 KIAS
— in IFR/VFR approaches .... 115 KIAS
— in moderate to high turbulence .... 115 KIAS

Maximum airspeed for searchlight extension, orientation
and retraction ....135 KIAS

GROUND SPEED LIMITATIONS

ON CONCRETE OR EVEN SURFACES

Maximum speed for running take-off and landing....................40 knots

Maximum taxiing speed (nose wheel unlocked):
— straight ....20 knots
— turning..10 knots

ON UNPREPARED OR UNEVEN SURFACES

Maximum speed for running take-off and landing....................20 knots

Maximum taxiing speed (nose wheel unlocked):
— straight ....20 knots
— turning..10 knots

WIND SPEED LIMITATIONS FOR ROTOR STARTING AND STOPPING

The maximum wind speed for rotor starting and stopping is 40 knots from any direction.

ALTITUDE LIMITATIONS

Maximum altitude for take-off and landing ....see Figure 1-4

Minimum and maximum operating altitudes...................see Figure 1-4

Note

For low speed controllability, IGE and OGE hover performance and H-V refer to Section 4.

AMBIENT AIR TEMPERATURE LIMITATIONS

Minimum ambient air temperature ....-25 °C (-13 °F)

Maximum sea level ambient air temperature ....50 °C (122 °F)

The maximum ambient air temperature for operation decreases with pressure altitude at the standard lapse rate of 2 °C (3.6 °F) every 1000 ft (305 m) up to 20000 ft (6096 m).

SLOPE LIMITATIONS

Slope operations are prohibited on surfaces with a slope angle steeper than the following:

From 90° to the left of nose up to 90° to the right of nose up..... 10 deg

Nose down .... 2 deg

AIRSPEED LIMITATION

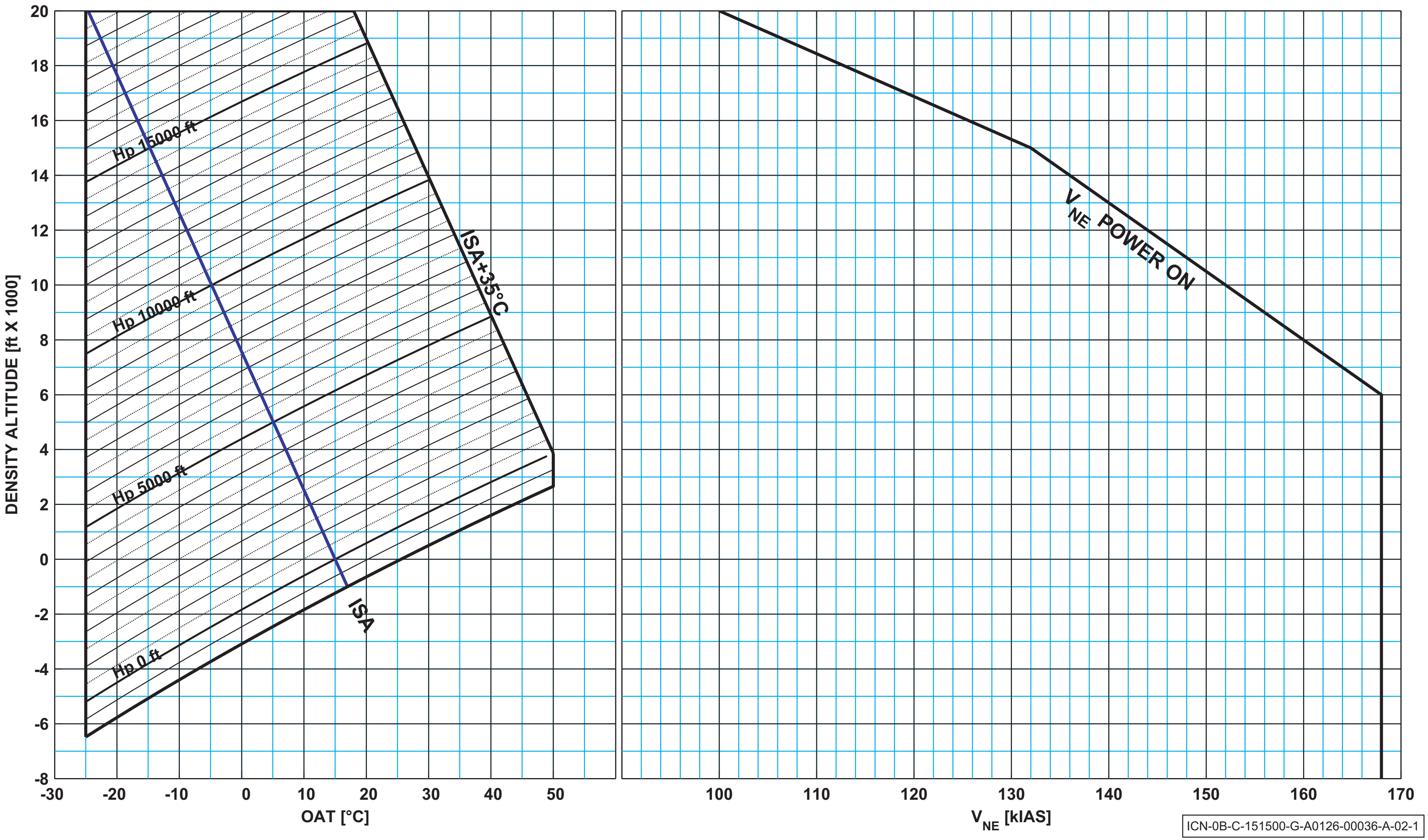

Figure 1-3 Airspeed Limitation V_{NE} (Power-ON)

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FLIGHT and T.O/LANDING ENVELOPE

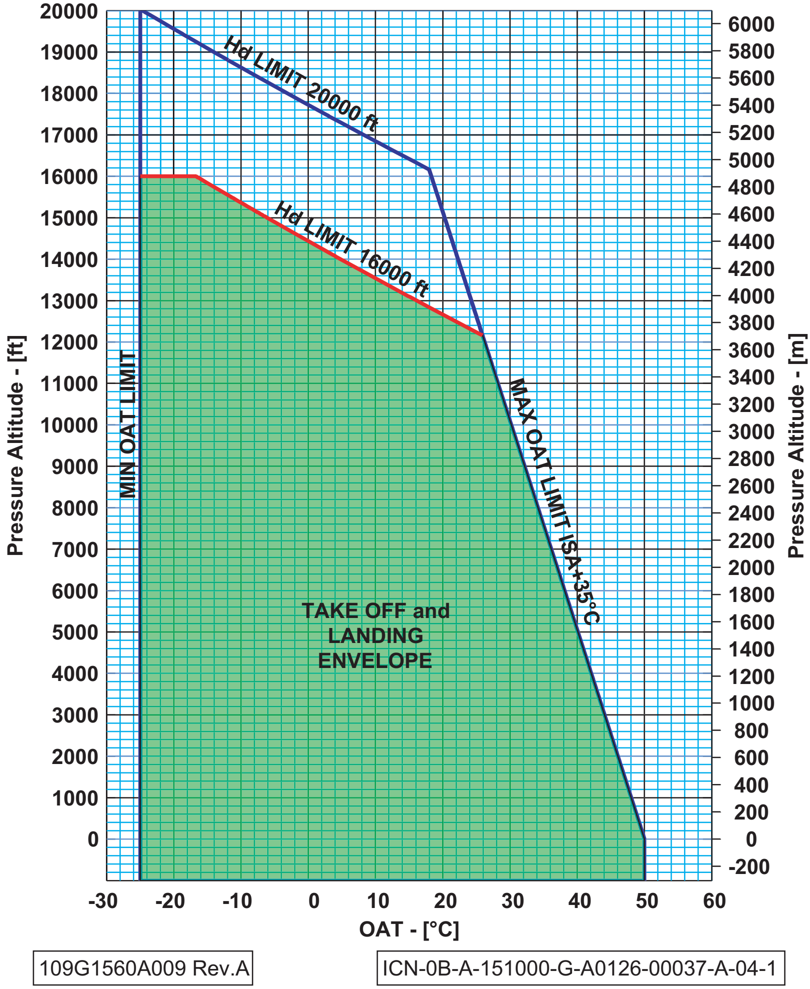


Figure 1-4 Pressure altitude / OAT Envelope

DOORS OPENED OR REMOVED

Flight with either one or both passengers cabin sliding doors open is prohibited if passenger door modification P/N 109-0814-35 is not installed.

It is possible to fly with pilot / copilot doors removed and/or passengers cabin sliding doors opened to lock position or removed in whichever combination.

V_{NE} with any door removed or open to lock position................75 KIAS

Maximum airspeed for passengers cabin sliding doors
opening or closing..50 KIAS

POWER PLANT AND TRANSMISSION LIMITATIONS

GENERAL

Note

The one engine inoperative (OEI) ratings are intended for emergency use only, when one engine becomes inoperative due to an actual malfunction.
OEI operations for maintenance or training purposes shall be limited to the OEI Continuous Operation power range.

Note

Transient range must not be used intentionally.

POWER PLANT LIMITATIONS

The helicopter is powered by Pratt and Whitney PW207C turboshaft engines.

GAS GENERATOR SPEED (N1)

All Engines Operating (AEO)

Continuous Operation .... 50 to 97.1%

Take-Off Range (5 minutes)... 97.2 to 99.7%

Maximum 99.7%

Transient (20 seconds) 104.1%

One Engine Inoperative (OEI)

Continuous Operation .... 50 to 99.7%

2.5 minutes Range .... 99.8 to 102.9%

Maximum 102.9%

Transient (20 seconds) 104.1%

POWER TURBINE SPEED (N2)

All Engines Operating (AEO)

Transient 95%

Minimum..99%

Continuous Operation (except for Take-Off and Landing)...99 to 101%

Take-Off, Landing and below Vy ....101 to 102%

Maximum..102%

Transient (20 seconds)... 112%

One Engine Inoperative (OEI)

Transient 85%

Minimum..90%

Cautionary Range ....90 to 98%

Continuous Operation (except for Take-Off and Landing).. 99 to 101%

Take-Off, Landing and below Vy ....101 to 102%

Maximum..102%

Transient (20 seconds)... 112%

TURBINE OUTLET TEMPERATURE (TOT)

Engine Starting

Maximum (unlimited)... 650 °C

Maximum transient (2 seconds).. 875 °C

Note

A linear variation applies between 20 seconds at 650 °C and 2 seconds at 760 °C.
A transient of 2 seconds applies above 760 °C.

All Engines Operating (AEO)

Maximum Continuous .... 840 °C

Take-Off Range (5 minutes).. 841 to 900 °C

Transient (20 seconds) .... 1000 °C

One Engine Inoperative (OEI)

Maximum Continuous .... 900 °C

2.5 minutes Range.. 901 to 970 °C

Transient (20 seconds) .... 1000 °C

ENGINE OIL PRESSURE

Note

The oil pressure limits vary as a function of the gas generator speed.

Minimum for ground idle...................................... Above lower red line

Continuous Operation ....Green band
(variable as a function of N1)

Cautionary Range ....Yellow bands
(variable as a function of N1)

Maximum during engine starting .... 200 psi

Note

The engine can operate with oil pressure up to 200 psi during or after start or if the oil temperature drops significantly below 71 °C. Oil pressure will decrease as oil temperature increases and is not expected to endure for more than 5 sec. The operation at an oil pressure up to 200 psi is permitted for a period of 10 minutes.

ENGINE OIL TEMPERATURE

Continuous Operation .... 10 to 125 °C

Maximum.. 125 °C

ENGINE STARTER LIMITATIONS

The engine starter duty cycle is the following:

— 45 seconds on, 1 minute off;

— 45 seconds on, 1 minute off;

— 45 seconds on, 30 minutes off.

TRANSMISSION LIMITATIONS

TORQUE (TRQ)

All Engines Operating (AEO)

Maximum Continuous 100%

Take-Off Range (5 minutes) .... 101 to 107%

Maximum 107%

Transient (6 seconds) 110%

One Engine Inoperative (OEI)

Maximum Continuous 133%

2.5 minutes Range .... 134 to 162%

Maximum 162%

Transient (6 seconds) 173%

TRANSMISSION OIL PRESSURE

Minimum .... 30 psi

Continuous Operation .... 30 to 50 psi

Cautionary Range .... 51 to 70 psi

Maximum .... 70 psi

Note

During cold starting conditions, the transmission oil pressure may temporarily rise up to 100 psi.

TRANSMISSION OIL TEMPERATURE

Continuous Operation .... 0 to 120 °C

Maximum .... 120 °C

ROTOR LIMITATIONS (NR)

POWER-ON (AEO)

Transient 95%

Minimum...99%

Continuous Operation (except for Take-Off and Landing)...99 to 101%

Take-Off, Landing and below Vy ....101 to 102%

Maximum..102%

POWER-ON (OEI)

Transient 85%

Minimum 90%

Cautionary Range ....90 to 98%

Continuous Operation (except for Take-Off and Landing)...99 to 101%

Take-Off, Landing and below Vy ....101 to 102%

Maximum..102%

POWER-OFF

Transient 90%

Minimum...95%

Continuous Operation ....95 to 110%

Maximum...110%

Note

Transient range must not be used intentionally.

FUEL SYSTEM LIMITATIONS

FUEL PRESSURE

Cautionary Range ....0 to 7 psi

Continuous Operation ....8 to 25 psi

Maximum ....25 psi

FUEL QUANTITY

Total .... 460 kg

(575 liters at 0.8 kg/liter)

TOTAL UNUSABLE FUEL

In coordinated flight... 10 kg

(12 liters at 0.8 kg/liter)

FUEL TYPES

The fuels shown in Table 1-1 have been authorized for use with the Pratt & Whitney PW207C engines.

Table 1-1 Authorized Fuels

Kerosene Type	Applicable Specification
JET A	ASTM D1655
JET A-1	ASTM D1655
JP-5 *	MIL-T-5624
JP-8 *	MIL-T-83133
R.T.	GOST 10227-86
R.T.	GSTU 320.00149943.007-97
TS-1**	GOST 10227-86
TS-1**	GSTU 320.00149943.011-99

Note

An approved fuel or any mixture of acceptable fuels may be used. However, changing to a fuel with a substantially different heating value or specific gravity may require maintenance in the form of engine fuel control (trimmer) adjustment. Refer to the Pratt & Whitney PW207C engine manual.

Note

Fuel grades marked with an asterisk (*) contain a fuel system icing inhibitor (FSII). For JP-8, MIL-T-83133C allows two grades. The grade meeting NATO code F-34 has FSII while the grade meeting code F-35 has no FSII unless specifically requested.

Note

Fuel grades marked with double asterisk (**) are for occasional use only. Refer to Pratt & Whitney PW207C Maintenence Manual for full details and restrictions.

Note

For operations below +4°C, the use of anti-ice additive is authorized but not mandatory since the helicopter is equipped with an air-frame anti-ice fuel filter. For additive require-ments and blending procedures refer to the Pratt & Whitney PW207C engine manual.

EMERGENCY FUELS

Table 1-2 presents a list of fuels approved for emergency use. Emergency fuels should only be used whenever any of the authorized fuels are not available.

Refer to the Pratt & Whitney PW207C Manual for full details.

Table 1-2 Emergency Fuels

Fuel Type	Applicable Specification	Restrictions
Automotive Diesel CPW 46	Arctic Grade	Do not use below -15 °C (5 °F) OAT
	Winter Grade	Do not use below -7 °C (20 °F) OAT
	Regular Grade	Do not use below 5 °C (40 °F) OAT
Aviation Gasoline Grades: 80, 100 and 100 LL	MIL-G-5572	Do not use for longer than 150 hours during any period between engine overhauls. Engine operation in suction mode is prohibited.

LUBRICANTS

AUTHORIZED ENGINE OILS

The oils shown in Table 1-3 have been authorized for use with the PW207C engines.

Table 1-3 Authorized Lubricating Oils, Type II (5 Centistokes)

Applicable Specification	Brand Names (For reference only)
MIL-PRF-23699	Aero-Shell Turbine Oil 500
	Aero-Shell Turbine Oil 560
	BP Turbo Oil 2380 (formerly Exxon Turbo Oil 2380)
	BP Turbo Oil 25 (formerly Exxon Turbo Oil 25)
	Castrol 5000
	Mobil Jet Oil II
	Royco Turbine Oil 500
	Royco Turbine Oil 560
	Turbonycoil 525-2A

Note

Engine oil tank capacity is 5.12 liters (1.35 US Gallon).

Note

Mixing of the different brands and types is prohibited.

AUTHORIZED TRANSMISSION OILS

The oils shown in Table 1-4 have been authorized for use with the main transmission and the tail rotor gearbox lubrication systems.

Table 1-4 Authorized Lubricating Oils

Designation	Specification
BP Turbo oil 2380 (formerly EXXON Turbo oil 2380)	MIL-PRF-23699
Mobil Jet Oil II	MIL-PRF-23699
Mobil Jet Oil 254	MIL-PRF-23699
Aeroshell Turbine oil 500	MIL-PRF-23699
Aeroshell Turbine oil 555	DOD-PRF-85734
Aeroshell Turbine oil 560	MIL-PRF-23699
Castrol Aero 5000	MIL-PRF-23699
BP Turbo oil 2197	MIL-PRF-23699

Oils are limited to ambient temperatures above -40 °C (-40 °F).

Note

Mixing of oils of different brands and types is prohibited.

HYDRAULIC SYSTEM LIMITATIONS

MAIN SYSTEM FLUID PRESSURE

Minimum... 1200 psi

Cautionary Range .... 1200 to 1400 psi

Continuous Operation .... 1401 to 1600 psi

Maximum... 1600 psi

NORMAL UTILITY SYSTEM - FLUID PRESSURE

Minimum... 500 psi

Cautionary Range ....500 to 1140 psi

Continuous Operation ....1141 to 1600 psi

Maximum... 1600 psi

EMERGENCY UTILITY SYSTEM - FLUID PRESSURE

Minimum..1140 psi

Maximum... 1600 psi

APPROVED FLUIDS

The following hydraulic fluids are approved:

MIL-PRF-5606
MIL-PRF-83282.

Note

Mixing of fluids of different brands and types is prohibited.

GENERATOR LOAD LIMITATIONS

For each generator

Up to 15000 ft Hp

— Continuous Operation: ....0 to 200 A

— Maximum load at MPOG (no time limitiation):..................160 A

— Maximum load at MPOG (10 minutes):.................160 to 200 A

Above 15000 ft Hp

— Continuous Operation: ....0 to 130 A

— Maximum load at MPOG (no time limitiation):..................130 A

Transient at starting (45 seconds):...300 A

NAVIGATION LIMITATIONS

MAGNETIC COMPASS

When operating the SRCH LT, the magnetic compass is unreliable.

FLIGHT MANAGEMENT SYSTEM LIMITATIONS

Operations in European B-RNAV airspace are demonstrated in accordance to EASA AMC 20-4, where GPS stand-alone equipment is used as the only means for B-RNAV operations. This does not constitute an operational approval.

It is prohibited to navigate using the Flight Management System in Dead Reckoning mode.

Note

The Dead Reckoning mode will be automatically activated as the navigation mode only if the GPS systems have failed. The accuracy of the Dead Reckoning mode cannot be guaranteed.

IFR en route and terminal navigations are prohibited unless the pilot verifies the currency of the navigation database or verifies each selected waypoint for accuracy by referring to current approved data.

IFR approaches and departures must be carried out in accordance with approved instrumental procedures that are retrieved from the FMS approved database. The FMS database must incorporate the current update cycle.

EN ROUTE AND TERMINAL

It is prohibited to use VNAV mode data for navigation.

Traditional IFR approved navigation equipment shall be available to continue the flight including the alternate when GPS integrity is lost (GPS LOI and/or GPS LON displayed on IDU caution list).
The ground based aids on the route to be flown or ground based aids for RNAV routes must be operational and aircraft equipment other than GPS suitable for the route to be flown must be serviceable.

PRECISION APPROACH

It is prohibited to use FMS-GPS for IFR precision approaches.

NON PRECISION APPROACH (NPA)

It is prohibited to perform an IFR approach after the Final Approach Fix if GPS LOI caution is displayed on IDU caution list. If GPS LOI caution is due to a GPS failure, carry out the GPS FAILURE malfunction procedure, or initiate a missed approach.

For EFIS with software version A109 7.0D

It is prohibited to continue a Non Precision Approach procedures following a PFD failure.

EFIS LIMITATIONS

Skyway simbology does not provide precision approach guidance nor it does guarantee terrain separation.
It remains the pilot's responsibility to provide self separation from terrain.

HTAWS LIMITATIONS

The TAWS installed is a class B HTAWS and it provides the following functions:

- Terrain and Obstruction display;
- Forward Looking Terrain Awareness (FLTA);
- Sink Rate after Takeoff or Missed Approach (GPWS Mode 3).

HTAWS must not be used for navigation.
Do not attempt to navigate using the terrain depiction.

HTAWS must not be used for terrain following flight.
Do not use hue nor grid as cues for altitude or direction.

The HTAWS database installed shall be the last update for the region being flown.

CAUTION

The terrain and obstacle display is intended to serve as a terrain and obstacle awareness tool only. The display and database may not provide the accuracy and fidelity on which to base routine navigation decisions and plan routes to avoid terrain or obstacles.

CAUTION

There are many towers , antennas, power lines, and obstructions that are not in the database.

AUTOMATIC FLIGHT CONTROL SYSTEM LIMITATIONS

Minimum AFCS configuration for IFR / VFR flight...2 AP in ATT mode

DUAL AP OPERATION

Flight at altitude below 1000 ft AGL ....Fly attentive

Flight at airspeed above V_{NE} (Power-ON) - 20 KIAS........Fly manually

SINGLE AP OPERATION

Flight at altitude above 500 ft AGL ....Fly attentive

Flight at altitude at or below 500 ft AGL ....Fly manually

Flight in moderate to high turbulence................................Fly manually

In single AP operation, the AFCS Upper Modes and Flight Director Modes must not be used unless conducting an approach, missed approach, transition to/from the hover or hover.

FLIGHT DIRECTOR LIMITATIONS

UPPER AND FLIGHT DIRECTOR MODES ENGAGEMENT LIMITS AND MINIMUM USE HEIGHT (MUH)

Hold Mode	Applicable Range	MUH	Notes
ALT	0 KIAS to Vne Altitude -1000 ft to 20000 ft	300ft AGL in cruise or 50 ft AGL in hover	
RHT	0 KIAS to Vne 15 ft to 2500 ft AGL	200 ft AGL in cruise (over flat surfaces only) or 50 ft AGL in hover and transition to/from hover	
HDG	0 KIAS to Vne	200 ft AGL in cruise or 50 ft AGL during approach in hover and transition to/from hover	
ALTA*	40 KIAS to Vne Altitude -1000 ft to 20000 ft	300 ft AGL	
VS*	40 KIAS to Vne within -1500 fpm and 2000 fpm	200 ft AGL	**3**
IAS*	40 KIAS to Vne	200 ft AGL in cruise or 50 ft AGL during approach in hover and transition to/from hover	
HOV	Airspeed less than 80 KIAS and Groundspeed — Longitudinal -20 to +60 kts — Lateral -20 to +20 kts	200 ft AGL in cruise or 50 ft AGL during approach in hover and transition to/from hover	
GA	40 KIAS to Vne	N/A	**1**

UPPER AND FLIGHT DIRECTOR MODES ENGAGEMENT LIMITS AND MINIMUM USE HEIGHT (MUH) (CONT.D)

Hold Mode	Applicable Range	MUH	Notes
TU	Airspeed less than 40 KIAS 10 ft to 2500 ft AGL	200 ft AGL in cruise or 50 ft AGL during approach in hover and transition to/from hover	**2**
NAV	40 KIAS to Vne	200 ft AGL in cruise or 50 ft AGL during approach in hover and transition to/from hover	
VOR	40 KIAS to Vne	200 ft AGL in cruise or 50 ft AGL during approach in hover and transition to/from hover	
VAPP	40 KIAS to Vne	200 ft AGL in cruise or 50 ft AGL during approach in hover and transition to/from hover	
LOC/ NLOC	40 KIAS to Vne	200 ft AGL in cruise or 50 ft AGL during approach in hover and transition to/from hover	
GS/ NGS	40 KIAS to Vne	200 ft AGL in cruise or 50 ft AGL during approach in hover and transition to/from hover	

Notes

1 GA mode engagement is inhibited (regardless of airspeed value) if HOV mode engaged.

2 TU mode engages also for airspeed less than 40 kts if HOV mode engaged.

3 VS Mode engagement above 2000 fpm or below -1500 fpm will results in the mode returning the aircraft to the maximum rates quoted (2000 fpm or -1500 fpm).

GENERAL LIMITATIONS

With AFCS Upper Modes and Flight Director Modes engaged the torque limiter must be active (LIMITER ON advisory message displayed).

ILS APPROACH

Steepest ILS approach gradient... 7,5 deg
Maximum Airspeed according to the following table:

GLIDE PATH ANGLE (deg)	AIRSPEED (KIAS)
3	140
4	120
5	110
6	100
7,5	90

GPS APPROACH

Steepest GPS approach gradient .... 9,0 deg

Maximun Airspeed according to the following table

GLIDE PATH ANGLE (deg)	AIRSPEED (KIAS)
3	140
4	120
5	110
6	100
7,5	90
8	80
9	70

UPPER MODES

HOV Mode

HDG datum shall not be changed with variation greater than ±10° when HOV mode is engaged with relative wind speed and/or non-zero ground-speed reference.

For EFIS with software version A109 7.0D

HOV mode is prohibited if the Magnetic Variation is in excess of 5° [5°W, 5°E].

RHT Mode

The use of Radar Height (RHT) mode is prohibited in cruise over non-flat surfaces.

When in RHT mode the RadAlt must be continuously cross-checked with BarAlt set to QFE.

TU Mode

Safe operation of TU mode has been demonstrated up to 25 kts wind speed coming from ±90 deg with respect to rotorcraft heading.

For EFIS with software version A109 7.0D

TU mode is prohibited if the Magnetic Variation is in excess of 5° [5°W, 5°E].

NLOC and NGS Modes

For EFIS with software version A109 7.0D

NLOC and NGS modes are prohibited, either coupled or uncoupled, if the Magnetic Variation is in excess of 2° [2°W, 2°E].

LOC and GS Modes

For EFIS with software version A109 7.0D

LOC and GS modes are prohibited, either coupled or uncoupled, if the Magnetic Variation is in excess of 2° [2°W, 2°E].

ROTOR BRAKE LIMITATION

Selection of the rotor brake to the ON position is limited as follows:

Rotor Speed (NR) ....below 40%

MISCELLANEOUS LIMITATIONS

Pilot(s) must not use polarized type sun glasses.

FUEL MANAGEMENT SYSTEM

It is prohibited to rely upon Fuel Flow data for flight planning and navigation.

PASSENGERS CABIN LIMITATIONS

Baggage storage under the seat is prohibited.

BAGGAGE COMPARTMENT LIMITATIONS

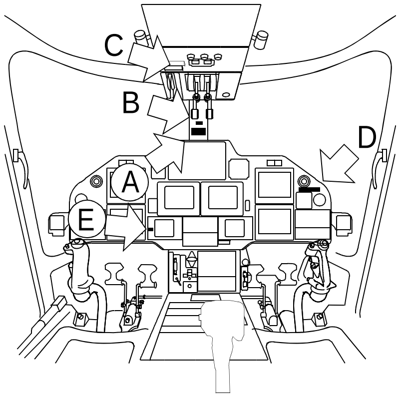
Maximum load.. 120 kg (264 lb)

Maximum Unit Load... 500 kg/m^2 (102 lbs/ft^2)

Note

Refer to Section 6, **Weight and Balance**, for load distribution.

PLACARDS



A

AIRSPEED LIMITATIONS V_{ne} KIAS									
OAT°C / Hp-ft	-25	-20	-10	0	10	20	30	40	50
-1000	168	168	168	168	168	168	168	168	168
0	168	168	168	168	168	168	168	168	168
2000	168	168	168	168	168	168	168	168	-
4000	168	168	168	168	168	168	166	161	-
6000	168	168	168	168	165	160	156	-	-
8000	168	168	164	160	155	151	146	-	-
10000	162	159	154	150	145	141	137	-	-
12000	152	149	145	140	135	131	-	-	-
14000	142	140	135	129	122	115	-	-	-
16000	132	128	121	113	107	-	-	-	-
18000	116	112	105	-	-	-	-	-	-
20000	100	-	-	-	-	-	-	-	-
V_{nei} :					REDUCE V_{ne} BY 20 KIAS				
AUTOROTATION OR OEI:					REDUCE V_{ne} BY 40 KIAS				
MAX FWD TOUCHDOWN SPEED: 40 Kts					V_{LO} AND V_{LE} : 140 KIAS				

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Figure 1-5 Airspeed (Vne) Placard

B

APPROVED TYPES OF OPERATION
DAY/NIGHT VFR
DAY/NIGHT IFR
ICING PROHIBITED

C

DO NOT APPLY ROTOR
BRAKE ABOVE 40%
ROTOR RPM

D

ALT. STATIC
DECREASE ALTIMETER READING BY 120 ft

E

IN ALTN POSITION
OPEN VENTS AND
TURN HEATER/ECS OFF

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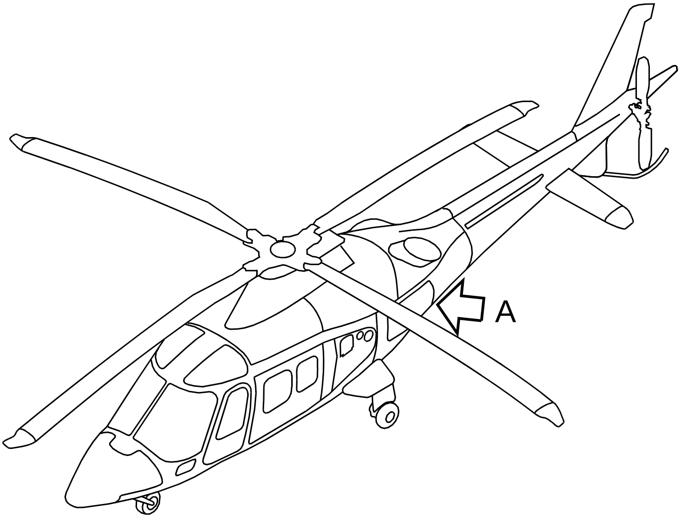
Figure 1-6 Cockpit Placards



A BAGGAGE COMPARTMENT

MAXIMUM ALLOWABLE BAGGAGE LOAD					
ZONE	1	2	3	4	5
KG (LB)	120 (264)	110 (242)	102 (224)	93 (205)	84 (185)
MAXIMUM TOTAL LOAD 120 KG (264 LB) MAX UNIT FLOOR LOADING 500 KG/M2(102 LB/SQ.FT.) ALL CARGO MUST BE SECURED MAX LOAD PER TIE DOWN RING 91KG (200LB)					

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Figure 1-7 Placards in Baggage Compartment

INSTRUMENT MARKINGS

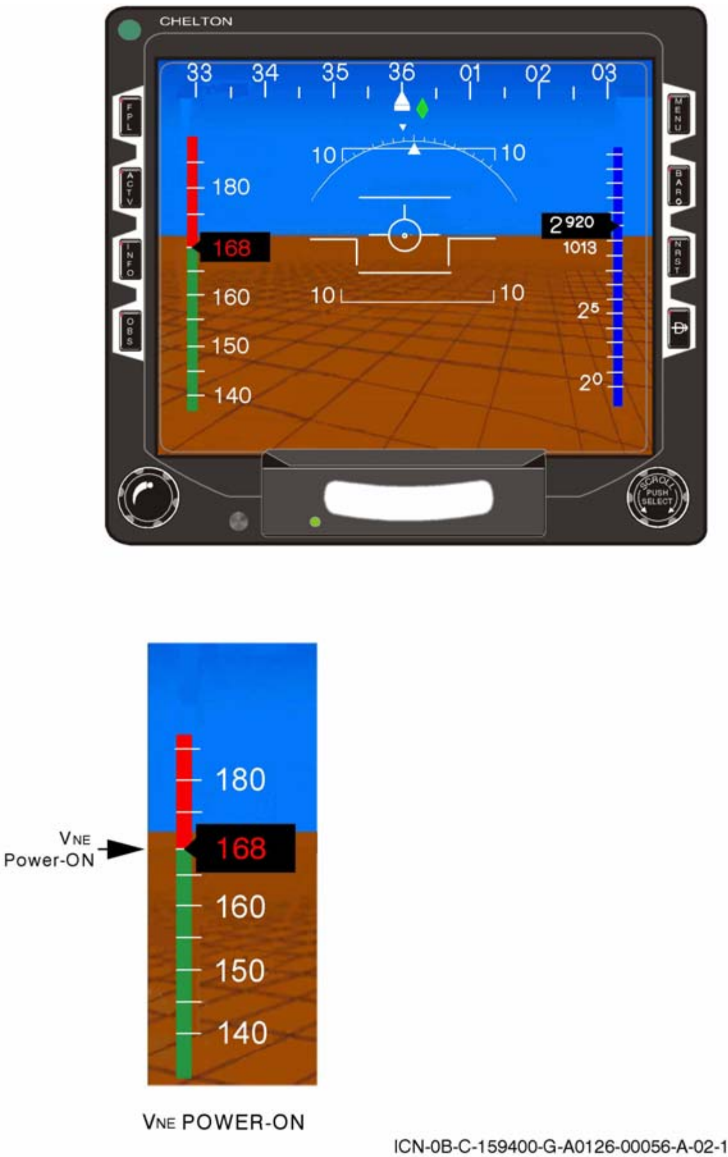

Figure 1-8 EFIS - Indicated Airspeed (IAS)

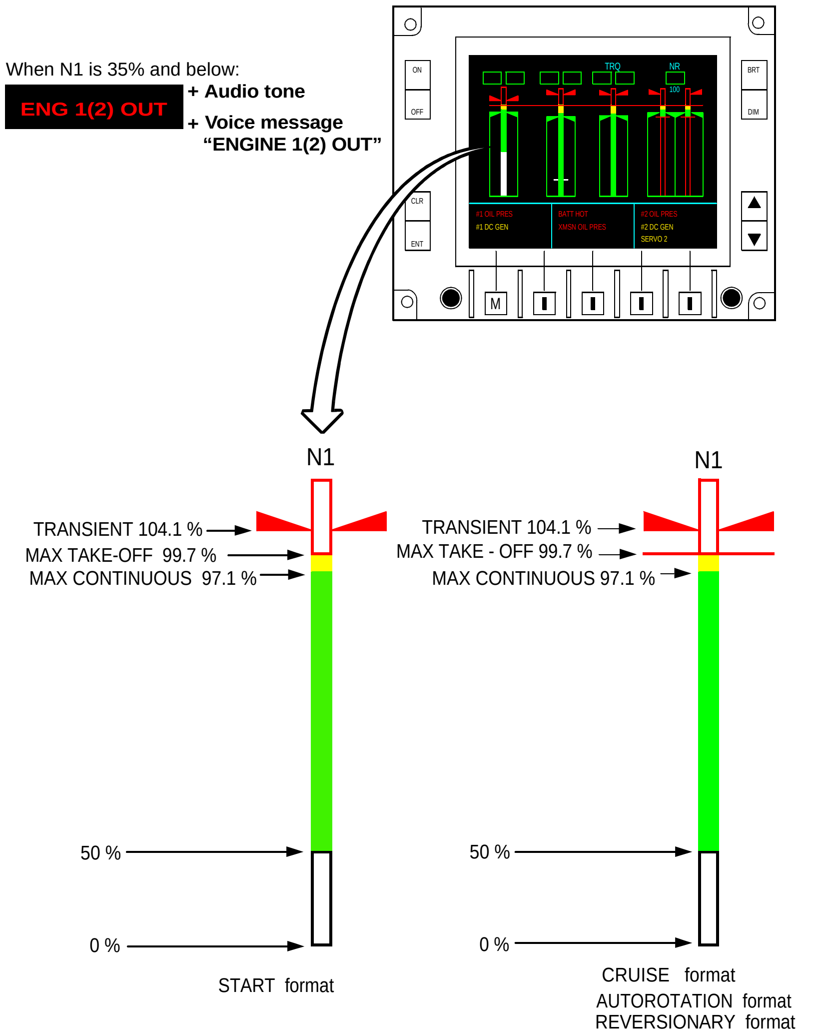

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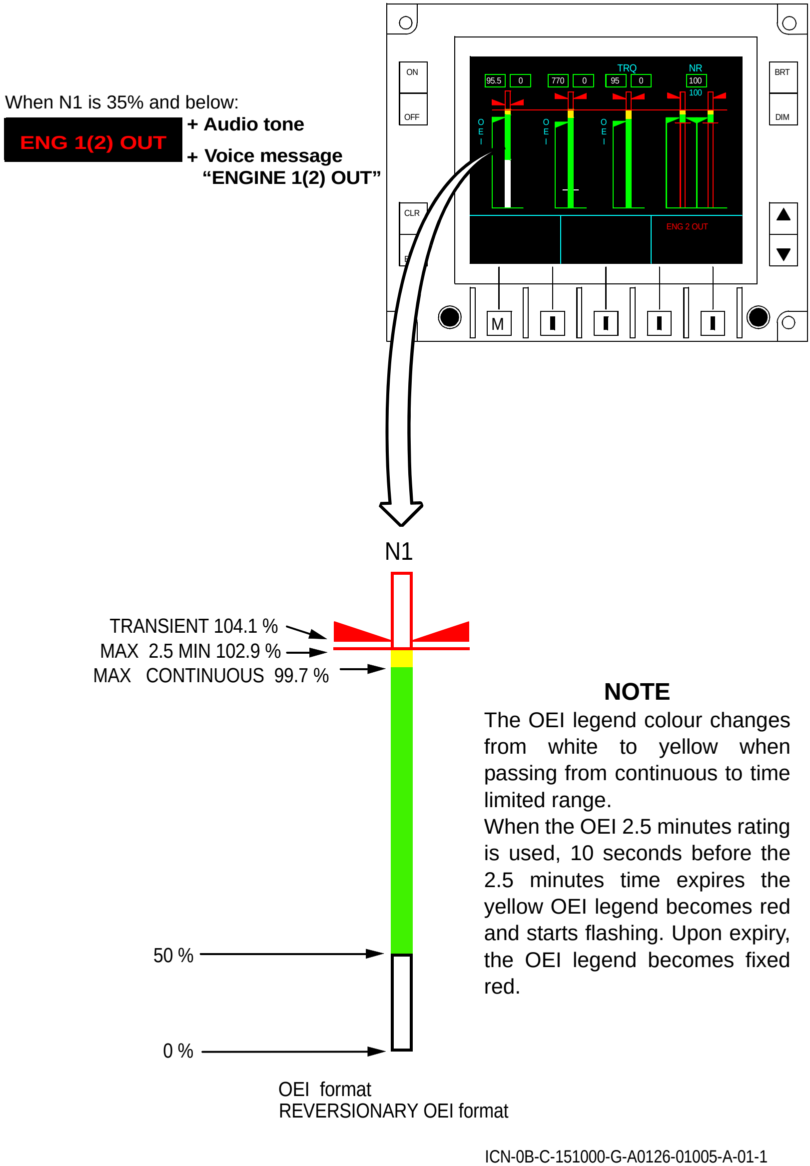
Figure 1-9 EDU1 - Gas Generator Speed (AEO)



NOTE

The OEI legend colour changes from white to yellow when passing from continuous to time limited range.
When the OEI 2.5 minutes rating is used, 10 seconds before the 2.5 minutes time expires the yellow OEI legend becomes red and starts flashing. Upon expiry, the OEI legend becomes fixed red.

Figure 1-10 EDU1 - Gas Generator Speed (OEI)

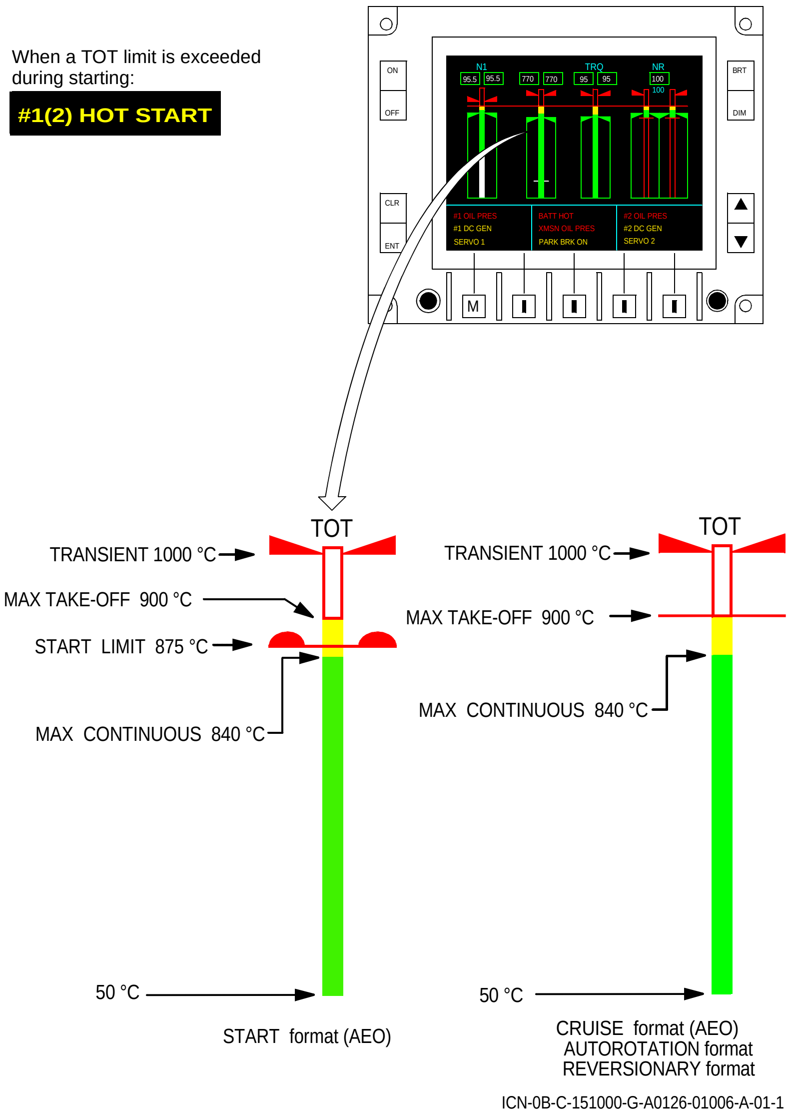
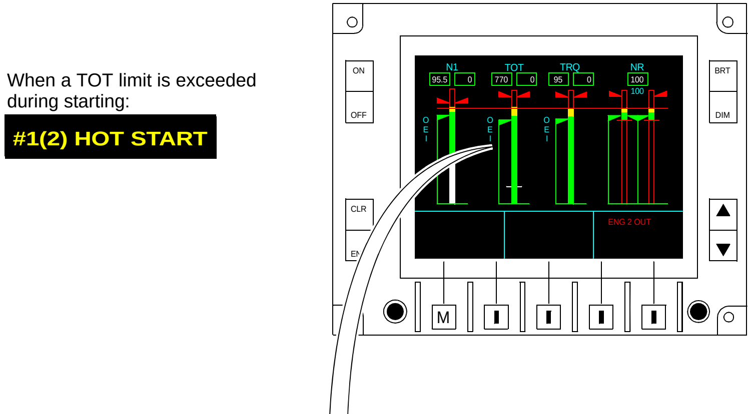
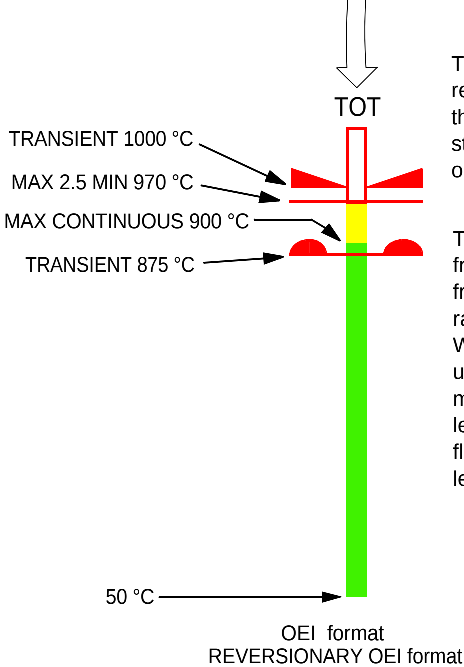


Figure 1-11 EDU1 - Turbine Outlet Temperature (AEO)





NOTE

The transient value at starting (half red dot at 875°C) is displayed on the scale of the engine being started, during the restart phase only.

NOTE

The OEI legend colour changes from white to yellow when passing from continuous to time limited range.
When the OEI 2.5 minutes rating is used, 10 seconds before the 2.5 minutes time expires the yellow OEI legend becomes red and starts flashing. Upon expiry, the OEI legend becomes fixed red.

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Figure 1-12 EDU1 - Turbine Outlet Temperature (OEI)

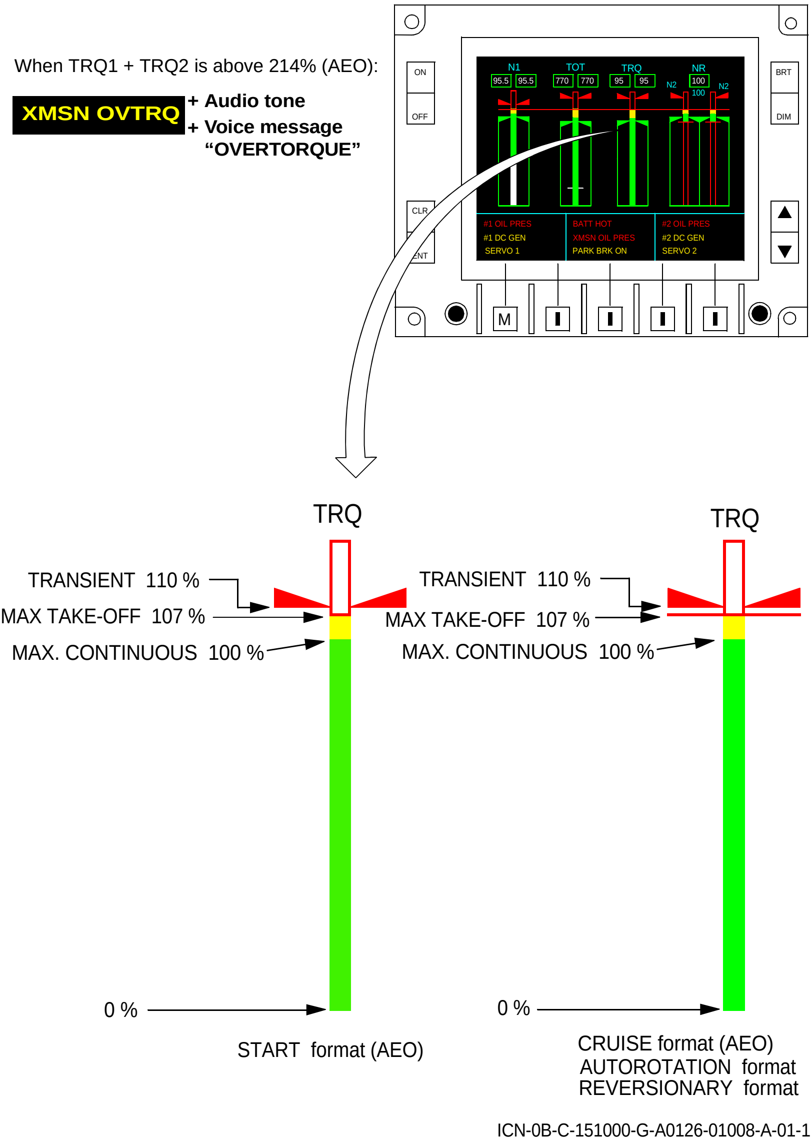
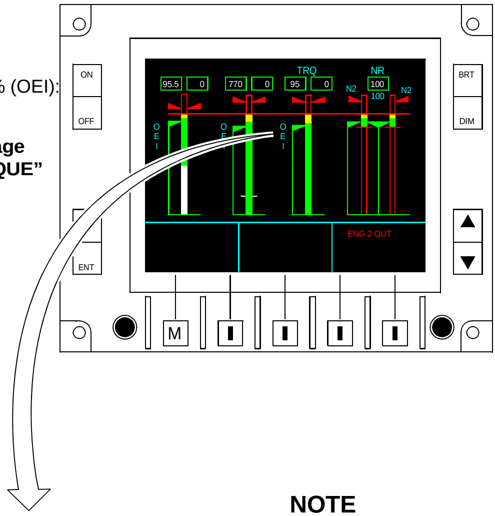
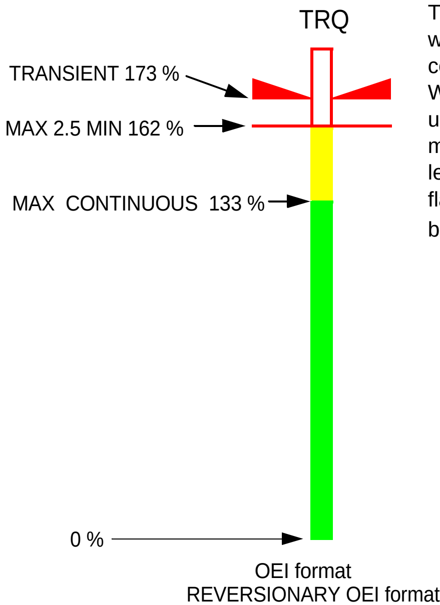


Figure 1-13 EDU1 - Torque (AEO)

When TRQ1 or TRQ2 is above 162% (OEI):

XMSN OVTRQ **+ Audio tone**
+ Voice message "OVERTORQUE"





NOTE

The OEI legend colour changes from white to yellow when passing from continuous to time limited range.
When the OEI 2.5 minutes rating is used, 10 seconds before the 2.5 minutes time expires the yellow OEI legend becomes red and starts flashing. Upon expiry, the OEI legend becomes fixed red.

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Figure 1-14 EDU 1 - Torque (OEI)

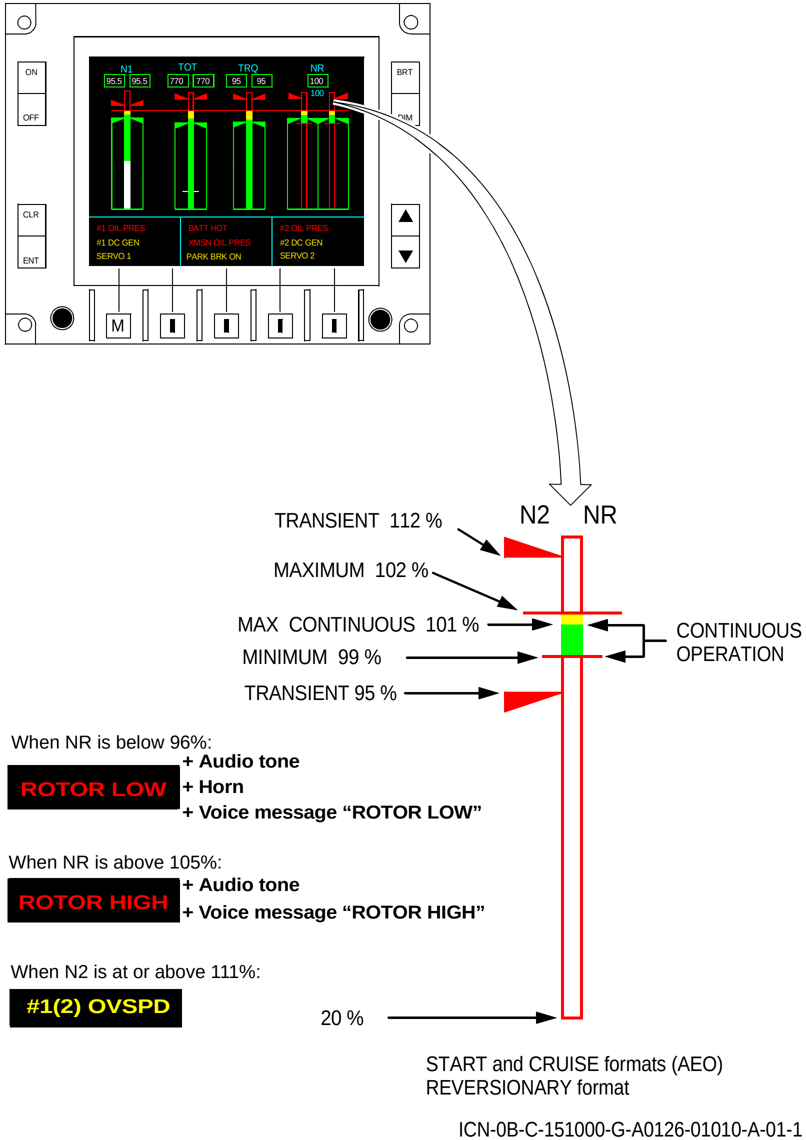

Figure 1-15 EDU1 - Rotor / Power Turbine Speed (AEO)

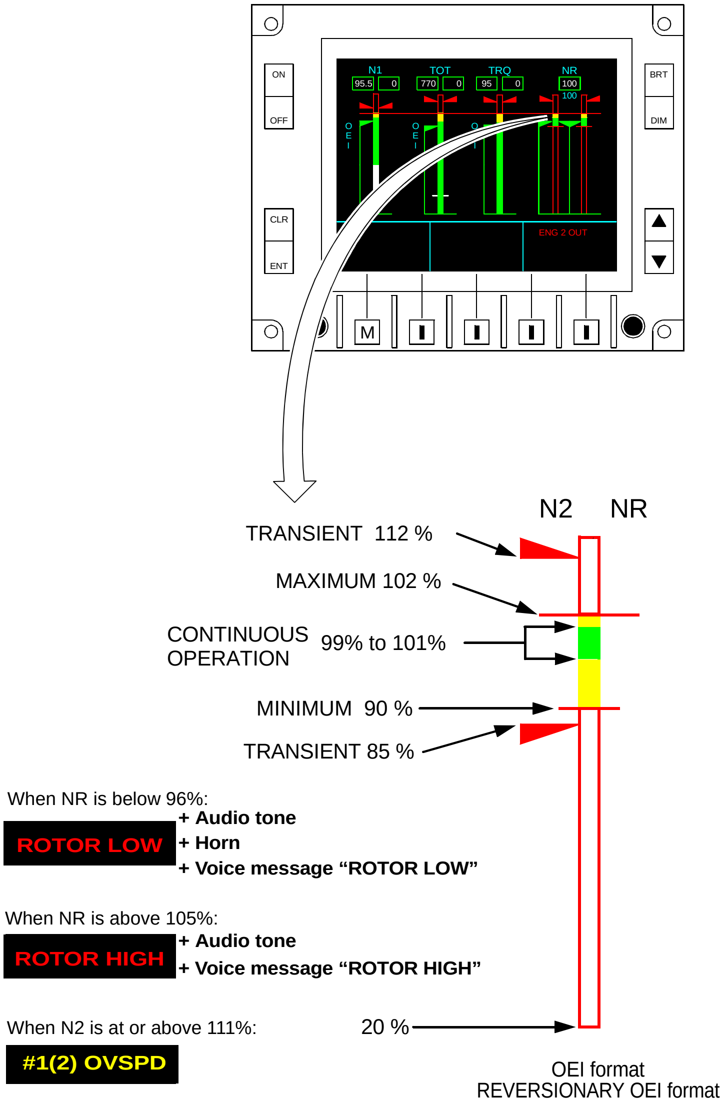

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Figure 1-16 EDU1 - Rotor / Power Turbine Speed (OEI)

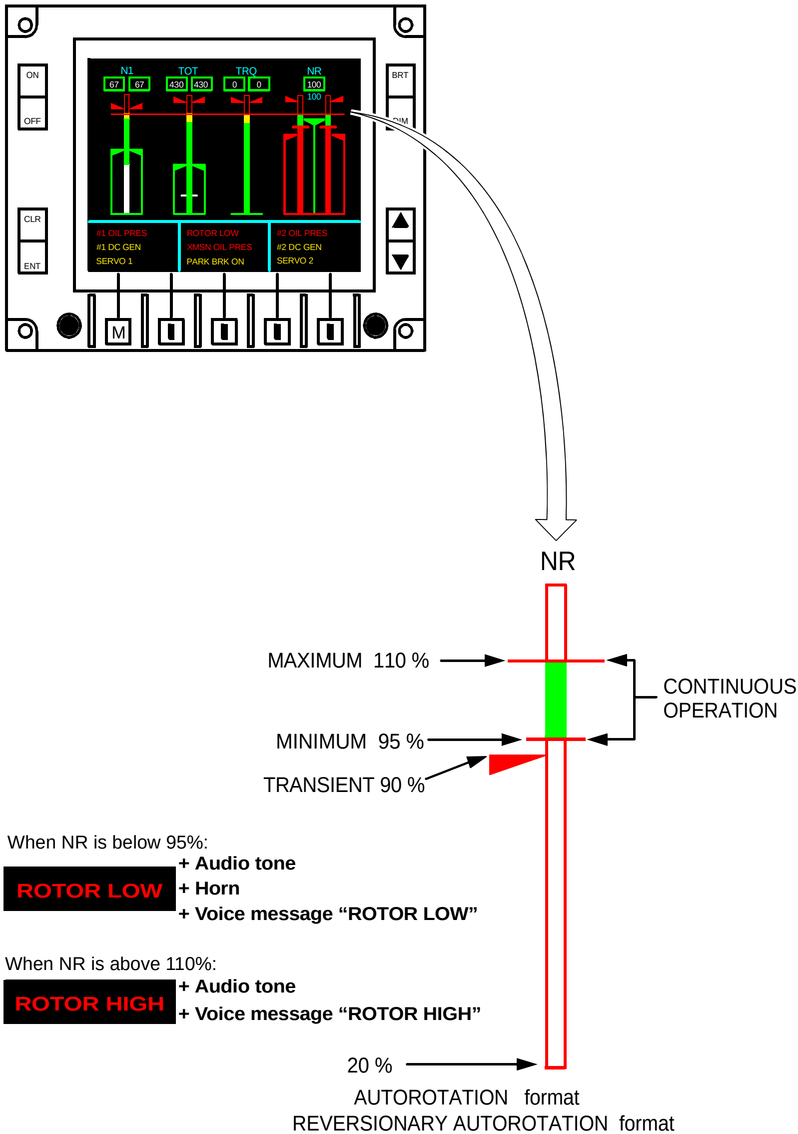

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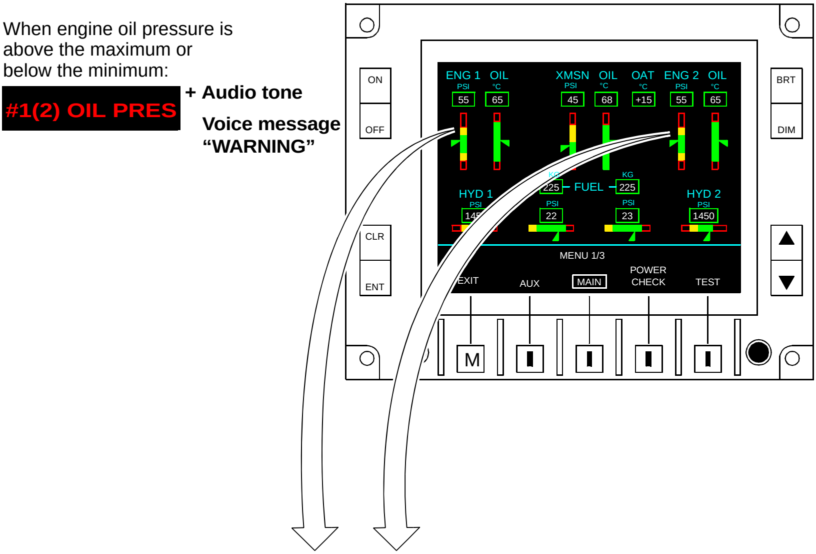
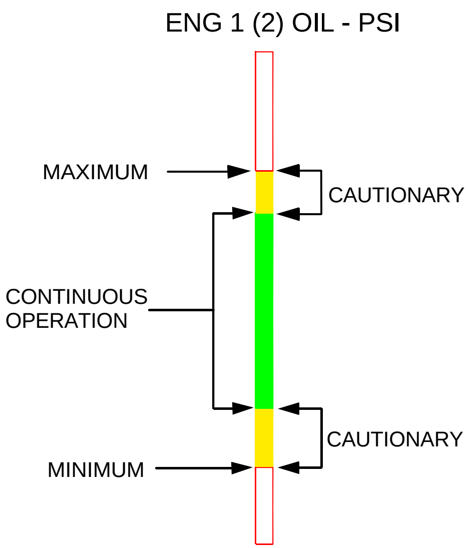
Figure 1-17 EDU1 - Rotor Speed (Power-OFF)





NOTE

Engine oil pressure limits vary as a function of N1.

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Figure 1-18 EDU2 (MAIN) - Engine Oil Pressure

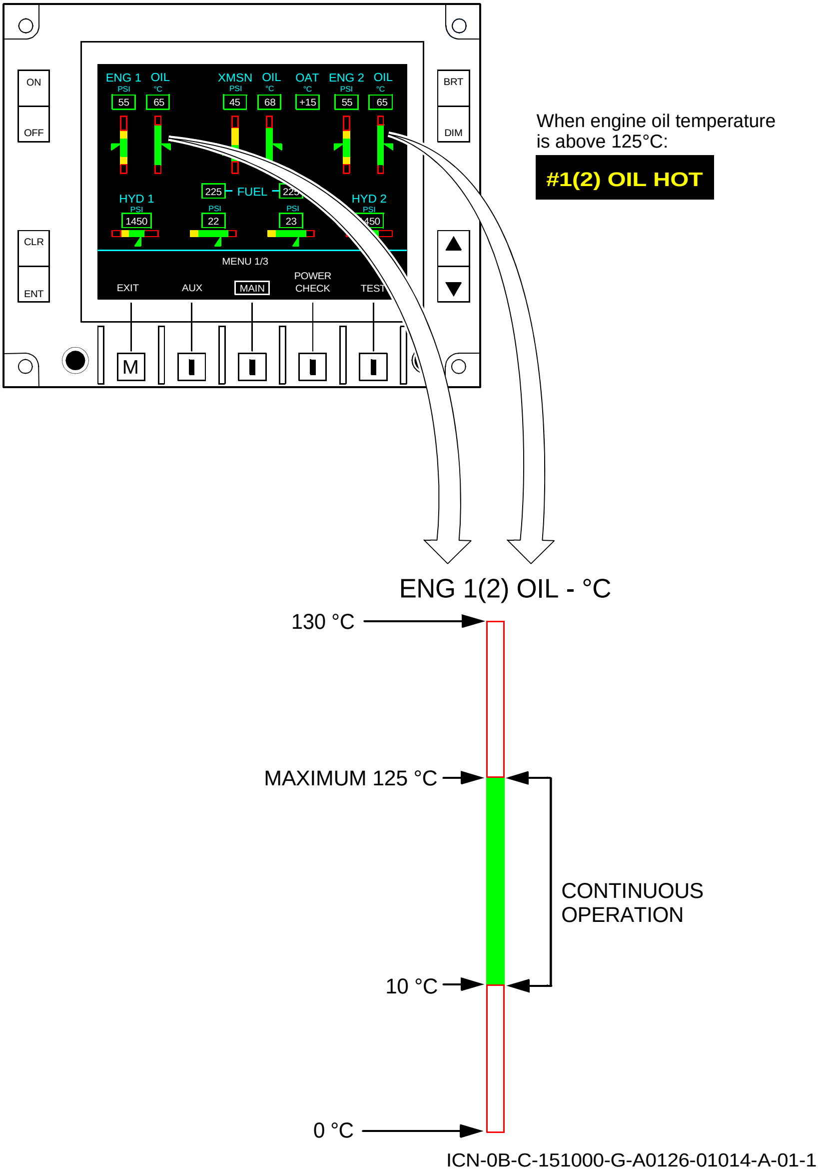

Figure 1-19 EDU2 (MAIN) - Engine Oil Temperature

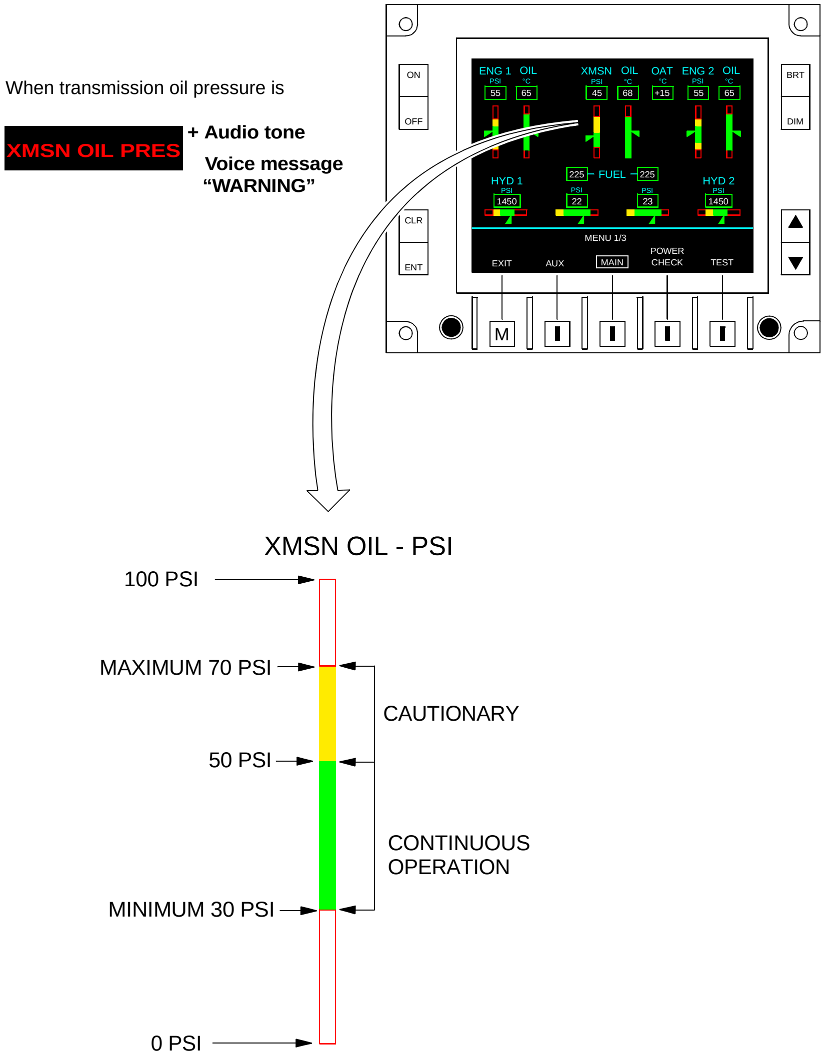

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Figure 1-20 EDU2 (MAIN) - Transmission Oil Pressure

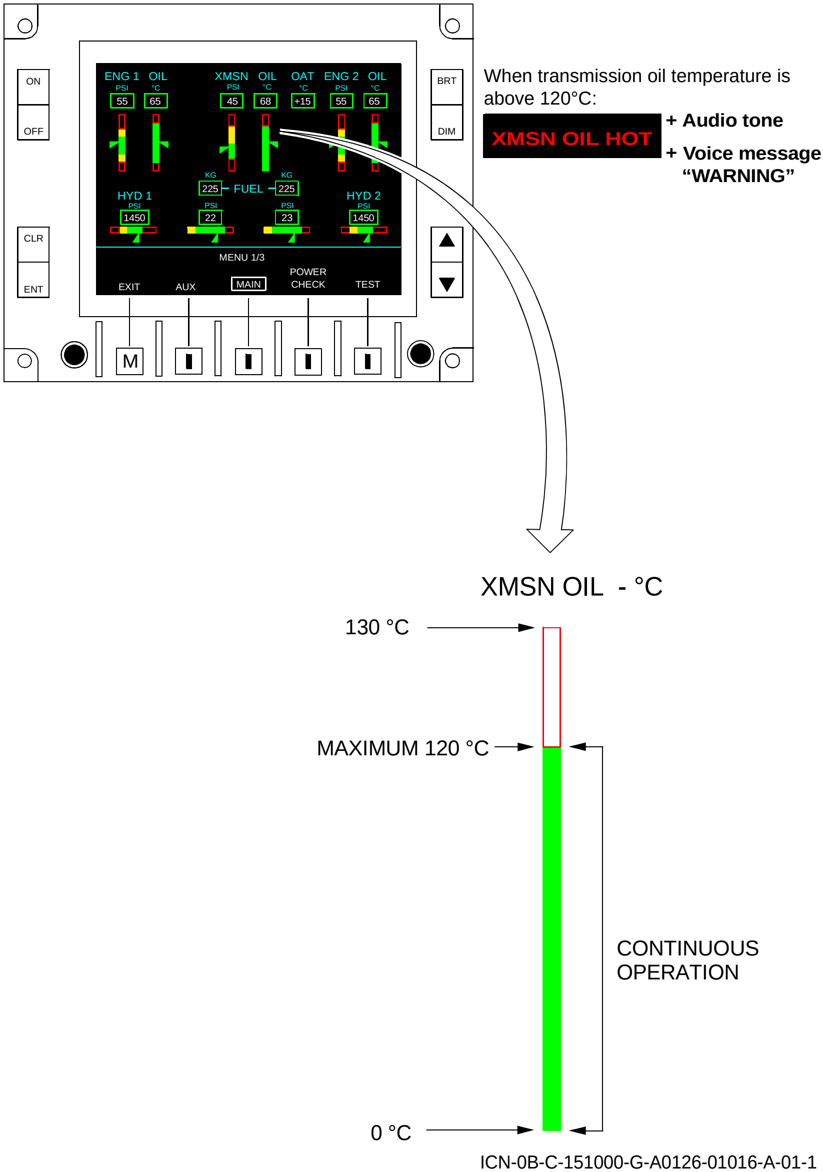

Figure 1-21 EDU2 (MAIN) - Transmission Oil Temperature

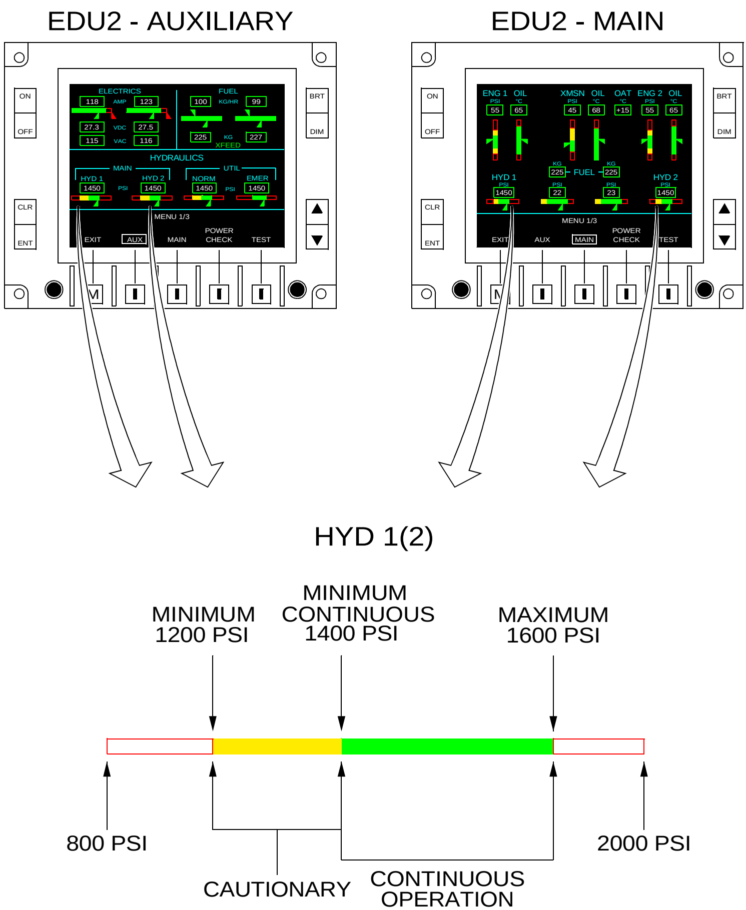

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Figure 1-22 EDU2 (MAIN / AUXILIARY) - Main Hydraulic System Pressure

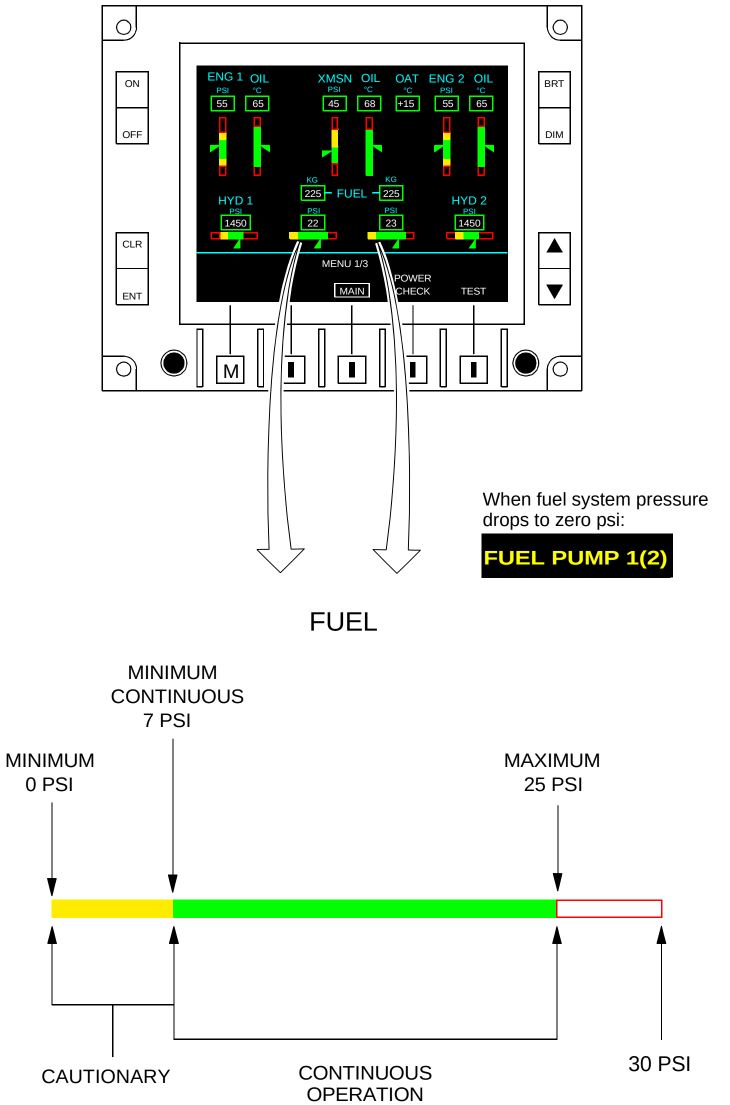

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Figure 1-23 EDU2 (MAIN) - Fuel Pressure

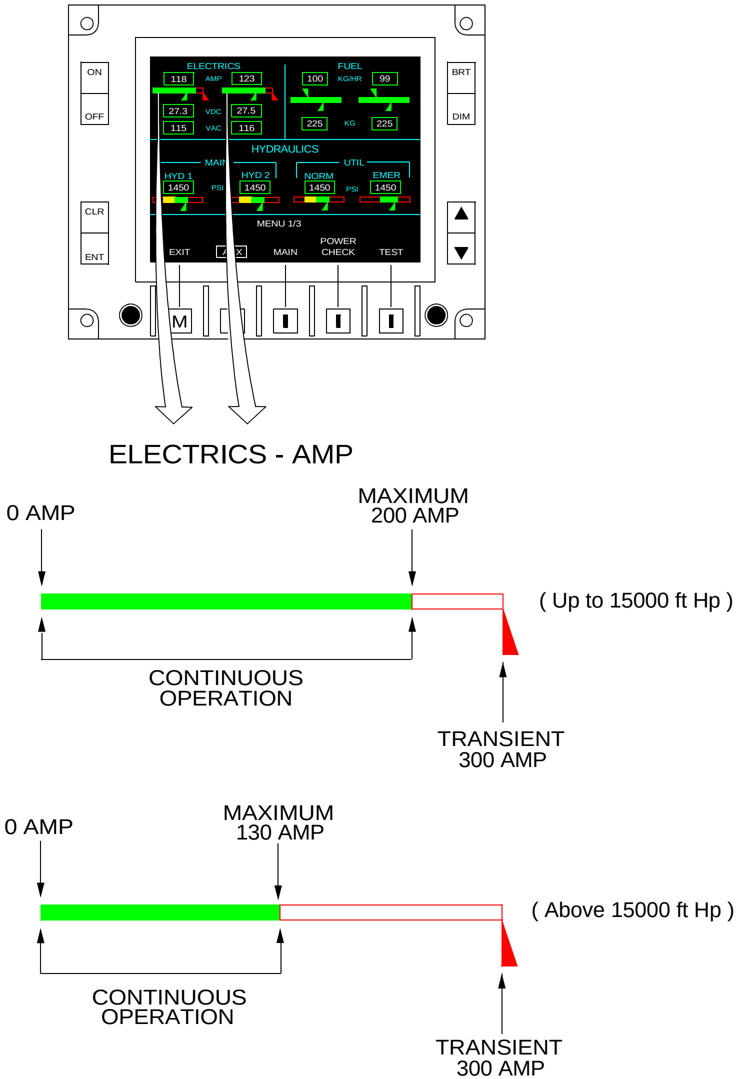

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Figure 1-24 EDU2 (AUXILIARY) - Generator Load

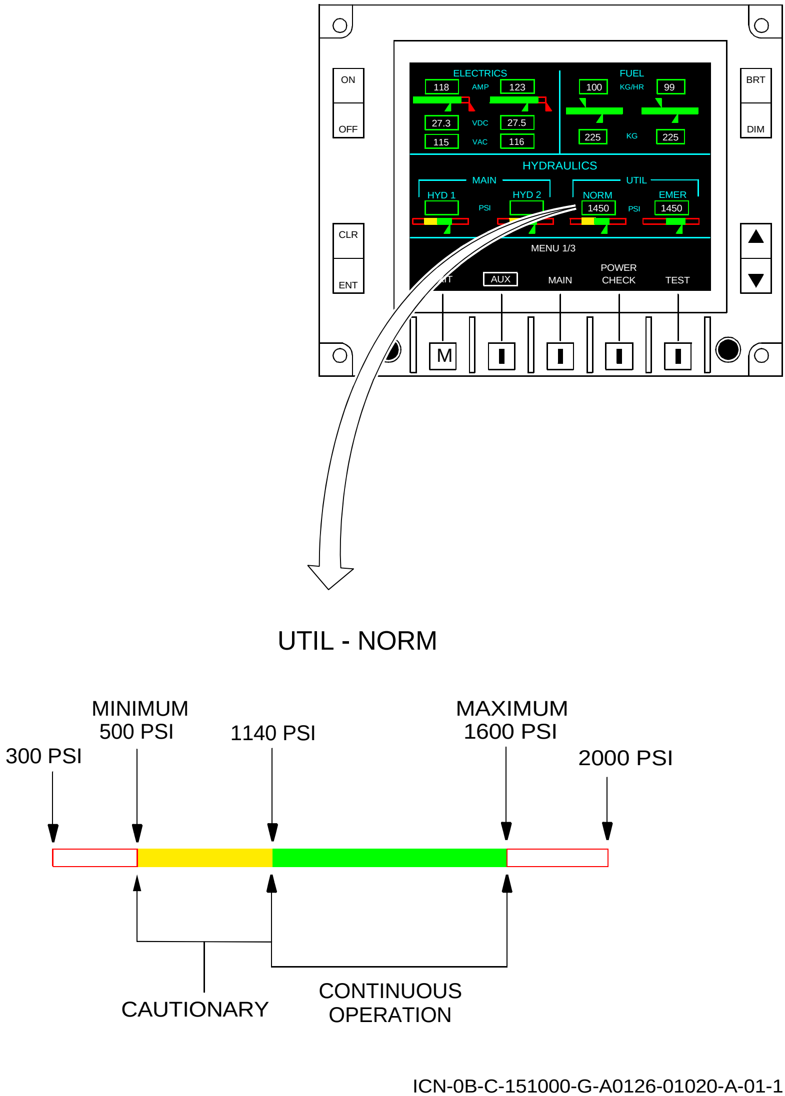

Figure 1-25 EDU2 (AUXILIARY) - Utility Hydraulic System Pressure (Normal)

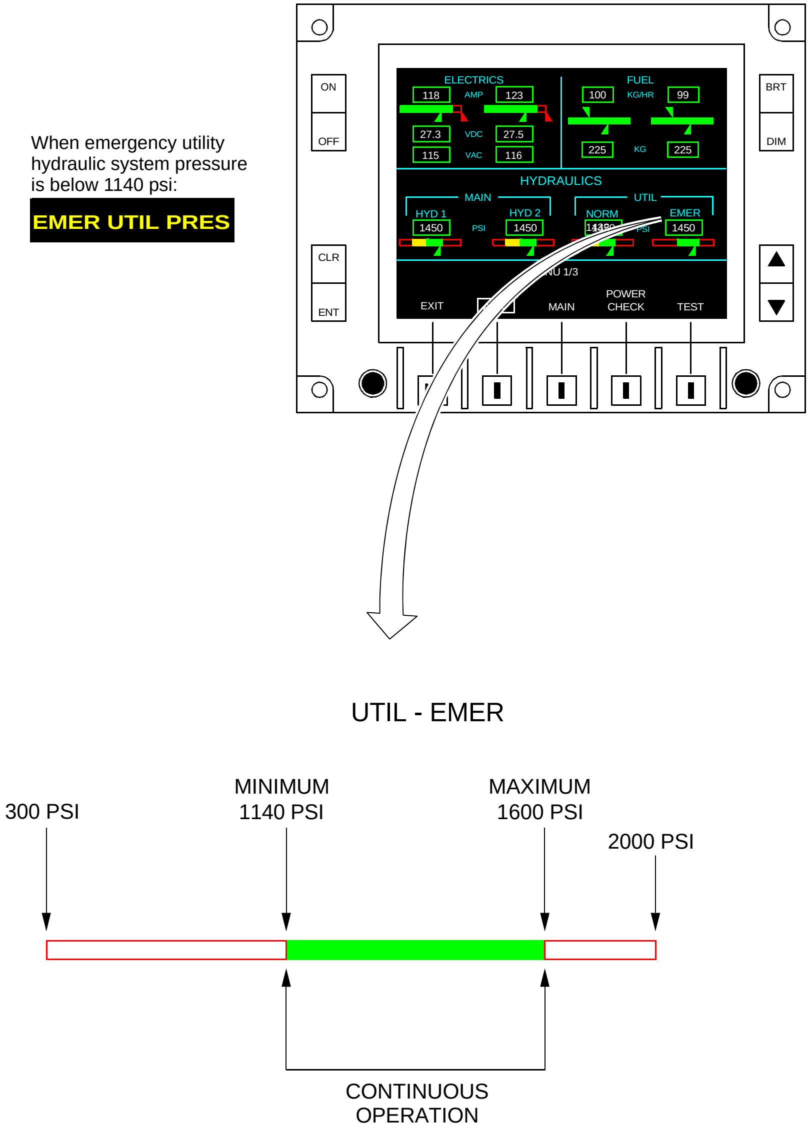

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Figure 1-26 EDU2 (AUXILIARY) - Utility Hydraulic System Pressure (Emergency)

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SECTION 2

NORMAL PROCEDURES

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SECTION 2

NORMAL PROCEDURES

GENERAL

This section contains instructions and procedures for operating the helicopter from the planning stage, through actual flight conditions, to securing the helicopter after landing.

Normal and standard conditions are assumed in these procedures. Pertinent data in other sections is referenced when applicable.

The instructions and procedures contained herein are written for the purpose of standardization and are not applicable to all situations.

The minimum and maximum limits, and the normal and cautionary operating ranges for the helicopter and its subsystems are indicated by instrument markings and placards. Refer to Section 1 for details of the operating limitations.

Each time an operating limitation is exceeded, a malfunction or an emergency occurs, an appropriate entry shall be made in the logbook (airframe, engine, etc.). The entry shall state which limit was exceeded, the duration of time, the extreme value attained, and any additional information essential in determining the maintenance action required. As an aid to this task, the IDS software includes logs of limit exceed data, fault data, and caution and warning signals which are stored in a Non-Volatile Memory for subsequent retrieval.

FLIGHT PLANNING

Each flight must be planned adequately to ensure safe operations and to provide the pilot with the data to be used during the flight.

Essential weight and balance, and performance information should be compiled as follows:

- Check type of mission to be performed and destination.
- Select appropriate performance charts to be used from Section 4.
- Review the appropriate Supplements of this Rotorcraft Flight Manual for the optional equipment installed.

Ascertain proper weight and balance of the helicopter as follows:

- Consult Section 6 - Weight and Balance;
- Ascertain weight of fuel, oil, payload, etc;
- Compute take-off and anticipated landing gross weights;
- Check helicopter center of gravity (CG) locations;
- Check that the weight and CG limitations in Section 1 are not exceeded during the flight.

PRE-FLIGHT CHECKS

GENERAL

Preflight checks are to be performed by the pilot in order to confirm that the helicopter is flightworthy and adequately equipped.

The checks are categorised as follows:

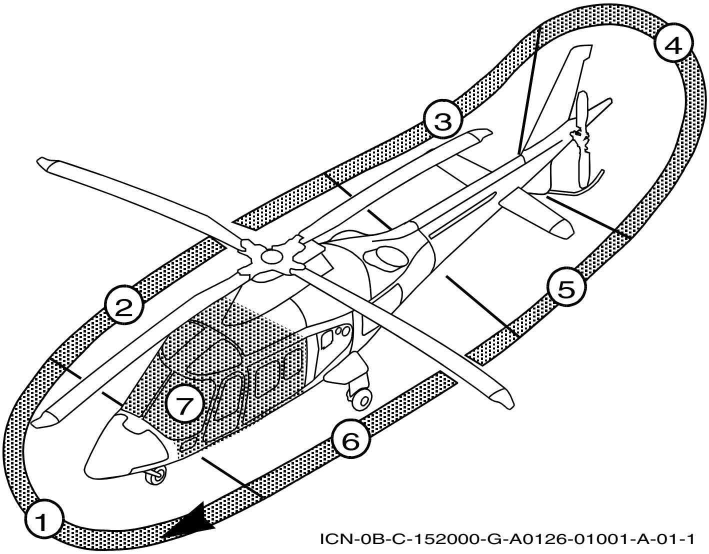
- First flight of the day
- Every flight

Checks marked with an arrow (➡) are required before the first flight of the day. All other checks are to be carried out before every flight.

Checks in blue may be omitted if the AIRWORTHINESS CHECK has been performed as per MAINTENANCE MANUAL.

The inspection commences at the nose and continues clockwise around the helicopter (Figure 2-1). During the inspection, confirm that there are no leaks from drains, that all vents, air intakes and air outlets are free of obstruction, and all access panels and antennas are secure.

PILOT'S PRE-FLIGHT CHECK



AREA N°1: Helicopter nose
AREA N°2: Fuselage - RH side
AREA N°3: Tail boom - RH side
AREA N°4: Fins, tail gearbox, tail rotor and skid
AREA N°5: Tail boom - LH side
AREA N°6: Fuselage - LH side
AREA N°7: Helicopter interior

Figure 2-1 Pre-flight Check Sequence

The following procedure outlines the pilot walk-around and interior checks (see Figure 2-1)

1. Main and tail rotor tie-downs : Removed.

Area N°1 (Helicopter Nose)

1. Nose exterior : Condition.
2. Ventilation air intake : Free of obstruction.
3. Pitot-static tubes : Cover removed, condition and free of obstruction.
4. Nose landing gear : Condition, shock strut extension, leaks, tyre condition and pressure.
5. Searchlight : Condition and cleanliness.
6. ➡ Nose compartment access door : Open.
7. ➡ Avionics components : Condition and secured.
8. ➡ Accumulators : Condition and free of leaks.
9. Nose compartment access door : Secure.
10. ➡ Accumulators : Discharge by pressing two relevant red pushbuttons.

CAUTION

The discharge of accumulators causes loss of parking brakes. Suitable measures (wheel chocks) should be taken to ensure helicopter will not move.

11. ➡ Drains and vents : Free of obstruction.

Area N°2 (Fuselage - RH side)

1.	Windshield	: Condition and cleanliness.
2.	Roof, lateral and lower transparent panels	: Condition and cleanliness.
3.	➡ Windshield wiper	: Condition.
4.	➡ RH OAT sensor	: Condition, free of obstruction.
5.	Fuselage exterior	: Condition.
6.	Pilot door window	: Condition, cleanliness, and secure.
7.	➡ Antenna(s)	: Condition.
8.	Emergency floats electrical connector (if installed)	: Cap locked. Chain condition and secure.
9.	Sliding door	: Condition and cleanliness of windows.
10.	➡ Sliding door jettison windows	: Security of windows and seal retainers, condition of emergency markings.
11.	Cowlings and fairings	: Condition and secure.
12.	➡ Fore and middle access doors	: Open.
13.	➡ Servo hydraulic system valves and filter group	: Check for leaks and status (Red button out: filter clogged).
14.	➡ Hydraulic system tanks	: Check fluid level and filler caps for security.
15.	Fore and middle access doors	: Secure.
16.	ENGINE OIL COOLER access door	: Secure.
17.	➡ Service step	: Open and use to reach upper part of helicopter.

18.	➡ Main rotor hub and blades	: Condition and secure.
19.	➡ Main rotor dampers	: Condition and secure. Check for correct charge indication.
20.	➡ Main rotor pitch change links	: Condition and secure.
21.	➡ Swashplate and driving scissors	: Condition and secure.
22.	➡ Upper anti-collision light	: Condition and cleanliness.
23.	➡ HYD. SERVOS access door	: Open.
24.	➡ Servo actuator (actuator with yellow decal)	: Condition and leaks.
25.	➡ Main transmission and accessories (visible area)	: Condition and leaks.
26.	➡ Transmission external oil filter	: By-pass indication (Red button out: filter clogged).
27.	HYD. SERVOS access door	: Secure.
28.	➡ ENGINE OIL COOLER access door	: Open.
29.	➡ Cooler blower air intake	: Free of obstruction.
30.	➡ Cooler system belt	: Condition and secured.
31.	ENGINE OIL COOLER access door	: Secured.
32.	➡ Airframe (A/F) fuel filter	: Condition and leaks.
33.	Service step	: Secure.
34.	Engine air intake screen and chamber	: Covers removed; free of damage and obstruction.
35.	➡ Engine access door	: Open.

36.	➡ Engine compartment drain filters	: Free of obstructions.
37.	➡ Engine area	: Check for fuel and oil leaks.
38.	➡ Engine oil	: Check gauge for oil level.
39.	➡ Engine oil filter impending bypass indicator	: Check for correct indication. (Red pop-up indicator not in sight).
40.	➡ Engine-transmission drive shaft	: Condition.
41.	➡ Engine supports (visible area)	: Condition.
42.	Engine access door	: Condition, secure.
43.	Fuel filler cap	: Secure.
44.	➡ Igniter access door	: Open.
45.	➡ Igniter box	: Condition.
46.	➡ Engine fire extinguisher bottle.	: Condition.
47.	Engine Exhaust Support	: Condition
48.	Igniter access door	: Secured.
49.	Engine fire extinguisher indicator disc	: Confirm in the red position.

Note

If the engine fire extinguisher indicator disc is not in the red position, it means that the relevant bottle has already been discharged and needs to be replaced.

50.	Tail rotor driveshaft support access door	Secure.

51.	Engine exhaust	: Cover removed, condition and free of fuel.
52.	Main landing gear	: Condition, shock strut extension, leaks, tyre condition and pressure.
53.	➡ Wheel brake disc	: Confirm freedom of movement
54.	➡ L/G locking system pin	: Confirm freedom of movement
55.	Landing, taxi light and transparent panel	: Condition and cleanliness.
56.	➡ Antenna(s)	: Condition.
57.	➡ Drain and vents lines	: Free of obstruction.
58.	External Power door	: Secure.
59.	Emergency floats electrical connector (if installed)	: Cap locked. Chain condition and secure.

Area N°3 (Tailboom - RH side)

1.	Tailboom exterior	: Condition.
2.	➡ Antenna(s)	: Condition.
3.	➡ Lower anti-collision light	: Condition and cleanliness.
4.	Stabilizer	: Condition and secure.
5.	Position lights and flood lights (if installed)	: Condition and cleanliness.

Area N°4 (Fins, tail gearbox and skid)

1.	Tail fin and skid	: Condition.
2.	Tail navigation light	: Condition and cleanliness.
3.	➡ Tail rotor driveshaft door	: Open
4.	➡ Tail rotor driveshaft bearing	: Condition and secured. Check for grease leaks

5.	➡ Tail rotor driveshaft door	: Secure
6.	➡ Tail rotor gearbox access door	: Open.
7.	➡ Tail rotor gearbox	: Confirm no leaks.
8.	➡ Tail rotor pitch link control lever	: Condition.
9.	Tail rotor gearbox access door	: Secure.
10.	Oil filler cap	: Secure.

Area N°5 (Tailboom and tail rotor - LH side)

1.	Tail rotor gearbox oil level	: Check oil level.
2.	Tail rotor hub and blades	: Condition, cleanliness and freedom of flapping.
3.	Tail rotor pitch change mechanism	: Condition and secure.
4.	Tailboom exterior	: Condition.
5.	Stabilizer	: Condition and secure.
6.	Position lights and flood lights (if installed)	: Condition and cleanliness.
7.	➡ Antenna(s)	: Condition.
8.	➡ Tail rotor driveshaft cover	: Open.
9.	➡ Tail rotor driveshaft bearings	: Condition and secured. Check for grease leaks. Check no marks of slippage.
10.	Tail rotor driveshaft cover	: Secure.
11.	➡ Tail rotor shaft inspection pins	Confirm freedom of movement

Area N°6 (Fuselage - LH side)

1.	Baggage compartment door	: Open.
2.	Baggage compartment	: Cargo (if on board) properly secured.
3.	➡ Tail rotor hydraulic servo actuator	: Check for oil leaks.

Note

The tail rotor hydraulic servo actuator is accessible in the baggage compartment through an inspection door.

4.	➡ Circuit breakers (in baggage compartment)	: All in.

Note

The circuit breakers in the baggage compartment are accessible through an inspection door.

5.	Baggage compartment door	: Secure.
6.	➡ Drains and vents lines	: Free of obstruction.
7.	Emergency floats electrical connector (if installed)	: Cap locked. Chain condition and secure.
8.	Main landing gear	: Condition, shock strut extension, leaks, tyre condition and pressure.
9.	➡ Wheel brake disc	: Confirm freedom of movement
10.	➡ L/G locking system pin	: Confirm freedom of movement
11.	Landing, taxi light and transparent panel	: Condition and cleanliness.
12.	Engine exhaust	: Cover removed, condition and free of fuel.

13.	Engine fire extinguisher indicator disc	: Confirm in the red position.

Note

If the engine fire extinguisher indicator disc is not in the red position, it means that the relevant bottle has already been discharged and needs to be replaced.

14.	➡ Tail rotor driveshaft support access door	: Open.
15.	➡ Igniter access door	: Open.
16.	➡ Igniter box	: Condition.
17.	➡ Engine fire extinguisher bottle	: Condition.
18.	➡ Tail rotor middle drive shaft bearings	: Check condition. Check no marks of slippage.
19.	Engine Exhaust Support	: Condition
20.	Igniter access door	: Secured.
21.	Tail rotor driveshaft support access door	: Secure.
22.	Engine air intake screen and chamber	: Cover removed; free of damage and obstruction.
23.	➡ Engine access door	: Open.
24.	➡ Engine compartment drain filters	: Free of obstructions.
25.	➡ Engine area	: Check for fuel and oil leaks.
26.	➡ Engine oil	: Check gauge for oil level.
27.	➡ Engine oil filter impending bypass indicator	: Check for correct indication.

28. ➡ Engine-transmission drive shaft : Condition.

29. ➡ Engine support (visible area) : Condition.

30. Engine access door : Condition, secure.

31. ➡ ENGINE OIL COOLER access door : Open.

32. ➡ Cooler blower air intake : Free of obstruction.

33. ➡ Cooler system belt : Condition and secured.

34. ENGINE OIL COOLER access door : Secure.

35. Service step : Open and use to reach upper part of helicopter.

36. ➡ Main rotor hub and blades : Condition and secure.

37. ➡ Main rotor dampers : Condition and secure.

    Check for correct charge indication.

38. ➡ Main rotor pitch change links : Condition and secure.

39. ➡ Swashplate and driving scissors : Condition and secure.

40. ➡ Transmission oil access door : Open.

41. ➡ Transmission oil filler cap : Secure.

CAUTION

The transmission oil level check is to be performed in any case before the first flight of the day and it can be considered valid for a maximum of 5 flights including the first.

Note

Transmission oil level must be between the MIN and the MAX markings.

42.	➡ Transmission oil	: Confirm correct level.
43.	➡ Transmission oil access door	: Secure.
44.	➡ HYD. SERVOS access door	: Open.
45.	➡ Main transmission and accessories (visible area)	: Condition and leaks.
46.	➡ Servo actuators (visible ones: actuator with red decal and one with blue decal)	: Condition and leaks.
47.	HYD. SERVOS access door	: Secure.
48.	➡ Airframe (A/F) fuel filter	: Condition and leaks.
49.	Service step	: Secure.
50.	Cowling and fairings	: Condition and secure.
51.	➡ Sliding door jettison windows	: Security of windows and seal retainer, condition of emergency markings.
52.	Sliding door	: Condition and cleanliness of windows.
53.	Fuselage exterior	: Condition

54.	Fore and middle access door	: Open
55.	Flight control rods	: Confirm freedom of movement
56.	Fore and middle access door	: Secure
57.	Co-pilot door and window	: Condition, cleanliness, seal retainer and secure.
58.	➡ LH OAT sensor	: Condition, free of obstruction.
59.	Windshield	: Condition and cleanliness.
60.	Roof, lateral and lower transparent panels	: Condition and cleanliness.
61.	➡ Windshield wiper	: Condition.
62.	➡ Antenna(s)	: Condition.
63.	Emergency floats electrical connector (if installed)	: Cap locked. Chain condition and secure.

Area N°7 (Helicopter interior)

Cabin interior

1.	➡ Sliding door jettison windows (RH and LH)	: Security and condition of seal retainer and red strap.
2.	➡ Sliding doors (RH and LH)	: Confirm correct operation of locking and mechanical stop devices.
3.	Passenger safety belts	: Condition and belts fastened

Note

Operation with passenger sliding doors open or removed requires removal or correct securing of all cabin equipment, installations and trim panels and that passenger safety belts are fastened if seats are unoccupied.

4.	Cabin interior	: Check security of equipment. Confirm presence of markings.
5.	➡ First aid kit	: Check on board and content.
6.	Sliding doors (RH and LH)	: Closed and secure.

Cockpit interior

7.	➡ Co-pilot door jettison handle	: Correct position and secure.
8.	➡ Co-pilot safety belt and inertia reel	: Condition and belt fastened.
9.	➡ Co-pilot seat	: Secure.
10.	➡ Co-pilot flight controls	: Condition and secure.
11.	➡ LH lower and lateral transparent panels	: Condition and cleanliness.
12.	Co-pilot door	: Closed and secure. Sliding windows closed.
13.	➡ Pilot door jettison handle	: Correct position and secure.
14.	➡ Pilot safety belt and inertia reel	: Condition.
15.	➡ Pilot seat	: Secure.
16.	➡ Pilot flight controls	: Condition and secure.
17.	➡ RH lower and lateral transparent panels	: Condition and cleanliness.
18.	➡ Cockpit fire extinguisher	: Charged and secure.
19.	➡ Passive vibration absorber (if installed)	: Check cover secured.
20.	➡ Instruments, panels and circuit breakers	: Condition and legibility.

21. Circuit breakers : All in.
(Main, pilot and co-pilot overhead).

22. ➡ MSTR AVNX switch : As required.

23. ➡ All other switches/controls : Confirm OFF / guarded / normal position when shutdown.

For the following checks connect the d.c. supply.

Note

The following checks may require a large electrical consumption. Beware of possible battery charge depletion if not using external power.

24. ➡ BAT switch : ON.

25. ➡ GEN BUS 1 and 2 switches : ON.

26. ➡ External Power : Connect (if required).
If external power connected, the battery is automatically disconnected. Confirm BATT OFF caution message is displayed.
If battery requires charging select BAT switch to EPU.

Note

Confirm that external power source supplies not less than 28 V.

27. ➡ Check following systems for correct operation:

— Anticollision lights.

— Position lights.

— Taxi lights.

— Landing lights.

28. ➡ LH airframe (A/F) fuel filter : Gently drain while respective fuel pump is operating. Push red button on filter and check for bypass indication and #1 A/F F FLTR caution message on EDU1.

Note

Fuel is pressurised, therefore drainage should be carried out by gently pushing red button. Failure to comply with this advice could result in some fuel being squirted around.

29. ➡ LH fuel pump : Drain by raising guard and setting FUEL DRAIN switch located in baggage compartment to TNK 1 (upper position).
Check for fuel dripping from the drain and verify FUEL DRAIN 1 caution message displayed on EDU 1.
Set switch to OFF (centre position).
Verify no fuel dripping and FUEL DRAIN 1 caution message suppressed.

30. ➡ RH fuel pump : Drain by setting FUEL DRAIN switch to TNK 2 (lower position). Check for fuel dripping from the drain and verify FUEL DRAIN 2 caution message displayed on EDU 1.
Set switch to OFF (centre position).
Verify no fuel dripping and FUEL DRAIN 2 caution message suppressed.
Lower guard.

31. ➡ RH airframe (A/F) fuel filter : Gently drain while respective fuel pump is operating. Push red button on filter and check for bypass indication and #2 A/F F FLTR caution message on EDU1.

Note

Fuel is pressurised, therefore drainage should be carried out by gently pushing red button. Failure to comply with this advice could result in some fuel being squirted around.

32. ➡ External Power (if used) : Disconnect.

33. ➡ BAT switch : OFF.

34. ➡ GEN BUS 1 and 2 switches : Check automatically to OFF.

35. Pilot door : Closed and secure.

ENGINE PRE-START CHECK

Note

Instruction in blue are applicable only if kit P/N 109-B811-02 (fixed landing gear) is not installed

INITIAL CHECKS

1.	Pedals and seats	: Adjust.
2.	Seat belts	: Fasten and adjust.
3.	Doors (pilot and co-pilot)	: Closed and locked.
4.	Jettison handles	: Correct position and secure.
5.	Landing gear lever	: Confirm DOWN.
6.	Landing gear EMER/ NORMAL switch	: Confirm in NORMAL position and wirelocked.
7.	Parking brake	: ON (pull out and turn).
8.	Nose wheel lock	: ON.
9.	Cyclic stick	: Centered, or positioned to counter wind, and friction adjusted.
10.	Collective lever	: Fully down and friction adjusted.
11.	All switches/controls	: Confirm OFF / guarded / normal position when shutdown.

ELECTRICAL POWER-UP

1.	MSTR AVNX switch	: Set to GND
2.	BAT switch	: ON.
3.	GEN BUS 1 and 2 switches	: ON.
4.	GEN 1 and 2 switches	: ON.
5.	INV 1 and 2 switches	: ON.
6.	Cockpit lights	: As required.
7.	External power	: Connect (if required). If external power is connected, the battery is automatically disconnected. Confirm BATT OFF caution message is displayed. If battery requires charging select BAT switch to EPU.

Note

Confirm that external power source supplies not less than 28 V.

8.	MWL and MCL	: Acknowledge and reset by: - Pressing on MWL and MCL or - Master Reset pushbutton on collective lever.
9.	Pilot PFD and MFD displays	: Confirm on. Verify System Status page: - SOFTWARE OK. - database validity and expiring dates. - Check EFIS software version for applicable limitations. Press any keys on pilot PFD and MFD to confirm. Adjust brightness as required.

Note

Refer to EFIS Pilot's Guide for database updating procedure.

10. EDUs and ESIS displays : Confirm on.
Adjust brightness as required.

<u>EDU Tests:</u>

Note

The TEST function of both EDUs can be initiated by pressing the TEST key on either EDU.

Note

During the test, the DAU will activate the MWL, MCL, the ENGINE FIRE voice warning message and audio tone; it will also illuminate the engine power lever grips and the FIRE warning lights on the engine control panel. If a failure is detected on engine fire and/or fuel low detectors, the caution message FIRE DET and/or F LOW FAIL will appear.

Note

Due to the short length of the EDU Test, only one of the ENGINE FIRE voice messages is heard. The ENGINE FIRE message of the second engine can be heard by pushing the MASTER WARNING RESET pushbutton.

11. EDU 1 or EDU 2 : Select MENU, press TEST key and check the following test sequences on both EDUs.

11a. EDU 1 : The test sequence shall display the CRUISE page with the following data:

N1	97.4%
TOT	820 °C
TRQ	100%
N2 / NR	101%

11b. EDU 2 : The test sequence shall display the MAIN page with the following data:

ENG OIL pressures	50 PSI
ENG OIL temperatures	100 °C
XMSN OIL pressure	40 PSI
XMSN OIL temperature	100 °C
FUEL pressure	20 PSI
HYD oil pressure	1500 PSI
FUEL quantity	Decreasing
OAT	+25° C

In the advisory area (lower part of the screen) the test sequence shall display:

EDU 1 and EDU 2 software identification number.

DAU-A and DAU-B software identification number.

EDU 1 BIT PASS

EDU 2 BIT PASS

DAU-A BIT PASS

DAU-B BIT PASS

11c. EDU 1 and EDU 2	: After 10 seconds the EDUs will automatically return to previous selected pages.
11d. EDU 1	: Select MENU and enter page 2. Confirm CH-A and CH-B legends are green and white respectively.

AWG Test:

12. Aural Warning Generator Test	: Select and hold AWG switch to TEST. Confirm the aural message "TEST OK" and after about 6 seconds the complete list of aural messages in the following sequence:

Tone 1	"ROTOR LOW"
Tone 2	"ENGINE ONE OUT"
Tone 2	"ENGINE TWO OUT"
Tone 3	"FIRE ENGINE ONE FIRE"
Tone 3	"FIRE ENGINE TWO FIRE"
Tone 4	"WARNING"
Tone 4	"ROTOR HIGH"
Tone 4	"AUTOPILOT"
Tone 4	"ENGINE IDLE"
Tone 5	"OVERTORQUE"
Tone 5	"AIRSPEED"
Tone 6	"LANDING GEAR"
Tone 6	"DECISION HEIGHT"
Tone 7	"TWO HUNDRED FEET"
Tone 7	"ALTITUDE"
Tone 8	Tone (for autopilot)

Engine Trim Function:

13.	ENG GOV 1 and 2 switches	: AUTO.

CAUTION

The engine power levers may be operated manually only in case of failure of the remote control (PLA MOTOR caution message active), or before starting, to position the levers to FLIGHT.

14.	ENG TRIM 1 and 2 toggles (on pilot's collective)	: Confirm correct operation, then leave the engine power levers in the FLIGHT position.
15.	EDU 1	: Confirm #1 PLA and #2 PLA caution messages suppressed.

Miscellaneous:

16.	Fire Extinguisher Bottle switch	: Centered.
17.	Fire Extinguisher Pushbutton covers	: Condition.
18.	STATIC source switch	: NORM and guarded.
19.	ELT switch	: Confirm in ARM position.
20.	Pilot's ICS	: Set to BK-UP then to EMER and check functionality. Reset to NORM.
21.	SERVO (Main hydraulics)	: NORM.
22.	Cabin ventilation knob	: As required.
23.	Landing gear indications	: Confirm 3 green lights illuminated (Nose, RH, LH) and red light extinguished.

24. LAMP switch	: Select and hold LAMP switch to TEST
24a. Landing gear indication	: Confirm red light illuminated.
24b. APMS panel	: Confirm all pushbutton lights illuminated.
25. LAMP switch	: Release.
25a. Landing gear indication	: Confirm red light extinguished.
25b. APMS panel	: Confirm all pushbutton lights extinguished.
26. OEI TNG switch	: Confirm in off position (centered).
27. Altimeter	: Set.
28. Fuel quantity	: Check.
29. LD-SH (Load share) switch	: As required.
30. NR switch	: Set to 100%.
31. Rotor brake	: Engage and confirm ROTOR BRK caution message displayed. Disengage and confirm caution message suppressed.

ENGINE START

1. EDU 1 : Confirm START page selected.

If a quick engine start is to be performed refer to **QUICK ENGINE START** procedure.

NORMAL ENGINE START

Either engine may be started first. In the following procedure Engine 1 is started first.

Engine 1 start

1. GEN 1 and 2 switches : Check ON.
2. FUEL VALVE 1 switch : OPEN (bar vertical).
3. FUEL PUMP 1 switch : ON.
   Confirm FUEL PUMP 1 caution message suppressed.
   Check fuel pressure.

CAUTION

Monitor engine start and if any of the following occur:

- light-up is not obtained within 15 seconds;
- abnormal noises are heard;
- TOT increases beyond start limits (#1(2) HOT START caution message displayed);
- rotor has not begun to rotate when N1 is 40%;
- N1 or N2 increase beyond limits;
- engine hangs (stagnation in N1 below 54%);

Abort the engine start and shutdown the engine by performing the **ABORTED ENGINE START PROCEDURE**.

4. ENG 1 MODE switch : Set to IDLE.
5. N1 : Note increasing and START legend vertically displayed.
6. TOT : Note increasing and IGN legend vertically displayed.
7. Rotors : Begin turning before N1 reaches 40%.

CAUTION

The operation at an engine oil pressure up to 200 psi is permitted for a maximum period of 10 minutes.

8. Engine oil pressure When N1 is greater than 40 % : Check rising.

CAUTION

Do not apply power or allow N1 to rise above 90% until engine oil temperature reaches 10 °C.

9. Engine oil temperature : Check.

CAUTION

Remain at IDLE until the transmission oil presure returns below 70 psi whilst the oil temperature increases.

10. Transmission oil pressure : Check rising.

11. Engine starter : At N1 = 50%, engine starter is automatically deactivated.

    #1 DC GEN caution message suppressed.

    START and IGN legends are also automatically suppressed.

12. Main hydraulic system : Confirm rise in main hydraulic pressure, when the main rotor begins to rotate.

13. Hydraulic utility system (normal and emergency) : When accumulators are initially discharged, note the activation of MAIN UTIL CHRG and EMER UTIL CHRG caution messages as main rotor begins to rotate. Note both caution messages are suppressed when systems are charged.

CAUTION

Below 85% NR, avoid any cyclic movement except to prevent hitting blade stops.

14.	N2	: Confirm stabilized to IDLE speed of 65% ± 1%.
15.	Engine and transmission oil	: Confirm pressure and temperature within limits.
16.	ENG 1 MODE switch	: Set to FLT.
17.	N2 / NR	: Confirm stabilized at 100%.

Engine 2 start

CAUTION

Ensure that the second engine engages as the N2 increases to FLT. A non-engaged engine shows positive N2 value and near zero torque. If a non-engagement occurs, shutdown the non-engaged engine first. When the non-engaged engine has stopped, shutdown the engaged engine. If a sudden or hard engagement occurs, shutdown both engines for maintenance action.

Repeat '**Engine 1 Start**' procedure for engine N°2

18.	Engine parameters	: Confirm within limits.
19.	Transmission parameters	: Confirm within limits.
20.	BAT switch	: Confirm ON. (BATT OFF caution message displayed if external power is connected).

21.	External power (if used)	: Disconnect and close door. Confirm EXT PWR ON and BATT OFF caution messages suppressed.
22.	N2 / NR	: Confirm stabilized at 100%.
23.	MSTR AVNX	: Set to NORM.
24.	Co-pilot PFD and MFD displays	: Confirm on. Verify System Status page: - SOFTWARE OK. - database validity and expiring dates. - Check EFIS software version for applicable limitations. Press any keys on co-pilot PFD and MFD to confirm.
		: Adjust brightness as required.

Note

Refer to EFIS Pilot's Guide for database updating procedure.

QUICK ENGINE START

Note

It is recommended to start the engine to IDLE using the NORMAL ENGINE START procedure. Nevertheless, when OAT is warmer than 0 °C or during a warm engine start, it is possible to start to FLIGHT using the QUICK ENGINE START procedure.

This procedure can be followed whenever the situation requires to speed up the take-off.

It is possible to start a single engine or both engines in succession.

1.	GEN 1 and 2 switches	: Check ON.
2.	FUEL VALVE 1 switch	: OPEN (bar vertical).
3.	FUEL PUMP 1 switch	: ON. Confirm FUEL PUMP 1 caution message suppressed. Check fuel pressure.
4.	FUEL VALVE 2 switch	: OPEN (bar vertical).
5.	FUEL PUMP 2 switch	: ON. Confirm FUEL PUMP 2 caution message suppressed. Check pressure.

CAUTION

Monitor engine start and if any of the following occur:

- light-up is not obtained within 15 seconds;
- abnormal noises are heard;
- TOT increases beyond start limits (#1(2) HOT START caution message displayed);
- rotor has not begun to rotate when N1 is 40%;
- N1 or N2 increase beyond start limits;
- engine hangs (stagnation in N1 below 54%);

Abort the engine start and shutdown the engine by performing the **ABORTED ENGINE START PROCEDURE**.

6. ENG 1 MODE switch : Set to FLT.
7. ENG 2 MODE switch : Set to FLT, when GEN 1 load is 200 A or less
8. N1 : Note increasing and START legend vertically displayed.
9. TOT : Note increasing and IGN legend vertically displayed.
10. Rotors : Begin turning before N1 reaches 40%.

CAUTION

Operating with an engine oil pressure of up to 200 psi is permitted for a maximum period of 10 minutes.

11. Engine oil pressures — When N1 is greater than 40 %

    : Check rising.

CAUTION

Do not apply power or allow N1 to rise above 90% until engine oil temperature reaches 10 °C.

12. Engine oil temperatures : Check.

CAUTION

Remain at IDLE until the transmission oil pressure decreases below 70 psi whilst the oil temperature increases.

13. Transmission oil pressure : Check rising.

14. Engine starters : At N1 = 50%, engine starters are automatically deactivated.

    #1 DC GEN and #2 DC GEN caution messages suppressed.

    START and IGN legends are also automatically suppressed.

15. Main hydraulic system : Confirm rise in main hydraulic pressure when the main rotor begins to rotate.

16. Hydraulic utility system (normal and emergency) : When accumulators are initially discharged, note the activation of MAIN UTIL CHRG and EMER UTIL CHRG caution messages as main rotor begins to rotate. Note both caution messages are suppressed when systems are charged.

CAUTION

Below 85% NR, avoid any cyclic movement except to prevent hitting blade stops.

17. Engine and transmission oil : Confirm pressure and temperature within limits.

18. Engine parameters : Confirm within limits.

19. BAT switch : Confirm ON. (BATT OFF caution message displayed if external power is connected).

20. External power (if used) : Disconnect and close door. Confirm EXT PWR ON and BATT OFF caution messages suppressed.

21. N2 / NR : Confirm stabilized at 100%.

22. MSTR AVNX : Set to NORM.

ABORTED ENGINE START PROCEDURE

CAUTION

Failure to follow the abort procedures may cause damage to the engine.

The affected engine should be shutdown by:

1. ENG MODE switch : OFF.

   If engine does not begin to shutdown:

   Engine power lever : OFF.

2. FUEL PUMP switch : OFF.
3. FUEL VALVE switch : CLOSED (bar horizontal).

Perform the following procedure before attempting to restart the engine:

- Allow 30 seconds fuel drain period.
- Perform a 30 seconds **DRY MOTORING RUN**.

Refer to **DRY MOTORING RUN** procedure.

Note

Engine start after Dry Motoring Run performance, refer to Section 1 for engine starter limitations.

DRY MOTORING RUN

The following procedure is used to clear the engine of internally trapped fuel and vapor from within the engine.

1.	ENG GOV switch	: AUTO.
2.	ENG MODE switch	: OFF.
3.	Engine power lever	: OFF.
4.	FUEL PUMP switch	: OFF.
5.	FUEL VALVE switch	: CLOSED (bar horizontal).
6.	IGN circuit breaker (on overhead circuit breaker panel - see Figure 2-2)	: Pull out.
7.	Starter pushbutton (on engine power lever)	: Push and hold as necessary.
8.	N1	: Note increasing.
9.	TOT	: Note decreasing.

After required Dry Motoring Run period:

10.	Starter pushbutton	: Release.
11.	IGN circuit breaker (on overhead circuit breaker panel - see Figure 2-2)	: Reset.

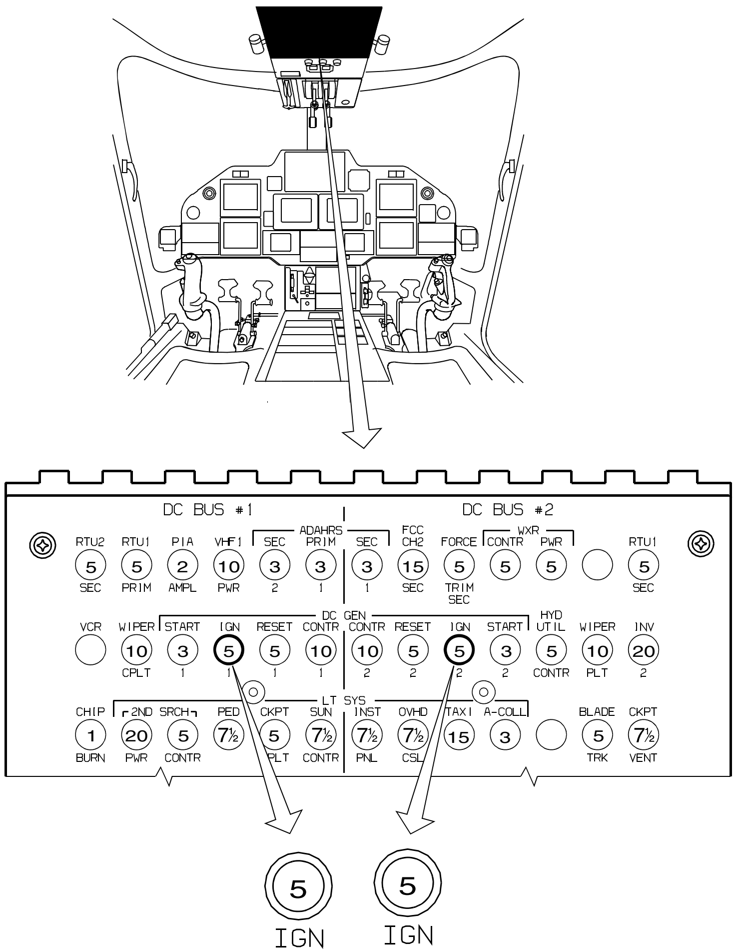


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Figure 2-2 IGN Circuit Breakers Location

SYSTEM CHECKS

HYDRAULIC SYSTEMS

1. SERVO switch (Main Hydraulic)	: NORM, confirm. Make small cyclic (clockwise), collective and pedal movements. Pressure drops must be equal for both systems (N°1 and N°2) and should not exceed 70 psi. Set SERVO switch to SOV 2 OFF. SERVO 2 caution message displayed. Check operation of system N°1 with same cyclic, collective and pedal movements. Pressure drop should not exceed 70 psi and there should be no force increase, discontinuity or cyclic/collective coupling. Repeat check setting the switch to SOV 1 OFF to check system N°2. Confirm that pedals are unboosted. Then set SERVO switch to NORM.

FUEL SYSTEM

1. XFEED VALVE switch : AUTO (bar vertical).

Note

When FUEL PUMP 1 (2) is OFF and crossfeed valve is open, N°2 (1) FUEL quantity box shall appear in magenta.

2. FUEL PUMP 1 switch : OFF.

   FUEL PUMP 1 caution message displayed.

   Crossfeed valve automatically open (bar horizontal) and XFEED advisory message displayed.

   Note N°1 fuel pressure gauge indicating normal operating pressure.

3. FUEL PUMP 1 switch : ON.
   FUEL PUMP 1 caution message suppressed.
   XFEED advisory message suppressed (bar vertical).

4. FUEL PUMP 2 switch : OFF.

   FUEL PUMP 2 caution message displayed.

   Crossfeed valve automatically open (bar horizontal) and XFEED advisory message displayed.

   Note N°2 fuel pressure gauge indicating normal operating pressure.

5.	FUEL PUMP 1 switch	: OFF. Note fall in fuel pressure, activation of FUEL PUMP 1 caution message, crossfeed valve still open (bar horizontal) and XFEED advisory message still present. Confirm correct operation of engine-driven fuel pumps by checking that engine parameters are stable.
6.	XFEED VALVE switch	: CLOSED (bar vertical). XFEED advisory message suppressed.
7.	XFEED VALVE switch	: AUTO (bar horizontal). XFEED advisory message activated.
8.	FUEL PUMP 1 and 2 switches	: ON. FUEL PUMP 1 and 2 caution messages suppressed and crossfeed valve automatically closed (bar vertical) and XFEED advisory message suppressed.
9.	XFEED VALVE switch	: OPEN (bar horizontal). XFEED advisory message activated. AUTO (bar vertical). XFEED advisory message suppressed.

ELECTRICAL A.C. SYSTEM

1.	INV 1 switch	: OFF. Confirm INV 1 caution message displayed. Check for proper reading (115 V) on both AC systems on EDU 2 AUX page.
2.	INV 1 switch	: ON. Confirm INV 1 caution message suppressed.
3.	INV 2 switch	: OFF. Confirm INV 2 caution message displayed. Check for proper reading (115 V) on both AC systems on EDU 2 AUX page.
4.	INV 2 switch	: ON. Confirm INV 2 caution message suppressed.

RCP

1.	AHRS knob	: Turn AHRS knob to position 1. Confirm SAME AHRS yellow message appears on pilot and co-pilot MFD’s and 2 AP AHRS FAIL CAS message displayed. : Turn AHRS knob to position 2. Confirm SAME AHRS yellow message appears on pilot and co-pilot MFD’s and 1 AP AHRS FAIL CAS message displayed.

	: Turn AHRS knob to NORM position. Confirm SAME AHRS yellow messages extinguish and no CAS messages displayed.
2. ADU knob	: Turn ADU knob to position 1. Confirm SAME ADC yellow message appears on pilot and co-pilot MFD's.
	: Turn ADU knob to position 2. Confirm SAME ADC yellow message appears on pilot and co-pilot MFD's.
	: Turn ADU knob to NORM position. Confirm SAME ADC yellow messages extinguish.
3. GPS Knob	: Turn GPS knob to position 1. Confirm SAME GPS and SAME NAV yellow messages appears on pilot and co-pilot MFD's.
	: Turn GPS knob to position 2. Confirm SAME GPS and SAME NAV yellow messages appears on pilot and co-pilot MFD's.
	: Turn GPS knob to NORM position. Confirm SAME GPS and SAME NAV yellow messages extinguish.

AFCS

1. APMS panel	: Confirm all pushbutton lights are extinguished except AP1 and AP2 (OFF caption illuminated) and the PFD indicates the Pilot flying.
2. PFD	: Check cyclic centered.

Note

During AFCS PFT do not touch flight controls.

3.	APMS panel	: Press TEST pushbutton and follow instructions displayed on IDS PFT page. Confirm that 1(2) AP PFT COMPLETED message is displayed on IDS and confirm that neither 1(2) AP TEST FAIL nor 1(2) AP DEGRADED CAS messages displayed.
4.	PFD display	: Check consistency between PFD selection arrow on APMS panel and advisory arrow on PFD's.
5.	P/R and C/Y TRIM pushbuttons	: Check OFF lights extinguished.
6.	AP1 and AP2 pushbuttons	: Press to engage (OFF lights extinguish).

MISCELLANEOUS

1.	PITOT 1 heat switch	: ON. Check Current peak on generator load meter. Confirm PITOT 1 HEAT advisory message displayed. Confim PITOT 1 FAIL Caution is not displayed.
		: Reset to OFF. Confirm PITOT 1 HEAT advisory message suppressed.

2.	PITOT 2 heat switch	: ON. Check Current peak on generator load meter. Confirm PITOT 2 HEAT advisory message displayed. Confim PITOT 2 FAIL Caution is not displayed.
		: Reset to OFF. Confirm PITOT 2 HEAT advisory message suppressed.

3. Set communication and navigation frequencies on RTU and ICS control panels as required and check audio panel.

4.	ADIs	: On PFD's check correct alignement and no failure indications. Cross-check pilot, co-pilot and ESIS ADI information.
5.	MAG/DG switch (on AHRS panel)	: Confirm in MAG position. Cross-check pilot, co-pilot and ESIS compass headings.
6.	Altimeters	: Verify automatic baro setting on local QNH. Cross-check pilot, co-pilot and ESIS altimeter information.
7.	VSIs	: Confirm pilot, co-pilot and ESIS VSIs indicating zero (± 50 ft/min).
8.	Radio altimeter	: Confirm between 0 ft and 5 ft.
9.	NR switch	: Set to AUTO and confirm N2 / NR stabilise at 102%. Set to 102% position and confirm N2 / NR remain at 102%. Return to AUTO.
10.	HTAWS	As required
11.	TAS	As required
12.	EFIS SINCH	As required

BEFORE TAKE-OFF CHECKS

1.	Cockpit lights	: As required.
2.	Position and anti-collision lights	: Check and leave as required.
3.	Landing lights	: As required.

CAUTION

Turn both pitot heat ON for flight in visible moisture and/or rain regardless of ambient temperature.

4.	Pitot heat	: As required.
5.	NR switch	: Confirm in AUTO and N2 / NR stabilised at 102%.

CAUTION

When engine torque limiter is enabled, the AEO engine total torque is limited to a combined torque value of 220%. OEI engine torque limit remains at 162%.

6.	TQ LIM pushbutton	: If required, push to enable TORQUE limiter function, and confirm LIMITER ON advisory message appears.
7.	Parking brake	: OFF. Confirm PARK BRK ON caution message suppressed.
8.	Warning and Caution messages	: Confirm none.
9.	Engine Power Assurance Check	: As required. Refer to Section 4.

GROUND TAXIING

1.	Nose wheel lock	: OFF.
2.	Collective and cyclic	: Increase collective slowly and move the cyclic stick gently forward to start movement.
3.	Pedal brakes	: Check operation.
4.	Pedals	: As required, to select direction.
5.	Collective and pedal brakes	: To reduce speed and stop, lower the collective and apply pedal brakes.

TAKE-OFF

Note

Instruction in blue are applicable only if kit P/N 109-B811-02 (fixed landing gear) is not installed.

HOVER TAKE-OFF

1.	Nose wheel	: Align forward.
2.	Nose wheel lock	: ON.
3.	Flight controls	: Apply as necessary to lift the helicopter to a 3 ft AGL hover.
4.	Flight instruments	: Check.
5.	Engine parameters	: Confirm within limits.
6.	Transmission parameters	: Confirm within limits.
7.	Hydraulic systems parameters	: Confirm within limits.
8.	N2 / NR	: Confirm N2 / NR 102%.

9. Cyclic / Collective : Rotate nose down approximately 10 deg nose down from hover datum. While accelerating increase slightly the torque to avoid loss of altitude.
At 30 KIAS increase torque by 15% and adjust cyclic to obtain 0 deg attitude.

Continue acceleration to Vy. At Vy increase torque as required for the desired flight path.

CAUTION

Do not fly with landing gear operating or extended at speeds above 140 KIAS.

10. Landing gear lever : UP (after 200 ft RAD ALT)

MAIN UTIL CHRG caution message is displayed during landing gear retraction, then clears when landing gear is locked.

Confirm landing gear is up and locked (3 green lights and red light extinguished).

Note

If flight operations require the landing gear to be down at all times then NR must be selected manually, using the NR switch, when appropriate.

11. N2 / NR : Confirm N2 / NR stabilizes at 100%.

ROLLING TAKE-OFF

1. Nose wheel : Align forward.
2. Nose wheel lock : ON.
3. Collective and cyclic : Apply as necessary to obtain forward speed on the ground. Apply collective as necessary to become airborne. Accelerate to Vy and rotate to desired climb attitude.

CAUTION

Do not fly with landing gear operating or extended at speeds above 140 KIAS.

4. Landing gear lever : UP (after 200 ft RAD ALT)

   MAIN UTIL CHRG caution message is displayed during landing gear retraction, then clears when landing gear is locked.

   Confirm landing gear is up and locked (3 green lights and red light extinguished).

Note

If flight operations require the landing gear to be down at all times then NR must be selected manually, using the NR switch, when appropriate.

5. N2 / NR : Confirm N2 / NR stabilizes at 100%.

IN FLIGHT

1.	Load Share (LD-SH) switch	: As required.
2.	TQ LIM pushbutton	: As required.
3.	Pedals	: Apply as necessary to maintain direction.
4.	Flight instruments	: Check.
5.	Engine parameters	: Confirm within limits.
6.	Transmission parameters	: Confirm within limits.
7.	Hydraulic systems parameters	: Confirm within limits.
8.	Landing lights (if used)	: OFF. LANDING LT ON advisory message suppressed.

CAUTION

Turn both pitot heaters on for flight in visible moisture and/or in rain, regardless of ambient temperature.

9.	Pitot heat	: As required.
10.	Compass	: Confirm all synchronized.
11.	Radios / Navigation	: As required.

Note

When the VHF (COM) systems are tuned on frequencies which are within ±4 MHz, the transmitting radio can produce loss of sensitivities on the receiving radios.

SEARCHLIGHT OPERATING PROCEDURE

CAUTION

Maximum speed for searchlight extension, orientation and retraction 135 KIAS.

Extension

1.	SEARCH CONTR / PWR circuit breakers	:ON.
2.	EXT / RETR / L / R switch (on collective grip)	: EXT (to extend light in the desired position).
3.	ON / OFF / STOW switch (on collective grip)	: ON.

Note

With the switch in OFF position the light remains extinguished in the position where it has been left.

4.	EXT / RETR / L / R switch (on collective grip)	: Set as necessary.

Note

Moving switch to L or R position the search-light rotates left or right. It is possible to adjust the light in an intermediate position, from stowed to extended, by temporarily moving the switch to EXT or RETR position.

Retraction

1.	ON / OFF / STOW switch (on collective grip)	: STOW then OFF.

Note

In STOW position the light is extinguished.

AUTOMATIC FLIGHT CONTROL SYSTEM

Collective Safety Function

The collective safety function (CSF) is active when any collective upper mode (ALT, ALTA,VS, RHT, GS, NGS, GA) is engaged. The function limits, if necessary, the collective commands to prevent:

— Exceedence of Maximum Continuous Power for torque, N1, TOT AEO or OEI. A PWR LIM alert message is displayed on the PFD when Power Limitation is active.

— Entering into autorotation (minimum torque 10% on each engine AEO or 20% OEI). A PWR LIM alert message is displayed on the PFD when Power Limitation is active.

— The aircraft descending through the safety height calculated as a function of descending speed and proximity to the ground. LOW HT alert message is displayed on the PFD when the Low Height Limitation is active.

When Collective Safety Function is active, engaged collective mode is annunciated as degraded (Mode annuciation in amber box flashing for 5 seconds then steady).

ILS APPROACHES

It is recommended that IFR approaches be carried out with both ILS1 and ILS2 receivers tuned on the same frequency.

EFIS SYSTEM

Refer to Chelton Pilot’s Operating Guide and Reference for more details on this installation.

FLIGHT MANAGEMENT SYSTEM

The procedures for the use of navigational equipment on Basic RNAV routes should include:

— During the pre-flight planning phase confirm the availability of GPS integrity (RAIM) for the intended flight (route and time). This should be obtained from a prediction program either ground-based or from an alternative method that is acceptable to the Autority.

— Dispatch should not be made in the event of predicted continous loss of RAIM of more than 5 minutes for any part of the intended flight.

— Check navigation database validity before the flight.

— Traditional navigation equipment should be selected to available aids so as to allow immediate cross checking or reversion in the event of loss of GPS navigation capability.

TAWS

— The FLTA function is automatically inhibited when in the Terminal, Departure, IFR Approach or VFR Approach modes and within 2 NM of the reference point.

— The FLTA function is automatically inhibited when IAS or groundspeed is below 50 Kts.

— GPWS Modes 3 is automatically inhibited when below 50 ft AGL (radar altimeter AGL altitude).

APPROACH AND LANDING

Note

The VFR approach function does not provide terrein or obstruction clearance. Extreme care should be taken when using the VFR approach function at night or in marginal VFR conditions.

Note

Instruction in blue are applicable only if kit P/N 109-B811-02 (fixed landing gear) is not installed.

1. Landing gear lever : DOWN.

   MAIN UTIL CHRG caution message is displayed during landing gear extension, then clears when landing gear is locked.

   Confirm landing gear is down and locked (3 green lights indicated, red light extinguished).

   If previously activated, LANDING GEAR caution message clears and audio warning ceases.

CAUTION

If flight operations require the landing gear to be down at all times then NR must be selected manually, using the NR switch, when appropriate.

2. N2 / NR : Below 75 KIAS confirm N2 / NR stabilizes at 102%.

3. Nose wheel lock : ON.

4. Parking brake : OFF. (ON if landing on a slope)

5. Landing lights : As required.

VERTICAL LANDING

CAUTION

While conducting a slope landing, care must be taken to avoid the tail making contact with the ground.

1. Landing path : Reduce the airspeed gradually and at 70 ft AGL flare and apply collective to pass 30 KIAS at 30 ft AGL. Bring the helicopter to a hover at 3 ft AGL. Descend vertically to the ground.

   After ground contact, lower the collective to the minimum pitch or as necessary if taxiing is required.

2. Pedal brakes : As necessary.

3. Nose wheel lock : OFF if taxiing is required.

RUNNING LANDING

1. Landing path : Reduce the airspeed gradually and apply collective to bring the helicopter to touchdown at a forward speed suitable for the landing surface and with minimum vertical speed.

   After ground contact, lower the collective as necessary to control the helicopter.

2. Pedal brakes : As necessary to stop the helicopter or to reach a suitable taxiing speed.

3. Nose wheel lock : OFF if taxiing is required.

SHUTDOWN

1. Nose wheel lock : ON.
2. Parking brake : ON.
3. Collective lever : Confirm fully down.
4. Cyclic stick and pedals : Centered and trimmed.

CAUTION

If there is evidence of any abnormal TOT increase after shutdown, perform a **DRY MOTORING RUN**.

CAUTION

Do not apply collective during rotor deceleration, particularly in windy conditions.
Below 85% NR, avoid any cyclic movement except to prevent droop stop contact.

5. ENG 1 and 2 MODE switches : Set to IDLE and then to OFF.

Note

If necessary, the engine may be shut down directly from FLT.

6. FUEL PUMP 1 and 2 switches : Set to OFF.

   FUEL PUMP 1 and FUEL PUMP 2 caution messages displayed.
7. FUEL VALVE 1 and 2 switches : CLOSED (bars horizontal).
8. XFEED VALVE switch : Confirm in AUTO position.

9.	Rotor brake	: Apply below 40% NR.
10.	PITOT 1 and 2 heat switches	: OFF.
11.	Miscellaneous switches	: OFF.
12.	MSTR AVNX switch	: As required.
13.	INV 1 and INV 2 switches	: OFF.
14.	External lights	: OFF.
15.	Cockpit lights	: OFF.
16.	Rotor brake lever	: Disengaged (fully forward).
17.	BAT, GEN 1 and GEN 2 switches (gang bar)	: OFF only when N1 at 0%.

POST FLIGHT CHECK

If conditions require, perform the following, referring to Section 8 for additional information:

1. Pitot-static tubes, engine intake and exhaust covers : Installed.

CAUTION

Following a slope landing, the parking brake holding capability is guarantee for 2 minutes from rotor stop.

2. Wheel chocks : As required.

FLIGHT HANDLING CHARACTERISTICS

Handling is conventional in normal forward, sidewards and rearwards flight manoeuvres. Flight control forces may be adusted by altering friction settings.

ENGINE LIMIT GOVERNING

The helicopter is provided with an engine torque limiter function aimed at reducing pilot's workload. However, it is still the pilot's responsibility to operate and maintain the helicopter within engine and transmission limits.

The torque limiter can be enabled, upon pilot's decision, by pressing once the TQ LIM pushbutton on the collective grip. Disabling the torque limiter is obtained by pressing again the TQ LIM pushbutton.

At helicopter power-up, the torque limiter is disabled by default, and the ECU will not prevent the pilot from exceeding the applicable limits. With the torque limiter disabled, the ECU controls the engine torque according to the following values:

AEO:	Torque (TRQ1 + TRQ2)	324%
OEI:	Torque	180%

When the TQ LIM pushbutton is depressed (LIMITER ON advisory message displayed), the torque limiter is enabled and the ECU will prevent the pilot from exceeding the following applicable limits:

AEO:	Torque (TRQ1 + TRQ2)	220%
OEI:	Torque	162%

Note

Regardless of torque limiter status, N1 and TOT always remain limited at 103% and 970 °C respectively.

CLIMB SPEED

The climb speed recommended is the best climb speed of the helicopter, Vy.

Vy is 80 KIAS up to 10000 ft, and then decreases by 2 kts every 1000 ft to become 60 KIAS at 20000 ft.

AUTOROTATIVE DESCENT

1. Maintain speed within Power-OFF limits.

2. Reduce collective to enter autorotation.

3. Adjust collective to maintain NR within appropriate limits.

4. Adjust attitude to obtain desired speed:

   - Maximum range is obtained at approximately 120 KIAS and minimum NR.

   - Minimum rate of descent is obtained at Vy and minimum NR.

   - Maximum rate of descent is obtained at Vne Power-OFF and maximum NR.

5. To recover to powered flight, increase collective pitch until freewheels are joined and finally, increase power to stop the rate of descent.

STEEP APPROACHES AND VERTICAL DESCENT MANOEUVRES

Low speed steep approaches (up to 20 kts) and vertical descent manoeuvres should be performed with a rate of descent not exceeding 900 ft/min.

EDU 1 CAUTIONARY CAPTION DEFINITIONS

The following cautionary captions are displayed vertically on EDU 1 and are intended to convey to the crew information related to the state of the engines. The meaning of the captions is as follows:

CAS Caption	Colour	Engine State
MAN	Yellow	Engine N°1 or N°2 in Manual mode. (Cautionary caption vertically displayed along the corresponding engine's TRQ scale).
OEI	White / Yellow / Red	Engine N°1 or N°2 in OEI mode. (Advisory / cautionary / warning, caption, commanded by the ECU, vertically displayed along the corresponding engine's N1, TOT and TRQ scales).
TNG	Yellow	Engine N°1 or N°2 in Training mode. (Cautionary caption vertically displayed along the corresponding engine's N1, TOT and TRQ scales).

EDU 1 ADVISORY CAPTION DEFINITIONS

The following advisory captions are displayed vertically on EDU 1 and are intended to convey to the crew information related to the state of the engines. The meaning of the captions is as follows:

CAS Caption	Colour	Engine State
START	Green	Engine N°1 or N°2 in Start mode. (Advisory caption vertically displayed along the corresponding engine's N1 scale.)
IGN	Green	Engine N°1 or N°2 igniter active. (Advisory caption vertically displayed along the corresponding engine's TOT scale.)
IDLE	Green	Engine N°1 or N°2 in Idle. (Advisory caption vertically displayed along the corresponding engine's N2 scale.)

EDU 2 ADVISORY CAPTION DEFINITIONS

The following advisory captions are displayed in green on EDU 2 and are intended to convey to the crew non critical information related to the state of various aircraft systems. The meaning of the advisory captions is as follows:

CAS Caption (Green)	System State
LIMITER ON	Engine torque limiter ON. ECU limits engaged.
FUEL XFEED	Fuel cross feed valve open. In EDU REVERSIONARY mode the caption FUEL XFEED will appear in the CAS message area.
LANDING LT ON	Landing lights switched on.
PITOT 1 HEAT	N°1 Pitot heating on.
PITOT 2 HEAT	N°2 Pitot heating on.
VENT ON	Vent fan running.
P/R TRIM OFF	Cyclic force trim off.
C/Y TRIM OFF	Collective/pedal force trim off.
HOIST ON	External hoist ON (if installed).
PRI SRCH LT	Primary searchlight ON.
SEC SRCH LT	Cabin searchlight ON (if installed).

VOICE ONLY ADVISORY

Voice Message	System State
‘TWO HUNDRED FEET’	When helicopter descends below 200 ft RAD ALT.

EFIS WARNING / CAUTION / ADVISORY SYSTEM

The EFIS system has an integrated audio/visual warning system that monitors a wide variety of parameters and provides annunciations for conditions that demand pilot awareness. There are four categories of annunciations: advisories, AFCS alerts, cautions and warnings.
Warnings are displayed with red flags and an aural annunciation that repeats until the condition goes away or is acknowledged by the pilot.
Cautions are displayed with yellow flags and a single aural annunciation.
AFCS alerts are displayed with grey flags and no aural annunciation.
Advisories are displayed with blue flags and a single aural annunciation.
The volume of aural annunciations are adjusted according to severity as follows:

warnings = Full Volume set into aircraft limits

cautions = .8 times volume set into aircraft limits

advisories = .6 times volume set into aircraft limits

Active aural annunciation is immediately muted by pressing the CAS RESET button on the collective grip.

Flags are visually prioritized so that active warning flags are displayed above active caution flags, which are displayed above active advisory flags. Within categories, active flags are stacked in chronological order so that the most recent annunciation appears on top.

Table 2-1 Warnings Annunciation Captions

Flag	Aural Annun.	Description of Indication	Page
OBSTRUCTION	"Warning, Obstruction"	Obstruction within TAWS FLTA warning envelope	Sect 7
TERRAIN	"Warning, Terrain"	Terrain cell within TAWS FLTA warning envelope	Sect 7

Table 2-2 Caution Annunciation Captions

Flag	Aural Annun.	Description of Indication	Page
ADC1(2) FAIL	Alert Tone	ADC #1(2) failure. Indicates no valid indicated airspeed, pressure altitude or VSI received from ADC #1(2)for more than 1 second.	3-170
AHRS1(2) FAIL	Alert Tone	AHRS #1(2) Failure. Indicates no valid bank, pitch or heading received from AHRS #1(2) for more than 1 second.	3-171
ALT MISCOMP	Alert Tone	Altitude Miscompare. Indicates that pressure altitude difference between ADC's is beyond limits. Inhibit for 10 minutes after startup. Limits are as follows: 10 - 20K' ∆70' < 10K' ∆50'	3-172
ATT MISCOMP	Alert Tone	Attitude Miscompare. Indicates that pitch or roll difference between AHRS is beyond limits (6°). Inhibit for 10 minutes after startup.	3-172
AUX SENSOR	"Auxiliary Sensor Failure"	Auxiliary Sensor. No valid message received from installed optional sensors. Sensor status displayed in FAULTS menu. This message applies to the following optional sensors: 1. RS-232 TAS System 2. WX-500 Lightning System	//
CHECK IDU1(2)	Alert Tone	Indicates that either: 1. the screen counter value has not changed in the last 1 second or 2. the intra-system monitor message is not fresh (i.e. no message received for longer than 1 second. The "#" parameter indicates which IDU is failing the check (IDU1 PFD, IDU2 MFD).	//
CHECK RANGE	"Check Range"	Less than 100NM buffer between calculated range and distance to: 1. The last waypoint if it is active; or 2. The airport if on a missed approach; or 3. Along-route distance to destination. Not activated in climbing flight.	//
DEC HT	"Decision Height"	Descending below decision height bug causes decision height readout to turn yellow and flash.	//

Table 2-2 Caution Annunciation Captions (Cont.d)

Flag	Aural Annun.	Description of Indication	Page
DR ##.##	None	GPS Dead Reckoning Mode. GPS in dead reckoning mode with valid ADC and AHRS data. Timer shows time since loss of navigation to indicate quality of DR solution.	//
EFIS COOL	"EFIS Cooling"	IDU Overtemperature. IDU core temperature greater than 95°C.	3-173
GPS LOI	Alert Tone	GPS Loss of Integrity.	3-177
GPS LON	Alert Tone	GPS Loss of Navigation.	3-176
GPS1(2) FAIL	Alert Tone	GPS #1(2) Failure. Indicates no valid message received from GPS #1(2) for more than 5 seconds.	3-174 3-175
GPS MISCOMP	Alert Tone	GPS Miscompare. Indicates that position, track or groundspeed difference between GPS units is beyond limits. Limits are as follows: Position: Enroute Mode 4NM Terminal Mode 2NM Departure Mode .6NM IFR Approach Mode .6NM VFR Approach Mode .6NM Track: If groundspeed is greater than 30 kts, miscompare if difference is more than 4°. Groundspeed: 10 kts.	3-178
GS MISCOMP	Alert Tone	Glideslope Miscompare. Indicates that difference between glideslope signals is beyond limits (0.25 Dots).	3-178
HDG MISCOMP	Alert Tone	Heading Miscompare. Indicates that heading difference between AHRS is beyond limits (6°). Inhibit for 10 minutes after startup.	3-179
IAS MISCOMP	Alert Tone	Airspeed Miscompare. Indicates that indicated airspeed difference between ADC's is beyond limits. Inhibit for 10 minutes after startup. Limits are as follows: >= 100KIAS Δ4KIAS < 100KIAS Δ10KIAS	3-180

Table 2-2 Caution Annunciation Captions (Cont.d)

Flag	Aural Annun.	Description of Indication	Page
IDU MISCOMP	Alert Tone	Indicates that a critical parameter being used by another display exceeds the miscompare thresholds when compared to the monitoring display. Compares the following critical parameters: - Attitude - Heading - Pressure Altitude - Indicated Airspeed - Localizer - Glideslope - Latitude - Longitude - Track - Groundspeed	//
IDU POWER	Alert Tone	Power Supply Fail. Indicates one of the dual redundant power supplies within an IDU is not functioning correctly.	3-180
LOC MISCOMP	Alert Tone	Localizer Miscompare. Only active when two valid localizers are being received. Indicates that difference between localizer signals is beyond limits (0.25 Dots).	3-182
MINIMUMS	"Minimums"	Minimum Altitude. Deviation from above to below minimum altitude bug. Causes minimum altitude readout to turn yellow and flash.	//
NO HEADING	Alert Tone	Heading Failure. No valid heading received from selected AHRS for more than 1 second.	3-171
NO GPS	"GPS Failure"	GPS Failure. No valid message received from selected GPS for more than 5 seconds.	3-174
NO TAWS	Alert Tone	TAWS FLTA Function Inoperative. Indicates that aircraft is currently beyond extent of terrain database or a failure condition exists that prevents the TAWS FLTA function from operating.	Sect 7
OAT SENSOR	Alert Tone	OAT Sensor Failed.	3-183

Table 2-2 Caution Annunciation Captions (Cont.d)

Flag	Aural Annun.	Description of Indication	Page
OBSTRUCTION	"Caution Obstruction"	Obstruction within TAWS FLTA caution envelope.	Sect 7
RADALT FAIL	Alert Tone	Radar Altimeter Failure. Indicates no radar altimeter reading received from Radar Altimeter for more than 1 second. Inhibited above 2000 ft AGL (where it is normal for radar altimeter to report no data).	3-184
SAME ADC	Alert Tone	Same ADC Source. Only active in dual-System (Pilot and Co-pilot), dual-ADC installation with good inter-System communications and neither ADC in failure condition. Indicates that both Systems are operating from same ADC source.	//
SAME AHRS	Alert Tone	Same AHRS Source. Only active in dual-System (Pilot and Co-pilot), dual-AHRS installation with good inter-System communications and neither AHRS in failure condition. Indicates that both Systems are operating from same AHRS source.	//
SAME GPS	Alert Tone	Same GPS Source. Indicates that both Systems are operating from same GPS source.	//
SAME NAV	Alert Tone	Same NAV Source. Indicates that both Systems are operating from same navigation source.	//
SCC FAIL	Alert Tone	SCC Card Failed. Indicates that SCC card could not be read upon power-up. This means that limits internal to the IDU are being used by the System.	3-185
TERRAIN	"Caution, Terrain"	Terrain cell within TAWS FLTA caution envelope.	Sect 7
TOO LOW	"Too Low Terrain"	Within GPWS Mode 3 envelope	Sect 7
TRAFFIC	"Traffic"	Traffic Advisory. Not given if own aircraft below 400 ft AGL. Not given if target is below 200 ft AGL (ground target). Audio not generated with TCAS-II system.	Supp. 21
VERT LON	Alert Tone	GPS loss of vertical navigation	1-24
XFILL FAIL	Alert Tone	Crossfill Failed. Indicates lack of inter-System communications. Inhibit for 30 seconds after startup.	//

Table 2-3 Advisory Annunciation Captions

Flag	Aural Annun.	System Condition	Page
- -	Chime	Countdown Timer Chime. Sounds chime when countdown timer reaches 00:00:00.	//
AHRS DG	Chime	Free Mode. Activated if AHARS 1 or 2 is in DG mode.	//
ANP: ##.##	Chime	GPS Actual Navigation Performance based upon current GPS HPL.	//
BARO MISCOMP	Chime	Barometric Setting Miscompare. Only active in dual-System (Pilot and Co-pilot) installation. Indicates mismatch of altimeter settings or altimeter modes between systems.	//
CHK BARO	Chime	Check Barometric Setting Ascending through transition level: Altimeter not set to 29.92 in. Hg or 1013 mB. Descending through transition level: Altimeter set to 29.92 in. Hg or 1013 mB. Descent warning times out in 10 seconds. Disabled during QFE operation.	//
FLTA INHBT	Chime	Shown when the FLTA function is automatically inhibited during normal operation.	//
LNAV APPR	Chime	GPS in LNAV Approach mode.	//
LNV/VNV APPR	Chime	GPS in LNAV/VNAV Approach mode.	//
LP APPR	Chime	GPS in LP Approach mode.	//
LPV APPR	Chime	GPS in LPV Approach mode.	//
MENU LOCK	None	Menu Locked. Menu system being used on another IDU.	//
PTK = ##	Chime	Parallel Offset. GPS Parallel Offset path advisory. ## is nautical miles left ("L") or right ("R") of main path.	//
RNP: ##.##A	Chime	GPS Automatic Required Navigation Performance as acquired from navigation database.	//
RNP: ##.##M	Chime	GPS Manual Required Navigation Performance as set by user.	//

Table 2-3 Advisory Annunciation Captions (Cont.d)

Flag	Aural Annun.	System Condition	Page
SUSPEND	Chime	Automatic Waypoint Sequencing Suspended. GPS automatic waypoint sequencing is suspended. Caused by being on final approach segment prior to arming missed approach, selecting manual GPS OBS, or being in holding prior to activating the "CONTINUE" tile.	//
TAS INHBT	Chime	Traffic Aural Inhibit. TAS aural inhibited through activation of TCAS/TAS Audio Inhibit.	Supp 21
TAWS INHBT	Chime	TAWS Inhibit. TAWS inhibited through activation of TAWS inhibit.	Sect 7
TAWS LOW ALT	Chime	TAWS Low Altitude Mode.	Sect 7
TERMINAL	Chime	GPS in Terminal mode.	//
TRUE NORTH	Chime	True North mode input discrete is asserted and system is operating in True North mode.	//
VECTORS	Chime	GPS in Vectors to Final Approach mode.	//
VFR APPR	Chime	GPS in VFR approach mode (active waypoint is part of VFR approach runway and within 6 NM of runway).	//
VSI BARO	Chime	Present vertical speed is barometric. The inertial vertical speed is lost When VSI BARO advisory is diplayed on Primary Flight Display on side in command, the AFCS LT, ALTA, VS, RHT, GA, GS, NGS Upper Modes are not available and IAS mode is degraded.	
VTF IFR APPR	Chime	GPS Vectors to Final IFR Approach Mode. GPS in Vectors to Final NPA mode	//
XFILL ARM	Chime	Crossfill Armed. Indicates that Systems are not synchronized and synchronization function is available.	//
XFILL INHBT	Chime	Crossfill Inhibited. Indicates that crossfill is manually inhibited through use of discrete input.	//

Table 2-4 AFCS Alerts Captions

Flag	Aural Annun.	System Condition	Page
LOW HT	None	Collective Safety Function active for Height below 15 feet or excessive closure rate approaching ground. Collective Safety Function active to control collective axis	//
PWR LIM	None	AP Collective Safety Function active	//
UCPL	None	AP Uncoupled	//
LINK FAIL	None	AFCS / EFIS Communication loss; AFCS indications on display are not available	//
> <	None	AP mode excessive deviation	//
	None	AP mode degradation	//
	None	AP mode transition	//

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SECTION 3

EMERGENCY AND MALFUNCTION PROCEDURES

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SECTION 3

EMERGENCY AND MALFUNCTION PROCEDURES

GENERAL

This section contains the procedures that must be performed in the event of an emergency or malfunction. These procedures are based on experience acquired in the operation of helicopters, in general, and on flight tests conducted on this particular model of helicopter.

The procedures used in each actual emergency or malfunction must result from consideration of the complete situation. Multiple emergencies or malfunctions may require a departure from normal corrective procedures detailed in this section.

All corrective action procedures listed herein assume the pilot gives first priority to helicopter control and safe flight path.

Note

Instruction and procedures in blue are applicable only if kit P/N 109-B811-02 (fixed landing gear) is not installed.

PROCEDURES LOGIC

The majority of the Emergency and Malfunction procedures that follow are presented in the form of logic trees (flow charts). These logic trees have been formulated based on analysis and test of the cockpit indications that would be available to the flight crew following the emergencies/malfunctions that are included in this section. For complex emergencies/malfunctions, cockpit indications coupled with the answers to “Yes/No” type questions (as indicated on the charts) should enable the flight crew to analyze the type of emergency/malfunction that has occurred, the branch of the “tree” that should be followed and the corrective action that should be taken.

In order to analyze some types of emergencies/malfunctions, answers to "+", "IF", "AND" and "OR" statements may be required. In these cases, the statements are presented in bold text ("**+**"**,** "**IF**"**,** "**AND**"**,** "**OR**") to be more conspicuous. It is emphasized that attention should be paid to this symbology to avoid a mistake in the emergency/malfunction analysis and subsequent incorrect crew action. **Required** crew actions are also presented in bold type.

DEFINITIONS

SAFE OEI FLIGHT

In general, safe OEI flight is defined to mean flight with one engine inoperative and:

1. a sustainable airspeed of not less than 45 KIAS,
2. the ability to obtain a positive rate of climb at acceptable power levels and
3. an altitude which provides sufficient clearance from the ground / obstacles / clouds so that required manoeuvring can be reasonably achieved.

EMERGENCY LANDING GUIDANCE

Throughout this Section, three terms are used to indicate the degree of urgency with which a landing must be effected. In cases where extremely hazardous landing conditions exist such as dense bush, heavy seas or mountainous terrain, the final decision as to the urgency of landing must be made by the pilot.

1. **Land immediately:** — Land at once, even if it means landing on water. Landing is the highest priority. The primary consideration is to assure the survival of the occupants. The consequences of continued flight are likely to be more hazardous than those of landing at a site normally considered unsuitable.

2. **Land as soon as possible:** — Do not continue flight for longer than is necessary to achieve a safe and unhurried landing at the nearest site.

3. **Land as soon as practicable:** — Land at the nearest aviation location or, if there is none reasonably close, at a safe landing site selected for subsequent convenience. Extended flight beyond the nearest approved landing area is not recommended.

PILOT ALERTNESS LEVEL

The level of alertness required by the pilot is a function of the flight regime.

Throughout this section, the following terms are used:

1. **Fly Attentive:** — Pilot to maintain close control of the flight path using hands-on when required.

2. **Fly Manually:** — Pilot directly controls the flight path using hands-on.

IDS CREW ALERTING SYSTEM

The Crew Alerting System (CAS) is a dedicated window in the lower portion of the EDUs, Ref. Figure 3-1. Many of the emergencies/malfunctions described in this section are indicated by the illumination of red CAS warning messages on EDU1 and with the flashing of the red Master Warning Light (MWL), Ref. Figure 3-2, or by the illumination of yellow CAS caution messages on EDU1 with the flashing of the yellow Master Caution Light (MCL), Ref. Figure 3-2.

The red warnings are accompanied by an audio tone and a voice warning and can be acknowledged by pressing the flashing red MWL (on the instrument panel) or the Master Reset pushbutton (on the collective grip). By cancelling the flashing red MWL, the audio tone and voice warning are also cancelled and the light is reset to indicate any future warnings.

The yellow cautions, which may be accompanied by a voice caution, can be acknowledged by pressing the flashing yellow MCL (on the instrument panel) or the Master Reset pushbutton (on the collective grip). This action also cancels the flashing yellow MCL and voice cautions if present and the light is reset to indicate any future cautions.

Green advisory and cyan status / maintenance messages are presented on the CAS system on EDU2 (secondary). Refer to Section 2 for more details.

In this section, the following convention is used:

WARNING	- A CAS RED WARNING
CAUTION	- A CAS YELLOW CAUTION

The ENG 1(2) FIRE warning message is displayed in white lettering on a red background:

ENG 1(2) FIRE

Note

In this manual, use of the caption notation #1(2) OIL PRES (for example) indicate captions that may apply to more than one component. An emergency involving N°2 engine oil pressure (for example) would be displayed as #2 OIL PRES.

The EDUs present a specific area partitioned in three columns, each column capable of displaying up to 3 lines of 14 characters, for warning, caution, advisory and status messages.

The display of the messages uses the following prioritization:

Priority 1: Warnings (red)

Priority 2: Cautions (yellow)

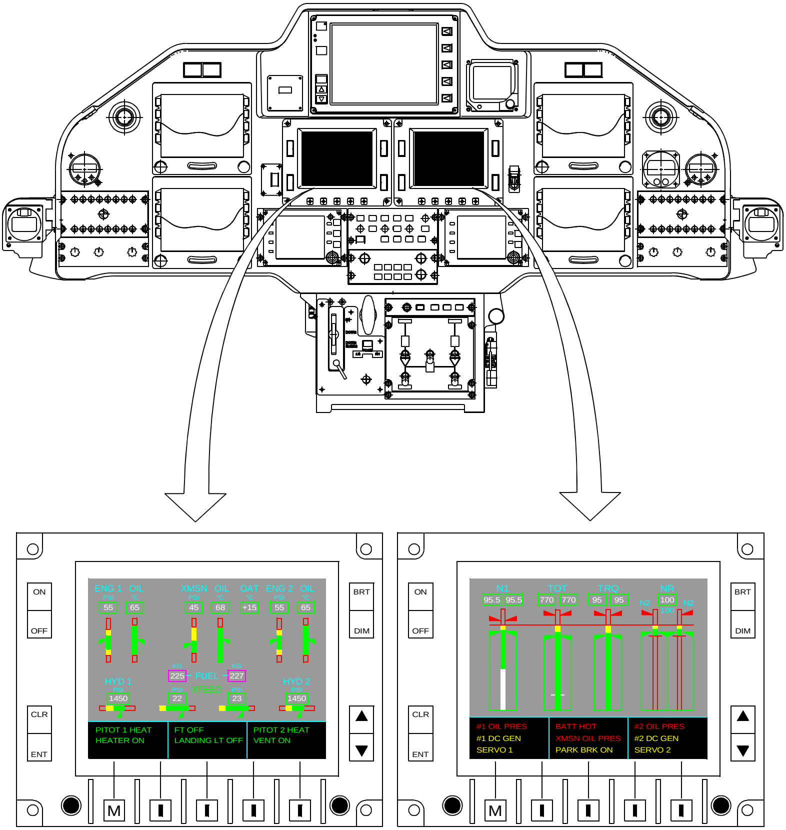
Priority 3: Advisory messages (green)

Priority 4: Status / Maintenance messages (cyan)

The “last-in” message is always displayed at the top of the relative column on the dedicated screen area, and displaces the list down. When messages are listed on more pages, the scroll function can be activated through the rocker switch with vertical arrows positioned on the right side of the EDU control panel.

Warning messages cannot be scrolled off the CAS display area.

Warning and Caution messages remain displayed until the cause of the condition has been corrected.



EDU 2 - ADVISORY AND STATUS MESSAGES

EDU 1 - WARNING AND CAUTION MESSAGES

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Figure 3-1 CREW ALERTING SYSTEM Layout

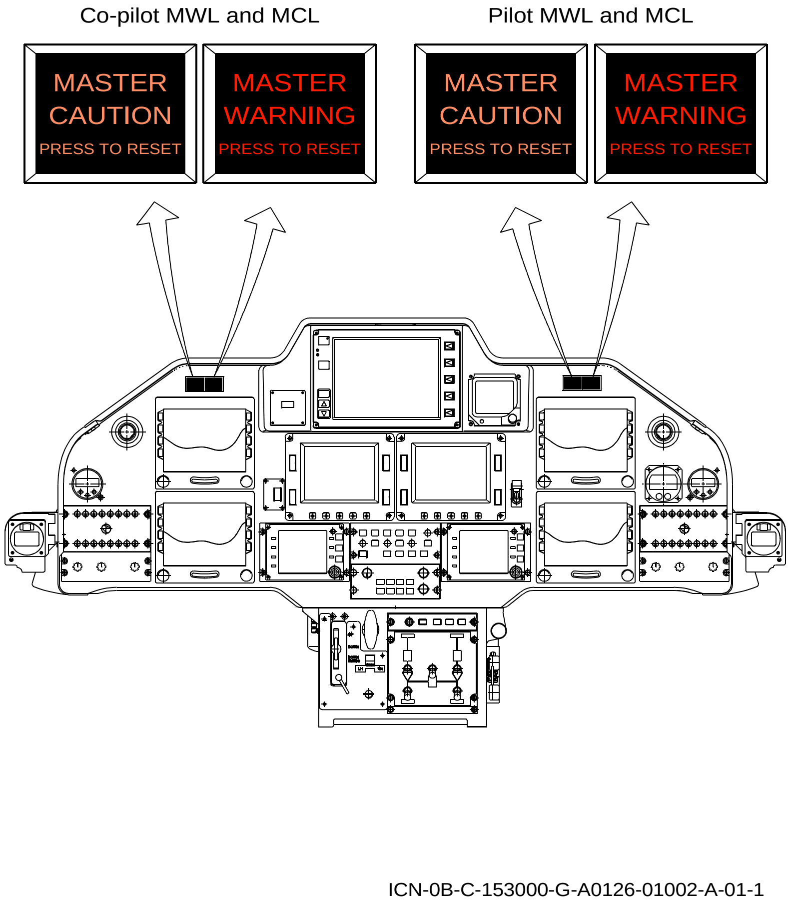


Figure 3-2 CREW ALERTING SYSTEM MWL and MCL Location

EFIS ALERTING SYSTEM

Many of the malfunctions described in this section are indicated by the illumination of a yellow caption on the pilot and/or copilot PFD's or MFDs (the captions are displayed on the functioning display in case that the other is failed).

Whenever a Caution illuminates, appropriate actions should be taken to deal with the indicated malfunction.

Cautions are displayed with yellow flags and a single aural annunciation.

Active aural annunciation is immediately muted by pressing the CAS RESET button on the collective grip.

The Caution flag remains until the cause of the caution is corrected.

Caution flags are stacked in chronological order so that the most recent annunciation appears on top. The caution flags not displayed on the screen are available scrolling the list with the left rotary knob of the display.

EMERGENCY PROCEDURES

IDS CAS WARNING MESSAGES

Whenever a warning illuminates on the IDS, appropriate actions must be taken to deal with the indicated emergency or malfunction.

Table 3-1 lists the Warning captions, the voice warnings and the corresponding emergency displayed by the IDS system.

Table 3-1 Table of IDS Warning Messages

CAS Caption	Voice Warning	Audio Tone	Emergency/System State
ROTOR LOW Page 3-15	ROTOR LOW	Tone + Horn	Rotor speed below 96% (Power-ON). Rotor speed below 95% (Power-OFF). When NR reaches 80%, voice warning, audio tone and horn are deactivated.
ROTOR HIGH Page 3-16	ROTOR HIGH	Tone	Rotor speed above 105% (Power-ON). Rotor speed above 110% (Power-OFF).
ENG 1(2) OUT Pages 3-18/3-19	ENGINE 1(2) OUT	Tone	Engine #1(2) N1 abnormally low (N1 35% and below). Probable engine failure.
ENG 1(2) FIRE Pages 3-22 / 3-23	ENGINE 1(2) FIRE	Tone	Fire in engine #1(2) compartment.
#1(2) ECU FAIL Page 3-27	WARNING	Tone	Critical hardware failure of the #1(2) engine electronic control unit (ECU). Automatic reversion of engine #1(2) to manual mode.
#1(2) OIL PRES Page 3-26	WARNING	Tone	Oil pressure in associated engine outside the operating range.

Table 3-1 Table of IDS Warning Messages (Cont.d)

CAS Caption	Voice Warning	Audio Tone	Emergency/System State
XMSN OIL PRES Page 3-29	WARNING	Tone	Low pressure in transmission lubricating system. Oil pressure below 30 psi.
XMSN OIL HOT Page 3-30	WARNING	Tone	Overheating of transmission lubricating system. Oil temperature above 120 °C.
ROTOR BRK ON Page 3-43	WARNING	Tone	Rotor brake in operation. Braking pads of rotor brake not in fully retracted position.
BATT HOT Page 3-52	WARNING	Tone	Battery overheating.
BATT DISCH Page 3-51	WARNING	Tone	Battery discharging. Output voltage of both generators below 27.0 V (± 0.5 V).
ELECTRICAL Page 3-44 / 3-46	WARNING	Tone	Failure of both DC generators.

The system logic will cause a higher priority voice warning to interrupt a lower priority voice message in accordance with the priority sequence listed on the next page.

AWG TEST PROCEDURE

The AWG test function can be initiated by holding the AWG switch in the TEST position. The aural message "TEST OK" will be generated to indicate a successful test. Maintaining the TEST position for more than 6 seconds, will generate the entire voice warning sequence in the following priority:

Tone 1	"ROTOR LOW"
Tone 2	"ENGINE ONE OUT"
Tone 2	"ENGINE TWO OUT"
Tone 3	"FIRE ENGINE ONE FIRE"
Tone 3	"FIRE ENGINE TWO FIRE"
Tone 4	"WARNING"
Tone 4	"ROTOR HIGH"
Tone 4	"AUTOPILOT"
Tone 4	"ENGINE IDLE"
Tone 5	"OVERTORQUE"
Tone 5	"AIRSPEED"
Tone 6	"LANDING GEAR"
Tone 6	"DECISION HEIGHT"
Tone 7	"TWO HUNDRED FEET"
Tone 7	"ALTITUDE"
Tone 8	Tone (for autopilot)

EMERGENCY AUTOROTATION PROCEDURE

The procedure which follows outlines the steps required to execute a successful emergency power-off landing. Time permitting, consult the appropriate Emergency procedure for the additional steps required to deal with a specific emergency.

1. Collective pitch — Reduce to enter autorotation.

2. Attitude — Adjust as required to obtain and maintain desired airspeed.

Note

An airspeed of Vy and 95% RPM ensures the minimum rate of descent in autorotation.

3. Collective pitch — Adjust as required to maintain rotor speed within limits. (95% to 110% NR)

4. Landing gear — DOWN.

5. Parking brake — Check OFF

6. Landing site — Select and manoeuvre into wind.

7. Briefing — Brief cabin crew and occupants.

8. Harness — Tight.

If time and conditions permits:

- ENG MODE switches — Both OFF.
- FUEL PUMP switches — Both OFF.
- FUEL VALVE switches — Both CLOSED.
- XPND / RADIO — Set emergency / transmit distress call.

9. Flare — At approximately 100 to 70 ft AGL, depending on the weight, initiate a flare, at approximately 10 deg per second, to a maximum 30 deg nose-up angle to reduce the rate of descent (500 ± 100 ft/min) and the forward speed (30 ± 10 KIAS).

10. Collective pitch — Adjust, as required, to maintain NR at 110% maximum during the flare.

11. Pitch attitude / Collective pitch — At approximately 10 ft AGL, reduce pitch attitude to a near level attitude. As the helicopter settles, apply collective pitch, as required, at approximately 4 ft to cushion touchdown.

12. Touchdown airspeed — As required by surface characteristics. If terrain permits, land with forward speed.

13. Collective pitch — Following touchdown, lower promptly.

14. Toe brakes — Apply as required.

15. Shutdown — Execute the **EMERGENCY/ POST CRASH SHUTDOWN AND EGRESS** procedure Page 3-21.

Note

Refer to Section 2 for FLIGHT HANDLING CHARACTERISTICS in AUTOROTATIVE DESCENT .

ROTOR UNDER SPEED

ROTOR LOW + Audio Tone and Horn and Voice Warning "ROTOR LOW"

Rotor NR outside limits:
Below 96%, Power-ON.
Below 95%, Power-OFF.

- Check NR.

IF rotor speed low.

Lower collective to increase rotor speed.

ROTOR OVERSPEED

ROTOR HIGH + Audio Tone and Voice Warning "ROTOR HIGH"

Rotor NR outside limits:
Above 105%, Power-ON.
Above 110%, Power-OFF.

Check NR.

IF rotor speed high.

Raise collective to decrease rotor speed.

ENGINE FAILURES

GENERAL

In the event of partial or complete power failure the establishment of a safe flight condition is the prime consideration until the cause of the failure can be analyzed.

CAUTION

Care should be taken in confirming the failed engine prior to commencing engine shutdown as given in the **ENGINE SHUT-DOWN IN AN EMERGENCY** procedure Page 3-20.

ENGINE FAILURE RECOGNITION

The following cues will be available to the crew following a single or double engine failure:

— Illumination of the MWL and CAS Warning ENG 1(2) OUT caption.

— An audio tone and an "ENGINE 1(2) OUT" voice warning (activated when N1 is 35% or below).

— Reduction in N1, N2, TOT and TRQ on effected engine.

— Depending on collective position at the time of the failure(s), a drop in rotor speed (NR) may occur, accompanied by a yawing tendency.

ENGINE FAILURE

ENG 1(2) OUT + Audio Tone and Voice Warning "ENGINE 1(2) OUT"

On affected engine:
N1, N2, TRQ and TOT rapidly decreasing.
Probable engine failure.

Adjust collective as required to maintain NR and live engine parameters within appropriate limits.

Achieve Safe OEI Flight

Carry out ENGINE SHUTDOWN IN AN EMERGENCY procedure Page 3-20..

Consider restart only if cause of failure is known, no mechanical failure is suspected and relight is essential.

engine restart possible .

YES

Refer to ENGINE RESTART IN FLIGHT procedure Page 3-90.

IF engine restart fails **land as soon as practicable**.

NO

Land as soon as practicable.

DOUBLE ENGINE FAILURE

Enter into autorotation immediately.

See **EMERGENCY AUTORATION PROCEDURE**.

If time and conditions permit, carry out the **ENGINE SHUTDOWN IN AN EMERGENCY** procedure Page 3-20 while the helicopter is manoeuvred towards the landing area.

If sufficient additional time is available, attempt an engine re-start, refer to the **ENGINE RESTART IN FLIGHT** procedure Page 3-90.

ENGINE SHUTDOWN IN AN EMERGENCY

CAUTION

Care should be taken in confirming the failed engine prior to commencing this shut-down procedure.

Following an engine failure/malfunction, achieve Safe OEI Flight. On the failed engine, carry out the following shutdown procedures:

1. ENG MODE switch — OFF.
2. Engine power lever — OFF.
3. FUEL PUMP switch — OFF.
4. FUEL VALVE switch — CLOSED.
5. XFEED VALVE switch — CLOSED.
6. GEN switch — OFF.

CAUTION

If there is evidence of combustion (i.e. a rise in TOT) after engine shutdown in flight, perform **DRY MOTORING RUN** procedure, as per Section 2, to prevent any possible fire.

7. Fuel contents — Monitor and use XFEED VALVE as necessary.

Land as soon as practicable. If terrain permits, land maintaining some forward speed.

EMERGENCY/POST CRASH SHUTDOWN AND EGRESS

In the event of an emergency or crash landing, priority must be given to ensure that personnel are egressed safely at the most appropriate time.

1.	ENG 1 & 2 MODE switches	— OFF.
2.	Engine power levers 1 & 2	— OFF.
3.	FUEL PUMP 1 & 2 switches	— OFF.
4.	FUEL VALVE 1 & 2 switches	— CLOSED.
5.	If risk of engine fire exists:	
5a.	S/OFF FIRE pushbutton	— Lift appropriate guard and press pushbutton (once only), S/OFF wording illuminates.
5b.	- Fire extinguisher switch	— Select BTL to 1 then, if required, select BTL to 2.
6.	Rotor brake	— ON.
7.	BAT switch	— OFF.
8.	GEN 1 & 2 switches	— OFF.

When rotor and helicopter stopped:

9.	Helicopter	— Egress as soon as possible.

FIRE

ENGINE FIRE ON GROUND

ENG 1(2) FIRE + Audio Tone and Voice Warning
"ENGINE 1(2) FIRE"

- affected engine power lever grip illuminated.
- affected engine control panel FIRE light illuminated.
- affected engine Eng 1(2) S/OFF FIRE pushbutton FIRE wording illuminated on fire extinguisher panel.

- Confirm engine fire.

- Parking brake:	**ON.**
- ENG MODE 1 & 2 switches:	**OFF.**
- FUEL PUMP 1 & 2 switches:	**OFF.**
- FUEL VALVE 1 & 2 switches:	**CLOSED.**
- XFEED VALVE switch:	**CLOSED.**

On affected engine:

- S/OFF FIRE pushbutton: **Lift appropriate guard and press pushbutton (once only).** S/OFF wording illuminates.

Fire extinguisher switch: BTL to 1.

fire warning extinguished

YES

NO

- Fire extinguisher switch: BTL to 2.

Carry out EMERGENCY/POST CRASH SHUTDOWN AND EGRESS procedure Page 3-21.

ENGINE FIRE IN FLIGHT

ENG 1(2) FIRE **+** Audio Tone and Voice Warning "ENGINE 1(2) FIRE"

- affected engine power lever grip illuminated.
- affected engine control panel FIRE light illuminated.
- affected engine ENG 1(2) S/OFF FIRE pushbutton FIRE wording illuminated on fire extinguisher panel.

Achieve Safe OEI Flight

On affected engine:
ENG MODE switch: IDLE.

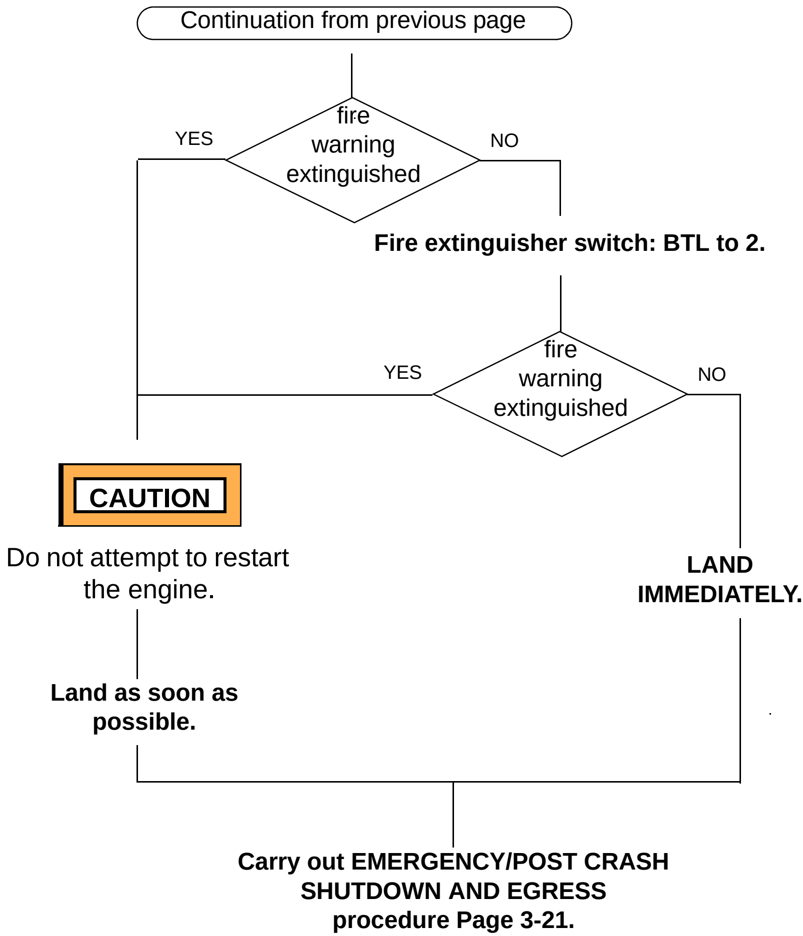
Confirm engine fire.

On affected engine:

- ENG MODE switch:	**OFF.**
- FUEL PUMP switch:	**OFF.**
- FUEL VALVE switch:	**CLOSED.**
- XFEED VALVE switch:	**CLOSED.**
- S/OFF FIRE pushbutton:	**Lift appropriate guard and press pushbutton (once only).** S/OFF wording illuminates.
- Fire extinguisher switch:	**BTL to 1.**
- GEN switch:	**OFF.**

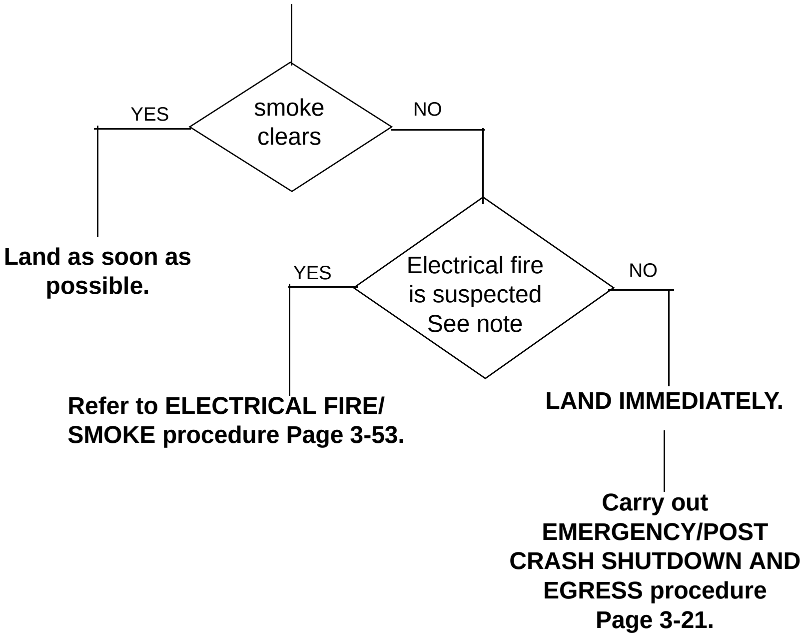
Continued Next Page

ENGINE FIRE IN FLIGHT (CONT'D)



SMOKE IN CABIN, TOXIC FUMES, ETC.

- **Front ventilation ports:** **Open.**
- **VENT CKPT switch:** **Set to HIGH.**
- **Sliding windows (if installed):** **Open.**
- **ECS / Heater (if installed) :** **Check OFF.**
- **Cockpit utility light :** **ON (at night).**

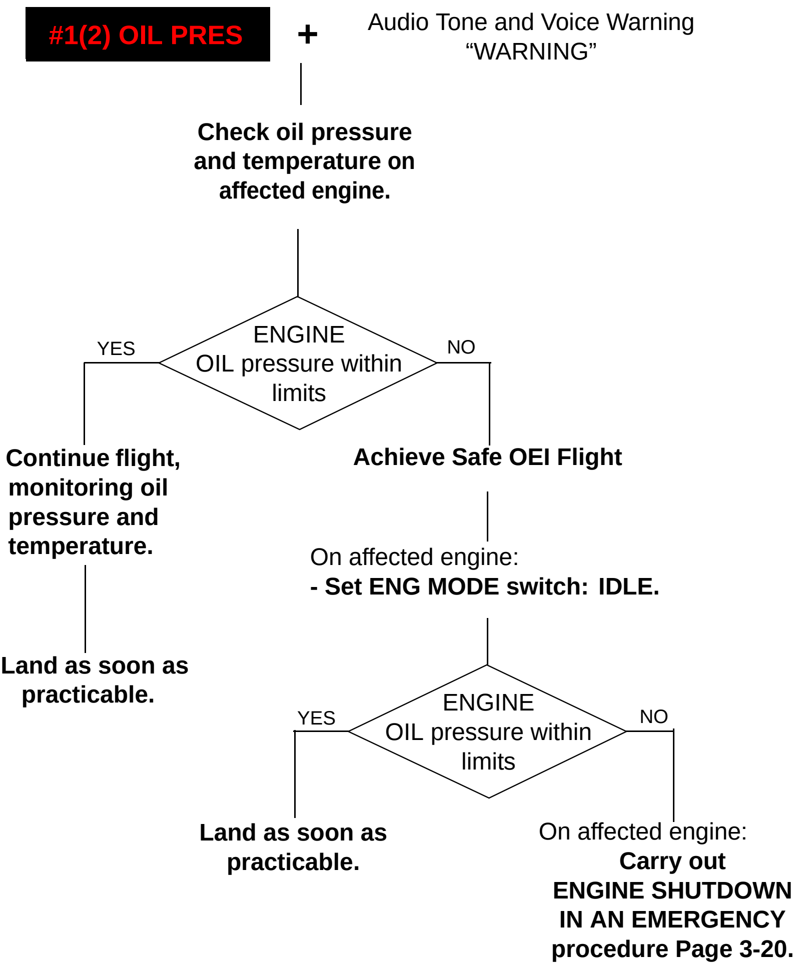


Note

Electrical fire is suspected when there are visible signs of smoke in cockpit, with distinct acrid smell of burning insulation.

ENGINE SYSTEMS

ENGINE OIL PRESSURE



ENGINE ECU FAIL

#1(2) ECU FAIL + Audio Tone and Voice Warning "WARNING"

Critical hardware failure of the engine electronic control unit (ECU).
Automatic reversion to manual mode of the affected engine.

Check MAN mode legend on EDU1.

Check engine parameters on affected engine.

Cycle ENG GOV switch to MANUAL, and then from MANUAL to AUTO.

warning message clears

YES: **Continue flight.**

NO: **Control engine using ENG TRIM or engine power lever manual control.**

Land as soon as practicable.

Note

In the presence of an ECU failure, the engine control system reverts to MANUAL mode regardless of the ENG GOV switch position. However, it is recommended to set the ENG GOV switch to MANUAL for congruence with the mode condition.

TRANSMISSION SYSTEM FAILURES

The most common transmission system (main and tail rotor gearbox) failures are of three general types:

1. Lubrication system failure (oil pump, ducts, nozzles, etc)

2. Transmission component failure (gears, bearings, etc)

3. Accessory component failure (coolers, etc)

The main gearbox is monitored with oil pressure and oil temperature indicators and chip detectors, whilst the tail rotor gearbox is monitored with chip detectors. These indicators and chip detectors, as well as the CAS Warning and Caution messages inform the pilot of the operating condition of the system. It is most likely that one or more of these indications will be present if a mechanical transmission failure is imminent. However, whether these indications are present or not, crew sensory perceptions such as:

- abnormal mechanical noise and/or
- heavy vibration levels and/or
- the odour of hot metal fumes

all play an important part in the diagnosis of impending transmission system failures and assist the pilot in determining what actions are required.

MAIN GEARBOX

OIL PRESSURE LOW

XMSN OIL PRES + Audio Tone and Voice Message “WARNING”

Transmission oil pressure below minimum limit.

Check XMSN OIL pressure.

XMSN OIL pressure within limits

YES

Continue flight monitoring oil pressure and temperature.

Land as soon as practicable.

NO

Reduce power as soon as operational conditions permit.

Land as soon as possible.

OIL TEMPERATURE HIGH

XMSN OIL HOT + Audio Tone and Voice Message "WARNING"

Transmission oil temperature above maximum limit.

Check XMSN OIL temperature.

XMSN OIL temperature within limits

YES

Continue flight monitoring oil pressure and temperature.

Land as soon as practicable.

NO

Reduce power as soon as operational conditions permit.

Land as soon as possible.

MAIN ROTOR CONTROLS SEIZURE

WARNING

If a seizure occurs in the flying controls, greater forces will be required to operate the controls. A reduction in the available control ranges may result and, in this situation, the low speed flight envelope may be restricted.

If the seizure occurred at an airspeed greater than 25 KIAS, the helicopter should be landed into the wind as soon as possible using a running landing procedure and a touchdown speed of approximately 25 KIAS.

If the airspeed is less than 25 KIAS, carry out a running landing at the speed at which the seizure occurred. If the helicopter is in a hover, land vertically.

TAIL ROTOR SYSTEM FAILURES

GENERAL

Tail rotor emergencies may be divided into three major categories:

1. Tail Rotor Drive failure:
   - Tail rotor drive failure in hover.
   - Tail rotor drive failure in forward flight, power-on.
   - Tail rotor drive failure in forward flight, low power or power-off.
2. Tail Rotor Control failure:
   - Tail rotor control failure in hover.
   - Tail rotor control failure in forward flight.
3. Tail Rotor Control Seizure:
   - Tail rotor control seizure in hover.
   - Tail rotor control seizure in forward flight.

Controllability Check

Controllability check consists of slow, progressive and cautious cyclic and collective manoeuvers in level flight at safe altitude aimed at determining the lowest airspeed at which the helicopter directional control can be maintained with the correct power setting. This airspeed should be recorded, and used as a minimum speed for touchdown.

Touchdown

Run-on landings and autorotation landings shall always be carried out nose wheel locked and park brake off. Toe brakes should be applied to slow down the helicopter only after the collective has been lowered to minimum pitch.

TAIL ROTOR DRIVE FAILURE

A tail rotor drive failure is a situation that involves a disconnect in the drive system, such as a severed drive shaft, wherein the tail rotor stops turning and thus delivers no thrust. This failure may be accompanied by noise, vibration or oscillation in the tail section.

The severity of the initial reaction, which is a sudden yaw rate, will be determined by the airspeed, altitude, gross weight, center of gravity and torque settings at the time that the failure occurs. The vertical fin produces an aerodynamic anti-torque component which is a function of forward speed. Fin effectiveness increases with higher airspeeds.

The severity of the resulting undemanded yaw due to a tail rotor drive failure, depends on the flight condition at the time of the failure. Three cases are to be distinguished:

- Tail rotor drive failure in hover.
- Tail rotor drive failure in forward flight, power-on.
- Tail rotor drive failure in forward flight, low power or power-off.

However in all three cases the following indications are present and determine a definite tail rotor drive failure:

- Attempt to maintain flight condition, may require unusual pedal position.
- Pedal input does NOT produce any sustained helicopter response.
- Possible noise and vibration from the aft fuselage area.

Although there might be some helicopter behaviour differences, the general procedure to bring the helicopter to a safe landing are basically the same in all flight conditions.

Following a tail rotor drive failure, safe landing can be achieved almost solely in autorotation. Even though in some circumstances, powered flight can be maintained or re-established through a correct combination of airspeed and power setting, autorotation must always be re-entered for a safe landing.

CAUTION

Powered run-on landing should not be attempted as airspeed required to maintain directional control is too high compared with landing gear limitations.

Tail rotor drive failure in hover

Further to the general indications of a drive failure, the helicopter will start a slow but accelerated swing to the right. Application of full left pedal does not diminish the phenomenon.

Collective: Lower immediately.

If time and conditions permit

- Engine power levers: Both OFF.

Collective: As required to cushion touchdown.

Carry out EMERGENCY/ POST CRASH SHUTDOWN AND EGRESS procedure Page 3-21.

CAUTION

If engines are not shutdown a yaw to the right can be expected when attempting to cushion the touchdown. The yaw rate will be a function of the weight and collective application.

Tail rotor drive failure in forward flight, power-on

Further to the general indications of a drive failure, the helicopter will exhibit symptoms that will differ depending on the combination of airspeed, power setting, density altitude, gross weight and rotor speed.

A possible behaviour is an undemanded yaw rate oscillating or rapidly diverging.

Tail rotor drive failure in forward flight, power-on (Continued from previous page)

Collective: Lower immediately.

Enter EMERGENCY AUTOROTATION as per procedure Page 3-13.

WARNING

Keep NR between 95% and 110%.

Airspeed: Vy.

suitable landing area found

YES

NO

Sufficient height to try to regain forward flight

NO

YES

Gently raise collective.

Find a TRQ / speed combination that will allow forward flight. If time and conditions permits review EMERGENCY AUTOROTATION PROCEDURE Page 3-13

Maintain EMERGENCY AUTOROTATION as per procedure.

ENG 1 & 2 MODE switches: Set to OFF.

Perform EMERGENCY AUTOROTATION landing as per procedure Page 3-13.

Note

If a run-on landing is not possible, decision should be made to minimize forward speed and accept the yaw rate.

Tail rotor drive failure in forward flight, low power or power-off

Depending on speed and torque applied, loss of thrust on the tail rotor may cause either LEFT or RIGHT yaw rate.

Two cases are thus possible:

1. Development of left yaw, typically coupled with a left roll.

2. Development of right yaw, typically coupled with a right roll and a pitch down attitude.

Note

In a low power descent or in autorotation, a tail rotor drive failure could cause the helicopter to yaw left because of the aerodynamics forces produced by the vertical fin.

Continued on next page.

Tail rotor drive failure in forward flight, low power or power-off (Continued from previous page)

Helicopter
develops **LEFT yaw.**

Development of left
yaw typically coupled
with left roll.

**Gradually raise collective to
regain directional control.**

**Maintain or carefully
reduce airspeed.**

Note

In this condition, a banked, unco-ordinated flight is acceptable to maintain directional control.

Helicopter
develops **RIGHT yaw.**

Development of right yaw
typically coupled with right
roll and pitch down attitude.

**Acquire EMERGENCY
AUTOROTATION
as per procedure Page 3-13**.

WARNING

**Keep NR
between 95%
and 110%.**

ENG 1 & 2 MODE SWITCHES: Set to OFF.

**Perform EMERGENCY AUTOROTATION
landing as per procedure**. **Page 3-13**

Note

If run-on landing is not possible, decision should be made to zero out forward speed and accept the yaw rate.

TAIL ROTOR CONTROL FAILURE

The indications of a tail rotor control failure are:

- Undemanded yaw to the left or to the right.
- Loss of yaw control:

  Pedals are free but ineffective or partially effective.

In the event of a control rod separation in the tail rotor system, between the pedals and the hydraulic servo actuator, the T/R pitch should remain at the last value retained by the hydraulic servo actuator before the failure.

In the event of a failure between the hydraulic servo actuator output and the tail rotor, the T/R pitch will be reduced to a value that will produce little or no thrust at all.

Once control of the helicopter is regained, if time and conditions permit, a controllability check at safe altitude should be performed.

Tail rotor control failure in hover

Collective: Lower to
LAND IMMEDIATELY.

Cyclic: As necessary to
maintain level attitude.

ENG 1 & 2 MODE switches: OFF.

Tail rotor control failure in forward flight

Helicopter develops **LEFT yaw.**	Helicopter develops **RIGHT yaw.**
Gradually raise collective and/or reduce speed to regain directional control.	**Gradually lower collective and/or increase speed to regain directional control.**

IF time and conditions permit:
- perform controllability check.

Approach: **Perform keeping nose to the left.**

Landing: **Touchdown at lowest airspeed obtained in the controllability check.**

Maintain collective on touchdown, then carefully lower it while slowing down.

Carefully use toe brakes to slow down helicopter.

Carry out EMERGENCY/POST CRASH SHUTDOWN AND EGRESS procedure Page 3-21.

Note

Wind from the front left quadrant of the helicopter may be beneficial.

TAIL ROTOR CONTROL SEIZURE

Tail rotor control seizure is characterised by locked pedals. Depending on power setting and airspeed combination, failure may not be detected until parameters are changed or until directional control application is required.

A control seizure is indicated by:

- Pedals seized or excessive force required.

If the pedals cannot be moved with a moderate effort, a control linkage fouling can be suspected.

CAUTION

If pedals cannot be moved with a moderate amount of force, do not attempt to apply maximum effort since a more serious malfunction could result.

Note

When conditions permit, a run-on landing should be planned, preferably on a flat strip/ runway, in order to minimise loss of directional control and avoid tipping over upon touchdown.
If a run-on landing is not possible, forward speed can be nearly zeroed by increasing the flare amplitude and duration. Landing will have to be accomplished while rotating since a yaw rate will develop.

Tail rotor control seizure in hover

Collective: **Lower gently to LAND IMMEDIATELY.**

Note

Lowering the collective may cause the helicopter to start rotating. Touching down while rotating at nearly zero ground speed should result in little or no hazard.

Note

Do not switch engines to OFF unless a severe right yaw develops.
If left yaw develops, reducing NR to 100% may be beneficial.

Carry out EMERGENCY/POST CRASH SHUTDOWN AND EGRESS procedure Page 3-21.

Tail rotor control seizure in forward flight

Speed and power setting to maintain directional control during approach and landing depends on the conditions at the time of failure.

In general:

- Slower touchdown speed results from a failure at low speed and high power setting.

- Faster touchdown speed results from a failure at high speed and/or low power setting.

If time and conditions permit:
- perform controllability check.

Approach and Landing:
Perform at lowest airspeed found in controllability check.

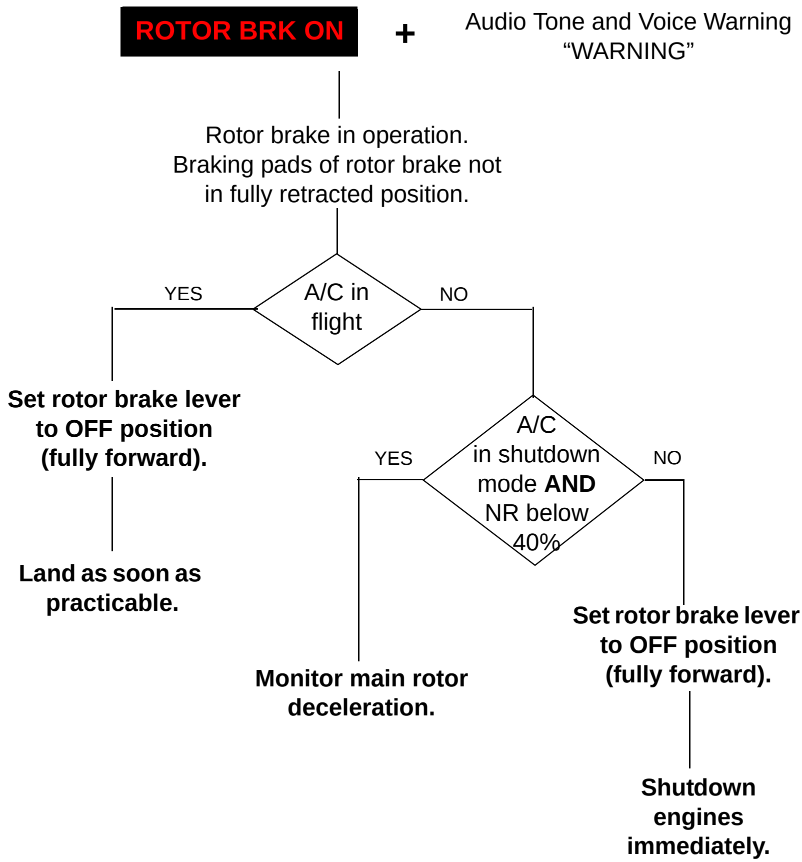
Maintain collective on touchdown, then carefully lower it while slowing down.

Carefully use toe brakes to slow down helicopter.

Carry out EMERGENCY/POST CRASH SHUTDOWN AND EGRESS procedure Page 3-21.

ROTOR BRAKE SYSTEM

ROTOR BRAKE ON



ELECTRICAL SYSTEM

DOUBLE DC GENERATOR FAILURE

ELECTRICAL + BATT DISCH + Audio Tone and voice warning "WARNING"

Both DC generators failed.

Cockpit pilot utility light: ON (at night)

RTU1: OFF.

Reduce airspeed to 128 KIAS maximum.

GEN BUS 1: Check OFF.
GEN BUS 2: Check OFF.

1 AP OFF + BUS TIE

caution messages displayed.

GEN 1 and GEN 2: Reset, then set to ON.

Both GEN remain ON

YES

GEN BUS 1 & 2: ON.

All caution and warning messages disappear.

AP1: ON.

Continue flight.

NO

Continued Next Page

DOUBLE DC GENERATOR FAILURE (CONT'D)

Continuation from previous page

#1(2) DC GEN caution message displayed

YES:

ELECTRICAL warning clears.

- GEN BUS 1: ON.
- GEN BUS 2: ON.

BATT DISCH and BUS TIE messages clear.

AP1: ON.

Continue flight.

Refer to FAILURE OF A GENERATOR procedure Page 3-127.

NO:

ELECTRICAL warning message still displayed.

Double DC generator failure confirmed.

GEN 1 & 2: OFF

Land as soon as possible (within 30 min. maximum).

Note

The battery endurance above assumes the VHF2 radio is operated in transmit for a maximum of 1 minute every 15 minutes.
The LDG LT or SCH LT can be turned ON for 1 minute prior to landing

Note

Refer to **SERVICES LOST DURING DOUBLE DC GENERATOR FAILURE** Page 3-50 for inoperative equipment.

Note

If MAIN BUS 1 & 2 are required, GEN BUS 1 & 2 may be switched ON. The battery endurance will be reduced to 20 minutes.

DOUBLE DC GENERATOR FAILURE
(EXTENDED FLIGHT ENDURANCE)

ELECTRICAL + **BATT DISCH** + Audio Tone and voice warning "WARNING"

Both DC generators failed.

Cockpit pilot utility light: ON (at night)

RTU1: OFF.

Reduce airspeed to 128 KIAS maximum.

GEN BUS 1: Check OFF.
GEN BUS 2: Check OFF.

1 AP OFF + **BUS TIE**

caution messages displayed.

GEN 1 and GEN 2: Reset, then set to ON.

Both GEN remain ON

YES

GEN BUS 1 & 2: ON.

All caution and warning messages disappear.

AP1: ON.

Continue flight.

NO

Continued Next Page

DOUBLE DC GENERATOR FAILURE (EXTENDED FLIGHT ENDURANCE)

(CONT'D)

Continuation from previous page

#1(2) DC GEN caution message displayed

YES:

ELECTRICAL warning clears.

- GEN BUS 1: ON.
- GEN BUS 2: ON.

BATT DISCH and BUS TIE messages clear.

AP1: ON.

Continue flight.

Refer to FAILURE OF A GENERATOR procedure Page 3-127.

NO:

ELECTRICAL warning message still displayed.

Double DC generator failure confirmed.

GEN 1 & 2: OFF.

Continued Next Page

DOUBLE DC GENERATOR FAILURE (EXTENDED FLIGHT ENDURANCE) (CONT'D)

Continuation from previous page

- Set XFEED VALVE: CLOSED.
- FUEL PUMP 1 and 2 switches: OFF.
- EDU2 switch: OFF.

WARNING

When the indicated fuel quantity in each tank is 100 kg or less, avoid bank turns above 30 deg and sustained pitch angle below 0 deg.

OAT less than 4 °C

YES:

Confirm PITOT HEAT 2 ON.

Land as soon as practicable within 43 minutes.

NO:

Confirm PITOT HEAT 2 OFF.

Land as soon as practicable within 54 minutes.

Note

The battery endurance reported above assumes the pilot operates the VHF2 radio system in transmission for a maximum of 1 minute every 15 minutes.

The LDG LT or SCH LT can be turned ON for 1 minute before landing.

Note

Refer to **SERVICES LOST DURING DOUBLE DC GENERATOR FAILURE** Page 3-50 for inoperative equipment.

SERVICES RETAINED DURING DOUBLE DC GENERATOR FAILURE

EMERGENCY BUS 1

INVERTER 1
ATC TRANSPONDER
APMS CHANNEL 1 CONTROL
LANDING GEAR HYDRAULIC CONTROL
LANDING GEAR CONTROL PANEL
POSITION LIGHTS
LANDING LIGHTS
EMERGENCY FLOATS (if installed)

EMERGENCY BUS 2

PILOT PFD (EADI)
PILOT RBP
PILOT FMS/GPS
PILOT ICS (AMU)
ICS SPEAKER AMPLIFIER
VHF 2
RTU 2
RADAR ALTIMETER
VOR/ILS 2
AHRS 2
PITOT HEAT 2
PILOT UTILITY LIGHT
SEARCHLIGHT
EMERGENCY FLOATS (if installed)

ESSENTIAL BUS 1

PILOT MFD (EHSI)
EDU SECONDARY
DAU CH-B
ENGINE 1 FIRE DETECTION
ENGINE 1 FIRE EXTINGUISHER
FUEL PUMP 1
FUEL SHUT-OFF VALVE 1
FUEL QUANTITY 1
FUEL CROSS FEED VALVE
ECU 1

ESSENTIAL BUS 2

HYDRAULIC CONTROL PANEL
ADU 2
APMS CHANNEL 2 CONTROL
AFCS CHANNEL 2 (AP 2)
ENGINE 2 FIRE DETECTION
ENGINE 2 FIRE EXTINGUISHER
FUEL PUMP 2
FUEL SHUT-OFF VALVE 2
FUEL QUANTITY 2
ENG GOV CONTROL
ECU 2

BATTERY BUS

EDU PRIMARY
DAU CH-A
ELT
ESIS
COPILOT ICS (AMU)
FORCE TRIM

SERVICES LOST DURING DOUBLE DC GENERATOR FAILURE

ELECTRICAL

BATT DISCH

MAIN BUS 1	MAIN BUS 2
COPILOT PFD (EADI)	COPILOT MFD (EHSI)
AHRS 1 (primary supply)	AHRS 1 (secondary supply)
VOR/ADF 1	EVS
RTU 1	TAS
VHF 1	DIGITAL MAP
CABIN ICS (PIA amplifier)	OVERHEAD CONSOLE ILLUMINATION
COPILOT RBP	INSTRUMENT PANEL ILLUMINATION
CHIP BURNER	TAXI LIGHTS
PEDESTAL ILLUMINATION	ANTI-COLLISION LIGHTS
COPILOT UTILITY LIGHT	COCKPIT FAN
CABIN LIGHTS	INVERTER 2
PASSENGER WARNING SIGN	PILOT WIPER
COPILOT WIPER	ENG 2 IGN/START (no ENGINE 2 restart)
ENG 1 IGN/START (no ENGINE 1 restart)	HYDRAULIC ACCUMULATOR
PITOT HEATER 1	(loss of MAIN & EMER PRESS indication)
HEATER/ECS (if installed)	

AUX BUS 1	AUX BUS 2
GPS 1	DME
ADU 1	

BATTERY DISCHARGING

BATT DISCH + Audio Tone and Voice Warning “WARNING”

Battery discharging.

- **Cockpit pilot utility light:ON** (at night)**.**
- **Reduce power** (if practicable)**.**
- **Reduce airspeed to 128 KIAS maximum.**

CAUTION

AP1 may be lost due to GEN 2, GEN BUS 2 and BAT switches disconnecting. AP1 may be re-engaged immediately.

ELECTRICAL Warning displayed

YES: Both DC generators failed.

Refer to DOUBLE DC GENERATOR FAILURE procedure.

NO:

- **GEN BUS 1: Check ON.**
- **GEN BUS 2: Check ON.**

BATT DISCH Warning remains

NO: **Continue flight.**

YES:

#1(2) DC GEN Caution displayed

YES: On affected generator:

- **GEN BUS switch: OFF.**
- **GEN switch: RESET, then to ON.**
- **GEN BUS switch: ON.**

Continued Next Page

NO: **BAT switch: OFF.**

Land as soon as practicable.

BATTERY DISCHARGING (CONT'D)

Continuation from previous page

BATT DISCH
Warning remains

YES

AP1: ON.

Land as soon as possible.

NO

Refer to BUS TIE OPEN procedure.

BATTERY HOT

BATT HOT + Audio Tone and Voice Warning "WARNING"

Battery temperature exceeding limits.

BAT switch: OFF.

Land as soon as practicable.

CAUTION

Do NOT attempt engine restart in case of engine flame-out.

ELECTRICAL FIRE / SMOKE (IN FLIGHT)

Visible signs of smoke in cockpit, with distinct acrid smell of burning insulation.

- Ventilate cockpit and cabin.

Maintain safe flight condition and use following procedure to isolate bus bars in order to establish source of fire.

Cockpit pilot utility light: ON (at night).

- GEN BUS 1: OFF.
- GEN BUS 2: OFF.
- GEN 1: OFF.
- VHF2 : Check tuned on required ATC frequency.

- BAT: OFF.
- AP 2: ON.

Loss of: EDU 1
ESIS
RTU
ATC TRANSPONDER
APMS ILLUMINATION
VHF1
LANDING LIGHTS
FM CONTROL PANEL

smoke clears

YES:

Land as soon as practicable.

NO:

- GEN 1: ON.
- AP2: Check ON.
- GEN 2: OFF.
- AP 1: ON.

Loss of: EDU 1
ESIS
FORCE TRIM
VHF2
SEARCHLIGHT

Continued Next Page

ELECTRICAL FIRE / SMOKE (IN FLIGHT) (CONT'D)

Continuation from previous page

smoke clears

YES

WARNING

Turning on the battery could cause the fire to re-ignite.

- BAT: ON.

smoke reappears

YES

- GEN 2: ON.

- AP 1: Check ON.

EMERG BUS 2 affected by fire.

No pilot action is possible.

Land as soon as possible.

IF smoke severe:
- LAND IMMEDIATELY.

Carry out EMERGENCY/POST CRASH SHUTDOWN AND EGRESS procedure Page 3-21.

NO

MAIN BUS 2 or AUX BUS 2 affected by fire.

Land as soon as practicable.

NO

- BAT: ON.
- GEN 2: ON.
- AP 2: Check ON.

ESS BUS 1 or 2 affected by fire.

No pilot action is possible.

Land as soon as possible.

IF smoke severe:
- LAND IMMEDIATELY.

Carry out EMERGENCY/ POST CRASH SHUTDOWN AND EGRESS procedure Page 3-21.

LANDING GEAR

LANDING GEAR FAILS TO LOCK DOWN
(TOTAL OR PARTIAL FAILURE)

IF, after selecting the landing gear DOWN, one or more green indicators remain blank or the red indicator remains on, do the following:

Check UTIL NORM pressure on EDU2 Auxiliary Mode.

pressure low

YES

Perform EMERGENCY DOWN LANDING GEAR procedure Page 3-56.

NO

Reduce speed and check if 3 green landing gear indicators illuminate.

Confirm LDG GEAR circuit breakers (2) not tripped.

Cycle LDG GEAR lever up to three times to achieve down lock.

Landing gear down and locked

YES

Continue to land.

NO

Perform EMERGENCY DOWN LANDING GEAR procedure Page 3-56.

EMERGENCY DOWN LANDING GEAR PROCEDURE

CAUTION

After extending the landing gear using the DOWN EMER lever position, the toe brakes and rotor brake are inoperative.
There is sufficient pressure in the emergency accumulator for only one extension of the landing gear, and for operation of the parking brake handle for emergency braking.

EMER/NORMAL selector:Turn clockwise breaking safety wire and selecting EMER.

Set landing gear lever to DOWN EMER (fully down).

Landing gear down and locked

YES

Land vertically and cautiously on flat level surface.

NO

Proceed to a hover at a height sufficient for ground personnel to check the gear.

STATIC PORT OBSTRUCTION

When operating in adverse weather conditions (rain, snow, etc.), if erratic readings from the airspeed indicator and altimeter are suspected, with the STATIC source switch in the NORMAL position, select the ALTERNATE static source which uses cabin air for ADU2 airspeed and altitude. Proceed as follows:

CAUTION

When the ALTERNATE static source is used, decrease the altimeter readings by 120 ft.

Open all vents and turn Heater/ECS OFF.

STATIC source switch: Lift guard. Select ALTERNATE.

Continue flight.

Note

The airspeed indication obtained through the alternate static source is slightly higher than the actual value in all speed range.

FLIGHT IN THUNDERSTORM - LIGHTNING STRIKE

When flying near thunderstorm activity, the helicopter may be struck by lightning. If it is suspected that the helicopter has been struck by lightning, proceed as follows:

CAUTION

Avoid performing extreme manoeuvers.

Reduce airspeed to 80 KIAS.

Land as soon as practicable.

If it is suspected that the pilot's Pitot system has been damaged by lightning, proceed as per **STATIC PORT OBSTRUCTION** procedure Page 3-57.

If it is suspected that the helicopter has been struck by lightning this must be noted in the helicopter log-book.

EMERGENCY EXITS

Figure 3-3 show the positions of the helicopter emergency entrances and exits. The emergency release mechanisms are also illustrated.

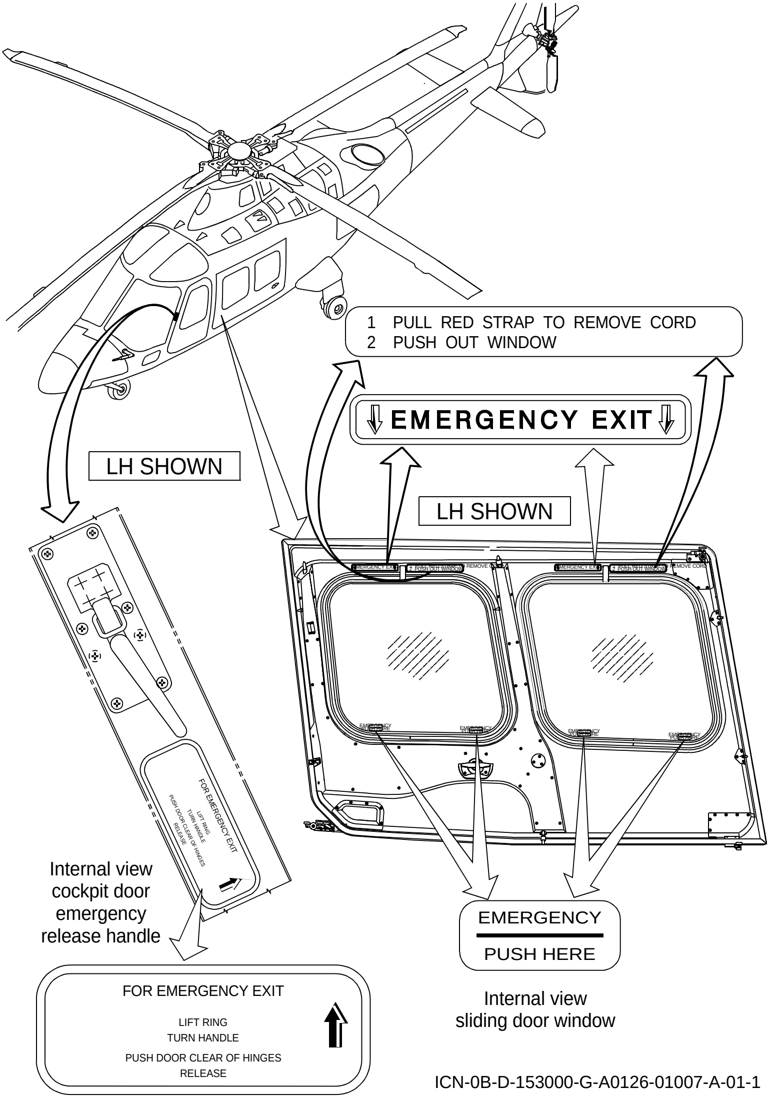


Figure 3-3 Markings and Placards

MALFUNCTION PROCEDURES

IDS CAUTION SYSTEM

Many of the malfunctions described in this section are indicated by the illumination of the yellow caution captions on the IDS CAS window and the flashing of the yellow Master Caution Light (MCL).

Whenever a Caution illuminates, appropriate actions should be taken to deal with the indicated malfunction.

An active Caution message is displayed in yellow text on a black background. The caution can be acknowledged by pressing the MCL or the Master Reset pushbutton on the collective grip. Once acknowledged, the MCL is reset to indicate any future Cautions.

The Caution message remains until the cause of the caution is corrected.

The “last-in” Caution message is displayed on top of the caution list of the CAS display area, and the existing list is displaced down. The Caution messages not displayed on the screen are available line by line using the Arrow rocker switch located on the display unit.

Table 3-2 contains a list of all IDS CAS Caution messages.

Table 3-2 Table of IDS Caution Messages

CAS Caption	Page	Malfunction/System State
ADC 1(2)	Page 3-163	Associated ADU failed.
#1(2) A/F F FLTR	Page 3-114	Associated airframe fuel filter partially blocked. Impeding by-pass.
AHRS 1(2)	Page 3-164	Associated AHRS failed.
AP AHRS 1(2) FAIL	Page 3-137	AFCS not receiving associated AHRS information.
AP-CAS FAIL	Page 3-136	AFCS CAS messages and audio attention getters are unavailable.
AP DEGRAGED	Page 3-138	AFCS not receiving ESIS data
1(2) AP DEGRADED	Page 3-140	Pre Flight Test completed with some tests skipped
1(2) AP MAINT	3-141	Any failure detected on AP channels
1(2) AP FAIL	Page 3-147	Uncommanded disengagement of associated AP channel.
1(2) AP HOT	Page 3-148	Associate FCC temperature is above limit.
APMS PNL FAIL	Page 3-141	Failure of upper mode pushbuttons on APMS panel.
1(2) AP OFF	Page 3-146	Associated AP channel not engaged.
1(2) AP P FAIL	Page 3-149	Failure affecting a single series actuator in the pitch axis on associated AP.
1(2) AP R FAIL	Page 3-149	Failure affecting a single series actuator in the roll axis on associated AP.
1(2) AP TEST FAIL	Page 3-139	Associated AP channel has failed the pre-flight test.

Table 3-2 Table of IDS Caution Messages (Cont.d)

1(2) AP Y FAIL	Page 3-150	Failure affecting a single series actuator in the yaw axis on associated AP.
ATT OFF	Page 3-135	ATT mode unavailable or deselected in pitch or roll.
AWG FAIL	3-131	Aural Warning Generator (AWG) system failure.
BAG DOOR	3-158	Baggage compartment door not correctly closed.
BATT OFF	3-125	Battery off-line.
BUS TIE	3-128	Bus tie open.
CABIN DOOR	3-157	Pilot's, copilot's and/or passengers' door not correctly closed.
COCKPIT FAN	Page 3-160	Cockpit fan failed.
C TRIM FAIL	Page 3-145	AFCS has lost trim actuator drive capability in collective.
DAU MISCMP-P	Page 3-156	Miscompare of DAU primary parameters. Possible degradation in system function.
#1(2) DC GEN	3-127	Associated DC generator failed.
#1(2) DC GEN OVLD	3-129	Associated generator in overload condition.
#1(2) DCU	3-82	Associated Data Collection Unit (DCU) failure.
#1(2) ECU DATA	Page 3-83	Loss of data from associated ECU.
#1(2) ECU MAINT	Page 3-83	Associated engine electronic control unit (ECU) non-critical failure.
EFIS FAN	Page 3-161	EFIS fans failed

Table 3-2 Table of IDS Caution Messages (Cont.d)

EMER UTIL CHRG	Page 3-123	Solenoid valve of emergency utility hydraulic system open. Emergency utility hydraulic system is operative.
EMER UTIL PRES	Page 3-122	Low pressure in emergency utility hydraulic system.
ENG 1(2) IDLE	Page 3-72	Attempted Take Off with one engine in Idle.
EXT PWR ON	Page 3-126	External power connected to the helicopter or external power door open.
FIRE BTL 1(2)	Page 3-89	Associated fire extinguisher bottle discharged.
#1(2) FIRE DET	Page 3-89	Associated engine fire detection system inoperative.
#1(2) F LOW FAIL	Page 3-115	Associated fuel-low sensor test failed. No fuel-low indication for associated fuel tank.
FUEL DRAIN 1(2)	Page 3-115	Associated fuel drain valve open.
#1(2) FUEL FLTR	Page 3-114	Associated fuel filter partially blocked. Impeding by-pass.
#1(2) FUEL LOW	Page 3-110	Fuel quantity in associated tank below 33 kg.
FUEL PUMP 1(2)	Page 3-111	Associated fuel pump failed.
#1(2) GEN CTL	Page 3-124	Associated DC generator control box breaker in relay box tripped.
#1(2) HOT START	Page 3-84	Associated engine TOT exceeded limit on engine starting.

Table 3-2 Table of IDS Caution Messages (Cont.d)

IDS	Page 3-152 & Page 3-154	Failure of Integrated Display System (IDS). Possible degradation in system function.
INV 1(2)	Page 3-129	Associated inverter failed.
LANDING GEAR	Page 3-161	Landing gear retracted below 200 ft .
MAIN RTR SERVO	Page 3-119	A servo valve in one of the three main rotor servo actuators is jammed.
MAIN UTIL CHRG	Page 3-123	Solenoid valve of main utility hydraulic system open. Main utility hydraulic system is operative.
MAIN UTIL PRES	Page 3-121	Low pressure in main utility hydraulic system.
MGT MATCH	Page 3-86	Crosstalk between the 2 ECU failed. No engine TOT matching possible.
MISTRIM	Page 3-143	Series actuators not centered.
#1(2) OEI TIME	Page 3-81	Cumulative time at OEI 2.5 min rating has been reached.
#1(2) OIL CHIP	Page 3-74	Presence of metal particles in associated engine oil circuit.
#1(2) OIL HOT	Page 3-73	Associated engine oil temperature above maximum limit.
#1(2) OVSPD	Page 3-78	Engine drive-shaft failure.
#1(2) OVSPD	Page 3-79	Associated engine N2 overspeed control system triggered.
#1(2) OVSPD DET	Page 3-80	Associated engine N2 overspeed control system failed.

Table 3-2 Table of IDS Caution Messages (Cont.d)

#1(2) OVSPD TEST	Page 3-81	Associated engine N2 overspeed control circuit test failure.
PARK BRK ON	Page 3-159	Parking brake ON.
PITOT 1(2) FAIL	Page 3-160	Associated Pitot heat failed.
#1(2) PLA	Page 3-76	Associated engine power lever (PLA) out of FLIGHT position whilst in AUTO mode.
#1(2) PLA MOTOR	Page 3-75	Associated engine power lever (PLA) remote control inoperative. Beep trim failed.
#1(2) PMS	Page 3-77	Associated ENG MODE switch (PMS) failed.
P TRIM FAIL	Page 3-143	AFCS has lost trim actuator drive capability in pitch.
RMS MESSAGES	Page 3-134	Radio Management System failure.
ROTOR BRK	Page 3-109	Rotor brake system degraded or rotor brake lever not in OFF position.
RPM SELECT	Page 3-88	NR switch inoperative. Inability to trim NR.
R TRIM FAIL	Page 3-144	AFCS has lost trim actuator drive capability in roll.
SERVO 1(2)	Pages 3-116 & 3-118	Associated servo-hydraulic system failed.
TGB OIL CHIP	Page 3-108	Presence of metal particles in the tail rotor gearbox oil.
#1(2) TOT LIMITER	Page 3-85	Associated engine TOT limiter inoperative.
TRQ LIMITER	Page 3-86	Engine torque limiter inoperative.

Table 3-2 Table of IDS Caution Messages (Cont.d)

XMSN OIL CHIP	Page 3-107	Presence of metal particles in the main gearbox oil.
XMSN OVTRQ	Page 3-106	Transmission overtorque: In AEO: TRQ1 + TRQ2 above 214%. In OEI: TRQ1 or TRQ2 above 162%.
Y TRIM FAIL	Page 3-144	AFCS has lost trim actuator capability in yaw.

CAUTIONS WITH A VOICE MESSAGE

Caution Message	Voice Message	System State	Page
XMSN OVTRQ	“OVERTORQUE”	Transmission limit has been exceeded.	Page 3-106
1(2) AP OFF	“AUTOPILOT AUTOPILOT”	Associated AP selected off.	Page 3-146
1(2) AP FAIL	“AUTOPILOT AUTOPILOT”	Uncommanded disengagement of associated AP channel.	Page 3-147
1(2) AP P FAIL	“AUTOPILOT AUTOPILOT”	AP channel is not engaged in pitch on associated AP channel.	Page 3-146
1(2) AP R FAIL	“AUTOPILOT AUTOPILOT”	AP channel is not engaged in roll on associated AP channel.	Page 3-146
1(2) AP Y FAIL	“AUTOPILOT AUTOPILOT”	AP channel is not engaged in yaw on associated AP channel.	Page 3-146
LANDING GEAR	“LANDING GEAR”	Height is less than 200 ft rad-alt and landing gear is retracted.	Page 3-161

VOICE ONLY CAUTIONS

Voice Message	System State	Page
“AIRSPEED”	V_{NE} exceeded.	Page 3-165
“ALTITUDE ALTITUDE”	Excessive deviation of altitude / height with respect to the pre-selected value.	Page 3-165

ENGINE MALFUNCTIONS

COMPRESSOR STALL

A compressor stall is normally recognized by an audible bang or pop accompanied by a possible increase in TOT and fluctuating N1 and TRQ. The compressor stall may be transient or steady.

The degree of compressor stall may be indicated by one or all of the following:

— Fluctuating N1 speed coupled with failure to respond to power demand.

— Loud banging or popping noises (similar to back-firing).

— A reduction in torque (due to reduced air flow through the engine).

— A rapid increase in TOT (due to mis-matching between fuel flow and N1 speed).

If compressor stall occurs, carry out the following procedure:

Achieve Safe OEI Flight

- Select MAIN mode on EDU2.
- Identify and confirm malfunctioning engine.
- Monitor engine parameters for abnormal indications.

IF stall persists, on affected engine:
- Select ENG MODE switch to IDLE.

IF TOT decreases, N1 stabilize and other abnormal indications clear.

ENG MODE switch: FLT.

engine still responds abnormally

YES

- ENG MODE switch: IDLE.
- Monitor engine parameters.

Land as soon as practicable.

NO

Land as soon as practicable.

IF stall does not clear and TOT continues to rise.

Carry out ENGINE SHUTDOWN IN AN EMERGENCY procedure Page 3-20.

UNUSUAL ENGINE NOISE

Compressor damage as a result of FOD may increase the engine noise level and is detectable by a high-pitched whining sound. The noise level of the high pitched whine should vary with N1 and should be significantly higher than the usual engine noise.

If an unusual noise is detected and FOD damage suspected:

Achieve Safe OEI Flight

Monitor engine parameters for abnormal indications.

Select each ENG MODE switch to IDLE then back to FLT sequentially to identify the affected engine.

Use ENGINE SHUTDOWN IN AN EMERGENCY procedure Page 3-20 to shutdown affected engine as soon as practicable to avoid possible secondary compressor damage.

Land as soon as practicable.

ATTEMPTED TAKE-OFF WITH ONE ENGINE IN IDLE

ENG 1(2) IDLE + Audio Tone and voice warning “ENGINE IDLE”

On ground, collective lever raised while Engine #1(2) is still in IDLE.

Do NOT take-off.

Lower collective.

Bring Engine #1(2) to FLIGHT regime.

ENGINE OIL TEMPERATURE

#1(2) OIL HOT

Associated engine oil temperature above the maximum limit.

On affected engine:
- Check oil temperature.

Oil temperature within limits

YES

Continue flight, monitoring oil pressure and temperature.

NO

Achieve Safe OEI Flight

On affected engine:
- Shutdown engine using ENGINE SHUTDOWN IN AN EMERGENCY procedure Page 3-20.

ENGINE OIL CHIP

(Helicopters not equipped with pulsed chip detector system P/N 109-0811-48)

#1(2) OIL CHIP

Presence of metal particles in associated engine oil.

- Check affected engine parameters and engine oil pressure.

Achieve safe OEI flight.

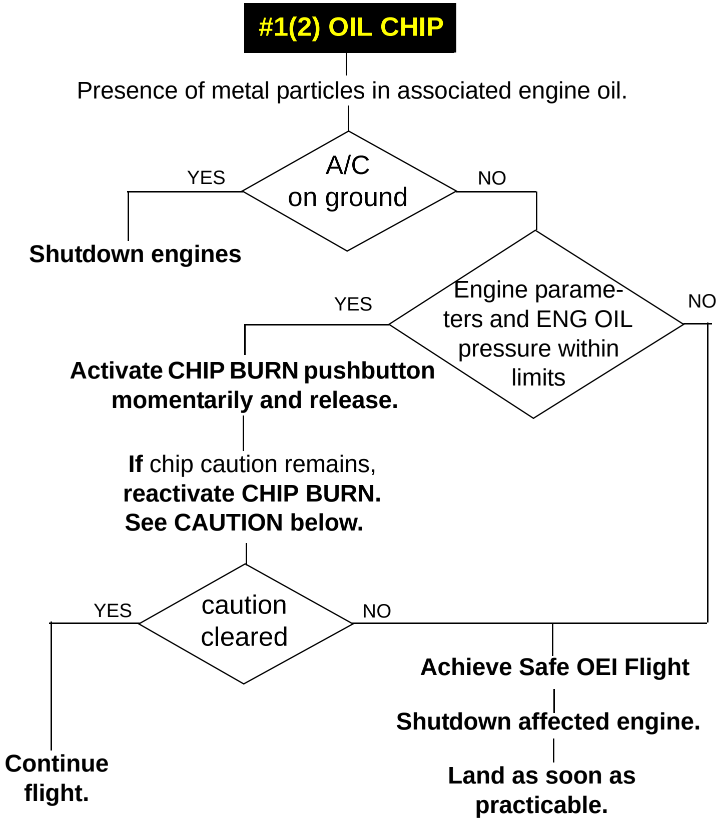
Reduce power by lowering collective.

Shutdown affected engine.

Land as soon as practicable.

ENGINE OIL CHIP

(Helicopters equipped with pulsed chip detector system P/N 109-0811-48)



CAUTION

A maximum of 3 chip burn activations are permitted per flight to clear the caution.

Note

Appropriate log book entry must be made for each message activation.
If chip burn activation has been recorded, perform maintenance action as per applicable maintenance manual before next flight.

ENGINE POWER LEVER REMOTE CONTROL FAIL

#1(2) PLA MOTOR

Associated engine power lever (PLA) remote control beep not functioning.

Continue flight.

If engine manual control is required, this must be carried out by moving the associated engine power lever manually.

ENGINE POWER LEVER POSITION

#1(2) PLA

Associated engine power lever (PLA) out of FLIGHT position detent. (Message active in AUTO mode only).

Check engine power lever position and correct if required.

Even if caution message remains:

Continue flight.

On affected engine:
Monitor engine parameters.

ENGINE MODE SELECT SWITCH FAILURE

#1(2) PMS

Associated ENG MODE switch failure.

Continue flight.

Note

Engine shutdown may not be accomplished through the affected ENG MODE switch. If necessary, use the engine power lever to shutdown the engine. Refer to ENGINE SHUT DOWN IN MANUAL MODE procedure Page 3-99

ENGINE DRIVE SHAFT FAILURE

Rapid decrease in Eng 1(2) TRQ to 0 %
with Eng 1(2) N2 above NR and

#1(2) OVSPD

Drive shaft failure on affected engine.

Achieve Safe OEI Flight

Carry out ENGINE SHUTDOWN IN AN EMERGENCY procedure Page 3-20.

Land as soon as practicable.

Note

Following an engine drive shaft failure N2 of affected engine will oscillate between N2 overspeed trip point (111 %) and 109 %.

ENGINE POWER TURBINE OVERSPEED

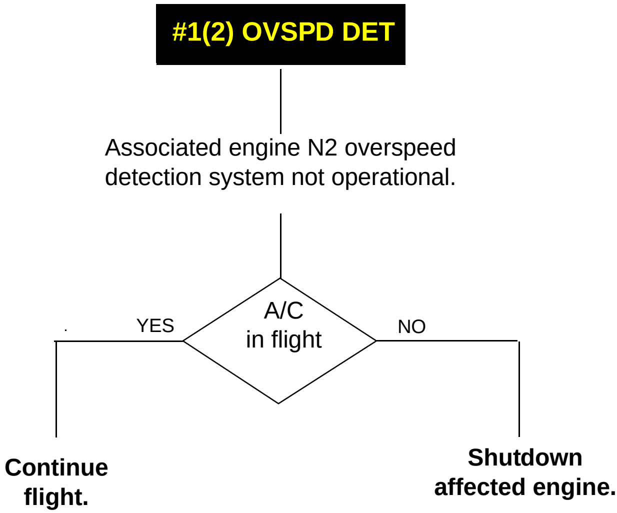
#1(2) OVSPD

Associated engine N2 at or above 111% and engine in overspeed condition.
N2 overspeed control system triggered.
Engine N2 will fluctuate between 111 % and 109 %.

Achieve Safe OEI Flight

Carry out
ENGINE SHUTDOWN IN AN EMERGENCY procedure Page 3-20.

ENGINE POWER TURBINE OVERSPEED DETECT FAILURE



CAUTION

Be aware that N2 overspeed protection system on affected engine will not function in the event of an N2 overspeed.

N2 OVERSPEED CONTROL CIRCUIT TEST FAILURE

#1(2) OVSPD TEST

Correct problem before flight.

OEI 2.5 MIN RATING OUTSIDE LIMITS

#1(2) OEI TIME

Cumulative time at OEI 2.5 min rating has been reached.

Continue flight.

Avoid flight conditions that require OEI 2.5 min power. Refer to Engine MM for maintenance action.

DATA COLLECTION UNIT MALFUNCTION

#1(2) DCU

Correct problem before flight.

Note

#1(2) DCU caution messages active only on ground with ENG MODE switch in OFF position.

ENGINE CONTROL UNIT - MAINTENANCE REQUIRED

#1(2) ECU MAINT

Correct problem before flight.

Note

#1(2) ECU MAINT caution messages active only on ground with ENG MODE switch in OFF position.

ENGINE CONTROL UNIT - LOSS OF DATA

#1(2) ECU DATA

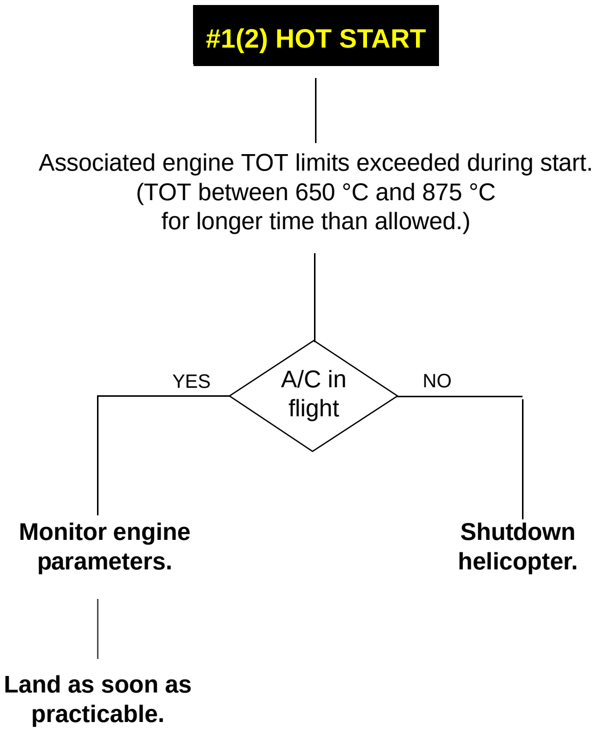
Loss of data from associated engine ECU.

On affected engine:

- Check engine parameters displayed on EDU1 for correctness of backup indications.

Continue flight.

ENGINE HOT START



TOT LIMITER MALFUNCTION

#1(2) TOT LIMITER

Associated engine TOT thermocouple failed.
TOT limiter function of Engine Control Unit inoperative.
Possible loss of TOT information.
Engine TOT matching inoperative.

Set LD-SH switch to TORQUE.

TOT information available

YES

Do not exceed TOT limits.

Continue flight.

NO

- **Select MAIN mode on EDU2.**
- **Monitor remaining engine parameters.**
- **Avoid operating near limits.**
- **Avoid exceeding 94.5% N1 and 10000 ft pressure altitude.**
- **Avoid, as far as possible, to bleed air from engine.**

Continue flight.

TORQUE LIMITER MALFUNCTION

TRQ LIMITER

Torque limiter function of Engine Control Unit inoperative and no communication between Engine Control Units. Engine matching inoperative.

Avoid operating near torque limits.

Continue flight.

CAUTION

The AFCS Upper Modes and Flight Director modes must not be used unless conducting an approach, missed approach, transition to/from the hover or hover.

MGT MISMATCH

MGT MATCH

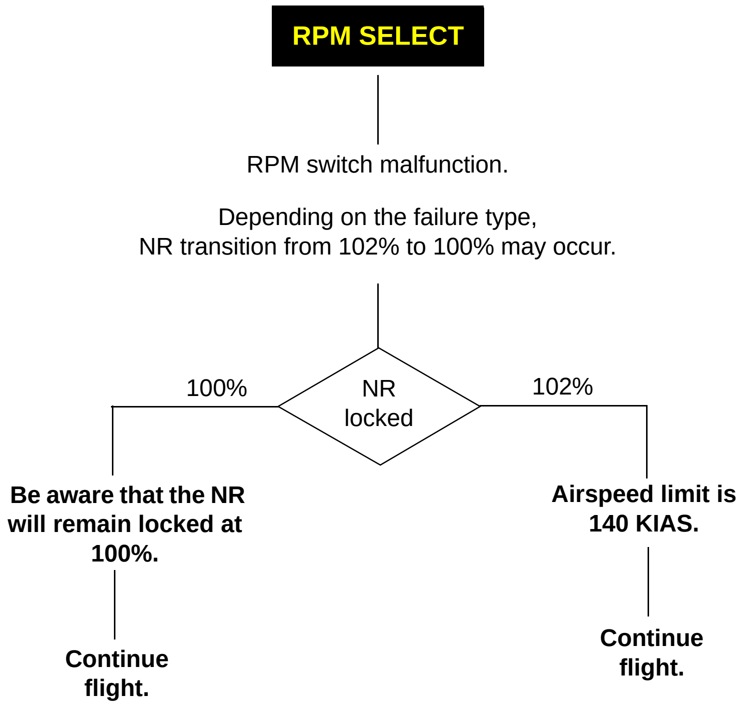
Crosstalk between the 2 ECU failed.
No engine TOT matching possible.

Set LD-SH switch to TORQUE.

Avoid operating near engine limits.

Continue flight.

ROTOR SPEED SELECTOR MALFUNCTION



ENGINE FIRE DETECTOR SYSTEM FAILURE

#1(2) FIRE DET

Associated engine fire detection system inoperative.

On affected engine:
Monitor engine parameters for abnormalities and check for signs of fire.

Land as soon as practicable.

FIRE EXTINGUISHER BOTTLE DISCHARGED

FIRE BTL 1(2)

Associated fire extinguisher bottle discharged.

Refill bottle before next flight.

ENGINE RESTART IN FLIGHT PROCEDURE

General

Refer to **ENGINE START PROCEDURE** for the indications associated with malfunctions during engine starting and to **ABORTED ENGINE START PROCEDURE** Page 3-94 for the recommended abort procedure.

CAUTION

If an engine is shutdown or a flame-out occurs during flight, and if there is no indication of a mechanical malfunction or engine fire, the engine may be restarted. Do NOT attempt to restart the engine if the cause of the flame-out has not been ascertained.

In flight restart procedure in auto mode

The following is the procedure for restarting an engine in flight with the ECU operative (in AUTO mode).

CAUTION

For engine restarts below 15000 ft, the ENG MODE switch should only be moved from the OFF position, once N1 is below 20%. Above 15000 ft, the ENG MODE switch should only be moved from the OFF position, once N1 is below 5%.

1. Recommended airspeed — Vy.
2. Engine power lever — FLIGHT.
3. FUEL VALVE switch — OPEN (Bar vertical).

4. FUEL PUMP switch — ON.

Note

It is recommended to start the engine to IDLE, nevertheless, if necessary, it is possible to start to FLIGHT by setting the ENG MODE switch directly to FLT.

5. ENG MODE switch — IDLE.
6. N1 — Note increasing and START legend vertically displayed.
7. TOT — Note increasing and IGN legend vertically displayed.
8. Engine oil pressure — When N1 is greater than 40%, check rising.
9. N2 — Confirm stabilised below 100%.

Note

If the engine was started to FLT, the N2 will stabilize with NR.

10. Engine oil — Confirm temperature and pressure within limits.
11. ENG MODE switch — Set to FLT.
12. N2/NR — Confirm stabilised and matched with NR.
13. XFEED switch — AUTO (bar vertical)

In flight restart procedure in manual mode

The following is the procedure for restarting an engine in flight with the ECU inoperative (in MANUAL mode).

CAUTION

For engine restarts below 15000 ft, the engine power lever must only be engaged once N1 is below 20%.
Above 15000 ft, the engine power lever must only be engaged, once N1 is below 5%.

1. Recommended airspeed — Vy.

2. Engine power lever — OFF

3. ENG GOV switch — MANUAL (MAN legend displayed on EDU1).

Note

In the presence of an ECU failure, the engine control system reverts to MANUAL mode regardless of the ENG GOV switch position. However, it is recommended to set the ENG GOV switch to MANUAL for congruence with the mode condition.

4. FUEL VALVE switch — OPEN (Bar vertical).

5. FUEL PUMP switch — ON.

6. ENG MODE switch — IDLE.

7. Engine power lever — IDLE.

8. Starting button — Push and hold. START and IGN legends vertically displayed on N1 and TOT scales on EDU1.

9. N1 — Note increasing.

10. Engine power lever — Move forward to obtain light-up.

Note

At low altitude (below 3000 ft), light-up is expected to occur when the engine power lever is nearly in FLIGHT position.
At high altitude (above 15000 ft), light-up may occur as soon as the engine power lever is out of the IDLE position.

11. TOT — Note increasing at light-up. Monitor TOT value and control with slight movements of the engine power lever to ensure TOT transient value is not exceeded.

Note

If engine hangs at N1 below 54%, slowly move engine power lever forward, if necessary even beyond FLIGHT position, until the engine accelerates. Monitor TOT, N1 and NR. If engine does not accelerate, shutdown engine by setting engine power lever to OFF position and release starting button.

12. Starting button — Release when N1 reaches 50%. START and IGN legends suppressed.

13. Engine power lever — Set the power as required.

14. Engine oil pressure — When N1 is greater than 40% Check rising.

ABORTED ENGINE START PROCEDURE

CAUTION

Failure to follow the abort procedure may cause damage to the engine.

Note

Observe the igniter and starter generator duty cycle limitations, as per Section 1, LIMITATIONS.

Monitor engine start and if any of the following occur:

- lightup is not obtained within 15 seconds;
- abnormal noises are heard;
- TOT increases beyond start limits (#1(2) HOT START caution message displayed);
- (on ground) rotor has not begun to rotate when N1 is 40%;
- N1 or N2 increase beyond start limits;
- engine hangs (stagnation in N1 below 54%);

shutdown engine by:

1) ENG MODE switch : OFF.

   If engine does not begin to shutdown:

   Engine power lever : OFF.

2) FUEL PUMP : OFF.

3) FUEL VALVE switch : CLOSED (bar horizontal).

AUTO TO MANUAL MODE TRANSFER

Transfer from AUTO to MANUAL mode can occur as a result of:

1. Setting ENG GOV switch on collective grip to MANUAL.

2. Automatic reversion to MANUAL due to ECU critical hardware failure.

Note

In the presence of an ECU failure, #1(2) ECU FAIL warning message will be displayed and audio tone and voice warning “WARNING” audible.

Note

In presence of an ECU failure, the engine control system reverts to MANUAL mode regardless of the ENG GOV switch position. However, it is recommended to set the ENG GOV switch to MANUAL for congruence with the mode condition.

Note

#1(2) PLA caution message is active only when operating in AUTO mode.

The MANUAL mode condition is indicated by the activation of the MAN legend displayed vertically on the TRQ gauge.

ENGINE START IN MANUAL MODE (ON GROUND)

Following an ECU failure, flight may be initiated only in order to fly (VFR) without passengers on board to a repair facility.

Note

An engine should be started in MANUAL mode only if a start in AUTO mode is not possible.

Before proceeding to start the engine in manual mode, perform an ECU power OFF-ON reset in an attempt to clear all faults.

If critical faults are not cleared (ECU FAIL warning and MAN legend still displayed), proceed as follows:

1. Perform ENGINE PRE-START CHECKS as per Section 2.
2. EDU 1 — Confirm START page selected.
3. ENG GOV switch — MANUAL

Note

In the presence of an ECU failure, the engine control system reverts to MANUAL mode regardless of the ENG GOV switch position. However, it is recommended to set the ENG GOV switch to MANUAL for congruence with the mode condition.

4. GEN 1 and 2 switches — ON.
5. FUEL VALVE switch — OPEN (Bar vertical).
6. FUEL PUMP switch — ON.
7. XFEED VALVE switch — Confirm in AUTO position. Confirm bar horizontal.
8. ENG MODE switch — IDLE.
9. Engine power lever — IDLE.

CAUTION

Monitor engine start and if any of the following occur:

- light-up is not obtained within 15 seconds;
- abnormal noises are heard;
- TOT increases beyond start limits (#1(2) HOT START caution message displayed);
- rotor has not begun to rotate when N1 is 40%;
- N1 or N2 increase beyond start limits;
- engine hangs (stagnation in N1 below 54%);

Abort the engine start and shutdown the engine by setting the appropriate power lever to OFF and release the starting button.

Note

If engine hangs at N1 below 54%, slowly move engine power lever forward, if necessary even beyond FLIGHT position, until the engine accelerates. Monitor TOT, N1 and NR. If engine does not accelerate, shutdown engine by setting engine power lever to OFF position and release starting button.

10. Starting button — Push and hold.
START and IGN legends vertically displayed on N1 and TOT scales on EDU1.

11. N1 — Note increasing.

12. Engine power lever — Move forward to obtain light-up.

13. TOT — Note increasing at light-up. Monitor TOT value and control with slight movements of the engine power lever to ensure TOT transient value is not exceeded.

14. Starting button — Release when N1 reaches 50%. START and IGN legends suppressed.

15. Engine power lever — Set the power as required.

CAUTION

The operation at an engine oil pressure up to 200 psi is permitted for a maximum period of 10 minutes.

Note

During cold starting conditions, the engine oil pressure can rise up to 200 psi and will decrease as the oil temperature rises.

16. Engine oil pressure — When N1 is greater than 40% Check rising.

Continue as per NORMAL ENGINE START procedure, Section 2.

ENGINE SHUTDOWN IN MANUAL MODE

This procedure applies in case of ECU failure and consequent reversion to MANUAL mode.

Note

#1(2) ECU FAIL warning message and MAN legend displayed vertically on the TRQ gauge.

ENG GOV switch : MANUAL.

Note

In the presence of an ECU failure, the engine control system reverts and operates in MANUAL mode regardless of the ENG GOV switch position. However, it is recommended to set the ENG GOV switch to MANUAL for congruence with the mode condition.

Engine power lever : Set to IDLE to allow N1 to reduce to approximately 60%.

: At 60% N1, set to OFF.

ENG MODE switch : OFF.

FUEL PUMP switch : OFF. FUEL PUMP 1(2) caution message displayed.

CAUTION

During shutdown, check that N1 decelerates freely. Note any abnormal noise or rapid rundown.

CAUTION

If there is evidence of any abnormal TOT increase after shutdown, perform a **DRY MOTORING RUN**, as per Section 2.

ENGINE OPERATION IN MANUAL MODE

Following a reversion to MANUAL mode (MAN legend displayed vertically on TRQ gauge), the electric stepper motor of the corresponding engine is frozen at its current position, maintaining the engine torque at its current value just prior to MANUAL reversion.

In case of failure during transient condition, the pilot should control engine ensuring that parameters stay within operating range.

The MANUAL mode control characteristics will vary depending on the engine condition at the time of the reversion to MANUAL mode.

One engine in MANUAL mode and one in AUTO mode

It is recommended to set the engine which is in MANUAL mode to a suitable fixed power related to the particular flight condition, and let the ECU of the other engine (in AUTO mode) maintain NR.

Monitor engine parameters and maintain within operating limits.

Any change of fuel flow and consequently of N1 may be accomplished through engine power lever movements.

Each engine power lever is motorized and should be operated through the relative ENG TRIM toggle switch on collective grip.

Land as soon as practicable.

Both engines in MANUAL mode

In case of critical malfunction of both ECU’s (double failure), the pilot shall control both engines in MANUAL mode.

Each electric stepper motor is frozen at its current position and engine power.

Land as soon as practicable.

Perform a running landing, if terrain permits.

Cruise flight to reach the landing site

CAUTION

When adjusting collective, monitor N2 to avoid exceeding limits.

ENG TRIM toggle switches : Operate to adjust torque of one engine to about 50% of the total torque required for the desired airspeed.

Note

To simplify manual control of the engines, only one engine should be adjusted by operating only one toggle switch.
Furthermore, avoid adjusting collective and engine at the same time.
Transient NR between 95% and 102% is acceptable during adjustments.

Note

Avoid torque settings close to engine operating limits.

Airspeed : It is recommended to respect OEI Vne limits.

Manoeuvres : Avoid any manoeuvre requiring large and rapid changes of torque.

Approach and landing

Landing gear lever	: DOWN.
Utility hydraulic system	: Check pressure within limits.
Nose wheel lock	: ON. (lever up).
Parking brake	: OFF.
External lights	: As required.
Approach	: If GW is 3000 kg or more: - Set one engine torque at 40%. If GW is below 3000 kg: - Set one engine torque at 30%. Use other engine to establish 35 ± 10 KIAS and 400 ft/min max rate of descent at 102% NR. Aim for landing area by changing the airspeed and minimize any further power adjustment.

CAUTION

When lowering collective, monitor N2 to avoid exceeding limits.

Landing	: Gently flare helicopter to minimize ground speed at landing. Use collective to cushion touchdown and, if necessary, allow NR to drop down below 95%. After touchdown, simultaneously lower collective and reduce both engines at the same time as necessary to maintain NR within limits.
Apply toe brakes.	
Shutdown	: Refer to paragraph **ENGINE SHUTDOWN IN MANUAL MODE Page 3-99**.

ENGINE POWER LEVER (PLA) JAMMED PROCEDURE

1. The engine power lever does not respond (i.e. no movement of the lever).

2. #1(2) PLA MOTOR caution message NOT displayed.

IF, when using ENG TRIM toggle, no response of the relevant engine power lever **AND** #1(2) PLA MOTOR caution message NOT displayed.

Achieve Safe OEI Flight

- Pull aft (towards IDLE position) the affected engine power lever.
- Pull strongly enough to break the internal pin holding the engine power lever stepper motor.

Once the stepper motor pin is broken, the engine power lever is free to be manually moved.

Set engine power lever as required.

DRIVE SYSTEM

MAIN TRANSMISSION OVERTORQUE

XMSN OVTRQ + Audio Tone and Voice Message "OVERTORQUE"

Transmission torque limit exceeded.

Lower collective to maintain torque within limits.

MAIN TRANSMISSION CHIP

(Helicopters not equipped with pulsed chip detector system P/N 109-0811-48)

XMSN OIL CHIP

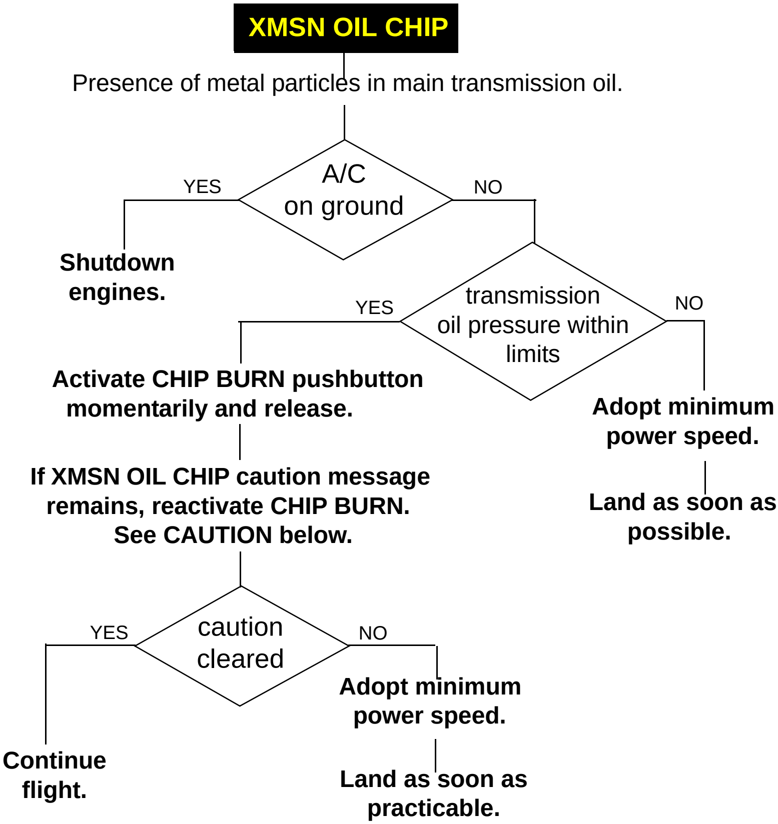
Presence of metal particles in main transmission oil.

Reduce power by lowering collective.

Monitor XMSN oil pressure and temperature.

Land as soon as practicable.

MAIN TRANSMISSION CHIP
(Helicopters equipped with pulsed chip detector system P/N 109-0811-48)



CAUTION

A maximum of 3 chip burn activations are permitted per flight to clear the caution.

Note

Appropriate log book entry must be made for each message activation.
If chip burn activation has been recorded, perform maintenance action as per applicable maintenance manual before next flight.

TAIL GEARBOX CHIP

(Helicopters not equipped with pulsed chip detector system P/N 109-0811-48)

TGB OIL CHIP

Presence of metal particles in tail rotor gearbox oil.

Reduce power by lowering collective.

Land as soon as practicable.

TAIL GEARBOX CHIP
(Helicopters equipped with pulsed chip detector system P/N 109-0811-48)

TGB OIL CHIP

Presence of metal particles in tail rotor gearbox oil.

A/C on ground

YES

Shutdown engines.

NO

Activate CHIP BURN pushbutton momentarily and release.

If TGB OIL CHIP caution message remains, **reactivate CHIP BURN. See CAUTION below.**

caution cleared

YES

Continue flight.

NO

Adopt minimum power speed.

Land as soon as practicable.

CAUTION

A maximum of 3 chip burn activations are permitted per flight to clear the caution.

Note

Appropriate log book entry must be made for each message activation.
If chip burn activation has been recorded, perform maintenance action as per applicable maintenance manual before next flight.

ROTOR BRAKE SYSTEM DEGRADED

ROTOR BRK

Rotor brake system degraded or
Rotor brake lever not in OFF position.

Confirm Rotor Brake lever in OFF position

caution cleared

YES: **Continue flight.**

NO: **Continue flight.**
Do not use rotor brake at shutdown

FUEL SYSTEM

FUEL LOW

#1(2) FUEL LOW

On affected tank, usable fuel quantity below 33 kg.

WARNING

Hovering in crosswinds or sideways flight with sustained roll angles greater than 10 degrees, when FUEL LOW caution message is illuminated, could lead to a single or double engine flame-out.

- **Fuel contents:** **Check.**
- **XFEED valve:** **Confirm AUTO (valve status Closed).** (see Note)

Land as soon as practicable.
(within time limits stated in the Note below).

Note

When each engine is supplied fuel from its respective tank, the remaining flight duration is approximately 15 minutes from caution message activation. In the unusual event that both engines are supplied fuel from the same tank, the remaining flight duration is approximately 6 minutes from caution message activation.

FUEL PRESSURE LOW

FUEL PUMP 1(2)

Associated fuel pump failed or
Low fuel pressure in associated fuel line.

- Switch OFF associated pump.
- Confirm crossfeed valve opens automatically (bar horizontal).
(XFEED advisory message displayed).

CAUTION

With one fuel pump failed (or OFF) and the crossfeed valve open, the fuel in the affected tank (up to 100 kg) will become unusable.

Fuel pressure within limits

YES

Continue flight.

If it is necessary to use the 100 kg of unusable fuel in the affected tank
AND
#1(2) A/F F FLTR caution message is NOT displayed

When the indicated fuel quantity in either tank is below 100 kg:
- Set XFEED VALVE:CLOSED.
(Procedure not applicable when Aviation Gasoline is used

NO

CAUTION

Possible fuel leak, be attentive for signs of fuel leak or loss of engine power.
- Set XFEED VALVE:CLOSED.
(See Note)

Land as soon as practicable.

WARNING

With the crossfeed valve closed, avoid banked turns above 30 deg and sustained pitch angle below 0 deg.

Continued Next Page

Continuation from previous page

Note

When a fuel pump has failed (FUEL PUMP 1(2) caution message displayed), #1(2) FUEL LOW caution message will be activated when fuel in affected tank is between 60 and 70 kg. The unusable fuel will be up to 16 kg.

FUEL PUMP FAILED AND AIRFRAME FUEL FILTER CLOGGED

FUEL PUMP 1(2) + **#1(2) A/F F FLTR**

Associated fuel pump failed
AND
Associated airframe fuel filter partially clogged.

Switch OFF associated pump.

Confirm cross feed valve opens automatically (bar horizontal) and increase of fuel pressure of the affected system. (XFEED advisory message displayed).

Continue flight monitoring fuel pressure considering the new range and endurance.

CAUTION

With one fuel pump failed and the cross feed valve open, a maximum of 100 kg fuel indicated will become unusable in the tank with the failed fuel pump.

AIRFRAME FUEL FILTER CLOGGED

#1(2) A/F F FLTR

Associated airframe fuel filter partially clogged.

Continue flight monitoring fuel pressure.

ENGINE FUEL FILTER CLOGGED

#1(2) FUEL FLTR

Associated engine fuel filter partially clogged.

Continue flight monitoring fuel pressure.

FUEL LOW SENSOR FAILURE

#1(2) F LOW FAIL

Associated fuel low sensor failure.
FUEL LOW caution message not available
or inconsistent with fuel level.

On affected system:
- Monitor fuel quantity.

Continue flight.

WARNING

Hovering in crosswinds or sideways flight with sustained roll angles greater than 10 degrees, when fuel quantity is below 33 Kg, could lead to a single or double engine flame-out.

FUEL DRAIN VALVE OPEN

FUEL DRAIN 1(2)

Associated fuel drain valve open.

Close relevant drain valve, before refuelling and/or take-off.

Continue flight.

Note

The FUEL DRAIN 1(2) caution messages can only be activated when the helicopter is on the ground.

HYDRAULIC SYSTEM

SERVO HYDRAULIC SYSTEM N°1 MALFUNCTION

SERVO 1

Also **1 AP Y FAIL** + **2 AP Y FAIL**

N°1 Servo hydraulic system failed.

Check N°1 servo hydraulic system pressure on EDU2.

pressure above 1200 psi

YES

Continue flight monitoring system pressure.

NO

- **Gradually reduce airspeed below 90 KIAS.**
- **Avoid pull-up manoeuvres and bank angles above 25 deg to maintain acceptable loads.**
- **Avoid rapid movements of flight controls.**

SERVO switch: SOV 1 OFF in order to set the N°1 Servo hydraulic system off.

Land as soon as practicable.

WARNING

Following the pressure loss of N°1 main servo hydraulic system, avoid landing and/or operating in conditions which require a high degree of control activity such as confined areas or out-of-wind hovering, particularly with wind from right.

Continued Next Page

Continuation from previous page

Note

Pedals will tend to move to zero thrust position (Right pedal slightly more forward than the left pedal).
Control force on pedals will increase in hover and in forward flight.

SERVO HYDRAULIC SYSTEM N°2 MALFUNCTION

SERVO 2

N°2 Servo hydraulic system failed.

Check N°2 servo hydraulic system pressure on EDU2.

pressure above 1200 psi

YES

Continue flight monitoring system pressure.

NO

- **Gradually reduce airspeed below 90 KIAS.**
- **Avoid pull-up manoeuvres and bank angles above 25 deg to maintain acceptable loads.**
- **Avoid rapid movements of flight controls.**

Control forces remain normal.

SERVO switch: SOV 2 OFF in order to set the N°2 Servo hydraulic system off.

Land as soon as practicable.

Note

In the event of pressure loss in N°2 main hydraulic system, sufficient pressure remains in the accumulator of the normal utility hydraulic system for one extension of the landing gear and to operate the toe brakes and the rotor brake.

JAMMING OF A MAIN ROTOR SERVO VALVE

The helicopter is equipped with three hydraulic servo actuators, "tandem" type, on main rotor controls (cyclic and collective).

MAIN RTR SERVO

A servo valve in one of the 3 main rotor servo actuators is jammed.

- **Gradually reduce airspeed below 90 KIAS.**
- **Avoid pull-up manoeuvres.**
- **Avoid bank angle above 25 deg.**
- **Avoid rapid movement of the cyclic and collective controls.**

Land as soon as practicable.

When on the ground, conduct HYDRAULIC SYSTEMS CHECK as described in Section 2.

If servo valve jamming is confirmed, do NOT resume flight.

TAIL ROTOR SERVO VALVE JAMMING

Jamming of the servo valve of the tail rotor servo actuator will result in an increase of pedal control force in hovering and in forward flight.

Also **1 AP Y FAIL** + **2 AP Y FAIL**

Procedure:

- **Gradually reduce airspeed below 90 KIAS.**
- **Avoid pull-up manoeuvres.**
- **Avoid bank angle above 25 deg.**
- **Avoid rapid movement of the cyclic and collective controls.**

Land as soon as practicable.

WARNING

Following the loss of tail rotor servo actuator avoid landing and/or operating in conditions which require a high degree of manoeuverability (i.e. avoid operating in enclosed areas, avoid operation with sideward winds, in particular with wind from the right.)

NORMAL UTILITY HYDRAULIC PRESSURE LOW

MAIN UTIL PRES

Failure of the normal utility hydraulic system. Pressure in the system is below minimum pressure of 500 psi.

- Select AUXILIARY mode on EDU2.
-- Confirm low pressure on normal utility system.

IF normal utility pressure below 500 psi:

- Reduce speed to 140 KIAS maximum

Perform EMERGENCY DOWN LANDING GEAR procedure Page 3-56 independently of landing gear position.

WARNING

In the emergency accumulator, there is sufficient pressure for one extension of the landing gear, after which there is sufficient pressure to operate the emergency brakes. The parking brake handle is used for the emergency brake system and does not provide differential braking. The toe brakes and the rotor brake are inoperative.

EMERGENCY UTILITY HYDRAULIC PRESSURE LOW

EMER UTIL PRES

Failure of the emergency utility hydraulic system. Pressure in the system is below minimum pressure of 1140 psi.

- Select AUXILIARY mode on EDU2.
- Check emergency utility system pressure.

IF emergency utility pressure below 1140 psi:

- Maintain AUXILIARY mode on EDU2.
- Continue flight, carefully monitoring UTIL NORM pressure.

IF normal utility pressure below 1140 psi:

Reduce speed to 140 KIAS maximum.

Extend landing gear using the normal procedure.

Continue flight.

NORMAL UTILITY HYDRAULIC SYSTEM CHARGING

MAIN UTIL CHRG

Normal utility hydraulic system solenoid valve open.
The normal utility hydraulic system is operative.

Continue flight.

Note

If the solenoid valve between the normal utility hydraulic system and the N°2 servo hydraulic system is locked open, an external oil leakage in any of the 2 systems will automatically induce failure of the other system.

EMERGENCY UTILITY HYDRAULIC SYSTEM CHARGING

EMER UTIL CHRG

Emergency utility hydraulic system solenoid valve open.
The emergency utility hydraulic system is operative.

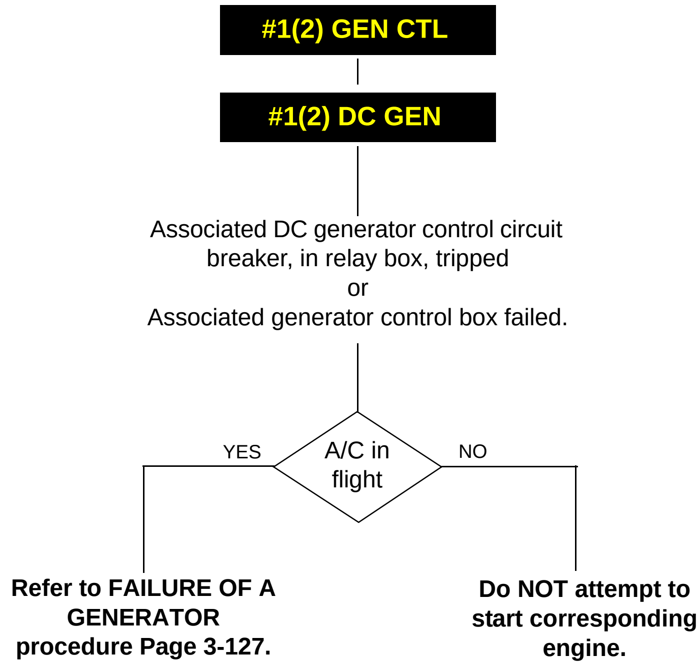
Do NOT take-off.

Note

EMER UTIL CHRG is only triggered on ground.

ELECTRICAL SYSTEM

DC GENERATOR CONTROL



BATTERY OFF

BATT OFF

Battery disconnected.

Check BAT switch ON.

caution clears

YES: **Continue flight.**

NO: **Land as soon as practicable.**

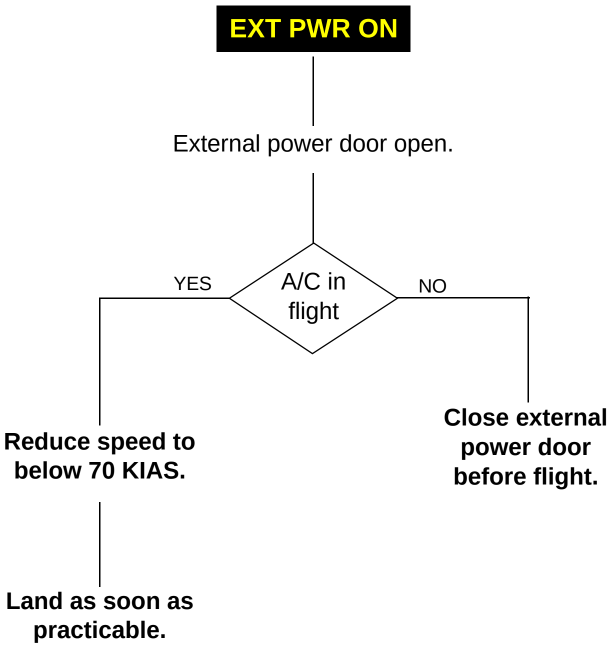
CAUTION

With the battery OFF, do NOT attempt an engine restart following an engine flame-out.

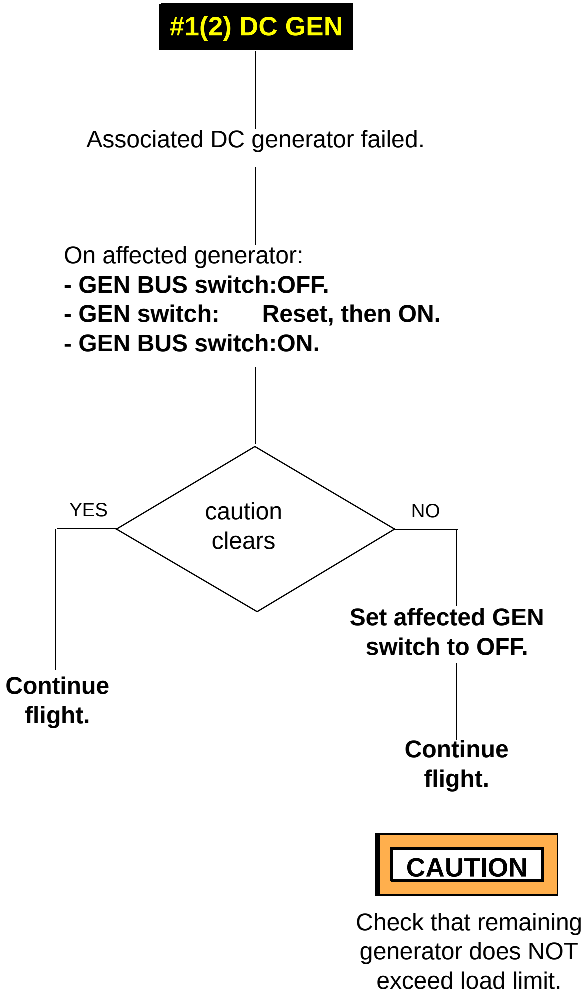
Note

When external power is connected, the battery is automatically disconnected and the BATT OFF caution message is displayed.

EXTERNAL POWER ON



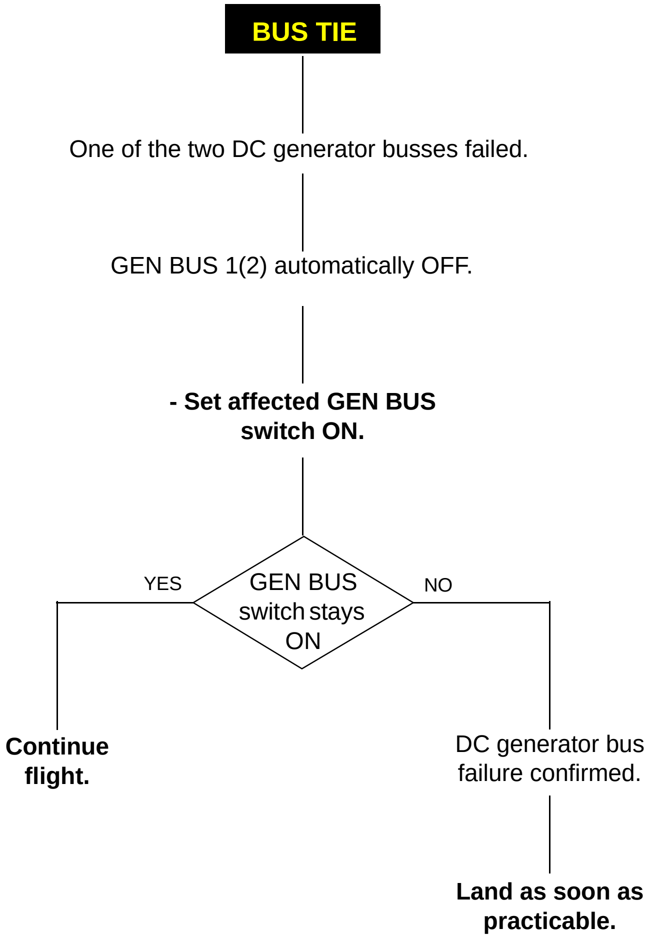
FAILURE OF A GENERATOR



Note

Power is supplied to all loads by remaining generator.

BUS TIE OPEN



CAUTION

With a Bus Tie open, do NOT attempt an engine restart following an engine flame-out.

GENERATOR OVERLOAD

1(2) DC GEN OVLD

Associated generator in overload condition

Reduce electrical load to within green range

Continue flight.

INVERTER FAILURE

INV 1(2)

Associated inverter failed.

Also:

2(1) AP OFF + Audio Tone and Voice Message "AUTOPILOT AUTOPILOT"

Confirm affected inverter circuit breaker not tripped.

Try to reset affected inverter.

Re-engage affected AP channel on APMS panel.

Continue flight.

COMMUNICATION SYSTEM

AURAL WARNING SYSTEM FAILURE

AWG FAIL

Aural warning generator system failure.
Loss of aural warnings and audio tones.

Continue flight monitoring CAS system as aural warnings and audio tones do not function.

PILOT’S ICS FAILURE

Pilot’s ICS (Inter-Communication System) failed

Pilot’s ICS: BK-UP.

If ICS problem persists

Pilot’s ICS: EMER.

Note

When ICS pilot stations are in BK-UP, they are automatically disconnected from the cabin ICS stations.

Note

When ICS pilot stations are in EMER, they are automatically disconnected from the cabin ICS stations and the HOT MIKE function is not available.

MASTER AVIONICS SWITCH FAILURE

Symptoms:
MSTR AVNX set to GND/NORM and equipment connected to these positions not powered.
or
MSTR AVNX set to GND COM and equipment connected to GND position powered.

Master Avionics (MSTR AVNX) switch failure.

MSTR AVNX set to GND / NORM

YES

Correct problem before flight.

NO

MSTR AVNX CONTR circuit breakers (2): Confirm IN. (in the baggage compartment)

problem persists

YES

- Do NOT take-off.
- Shutdown engines.

Correct problem before flight.

NO

Continue flight.

RADIO MANAGEMENT SYSTEM FAILURE

RMS MESSAGES

Radio Management System (RMS) failure.

Identify failed element of the RMS from the FAIL list on RTU's dedicated page.

Continue flight with associated limits of reduced communication and radio navigation capability.

RADIO TUNING UNIT FAILURE

RTU1 (2) failure (display becomes blank).

Confirm on active RTU the loss of some active frequencies (- - -).

Switch off the failed RTU and check that after 5 seconds all systems come back operative on the active RTU.

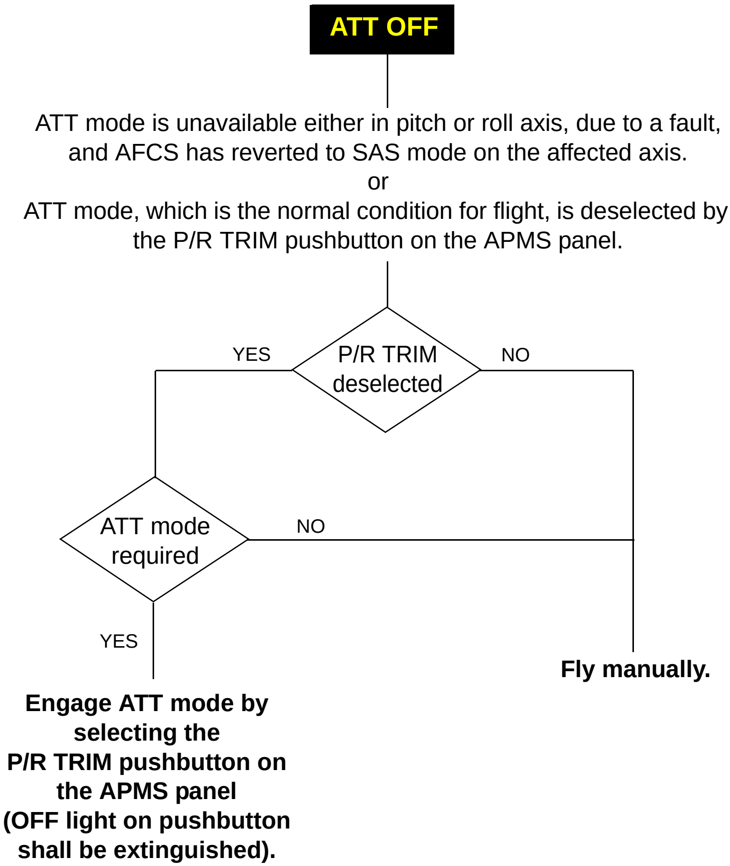
Continue flight.
Repair before next flight.

Note

In case of loss of both RTUs, operate the EMERG FREQ pushbutton to tune the 121.5 MHz emergency frequency.

AUTOMATIC FLIGHT CONTROL SYSTEM

ATTITUDE MODE OFF



AUTOPILOT CAS FAIL

AP-CAS FAIL

AFCS CAS messages and audio attention getters are unavailable.

Continue flight, monitoring the flight instruments for indications of AFCS faults and be aware that also “ALTITUDE ALTITUDE” attention getter is inoperative.

AUTOPILOT AHRS FAIL

AP AHRS 1(2) FAIL

The AFCS is not receiving information from the associated AHRS.
The associated AP channel disengages and cannot be re-engaged.

- Maximum speed is 128 KIAS.
Above 500 ft AGL:
- Fly attentive.
At or below 500 ft AGL:
- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

AP DEGRADED

AP DEGRADED

Loss of ESIS data.
Be aware that both AP channels disengage in case of AHRS failure

- Maximum speed is 128 KIAS.
Above 500 ft AGL:
- Fly attentive.
At or below 500 ft AGL:
- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

AUTOPILOT TEST FAIL

1(2) AP TEST FAIL

Associated AP channel has failed the pre-flight test.

Only one caution message displayed

YES

Ferry flight only.

Select unaffected AP channel to ON.

- Maximum speed is 128 KIAS.
- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

NO

Both

1 AP TEST FAIL

2 AP TEST FAIL

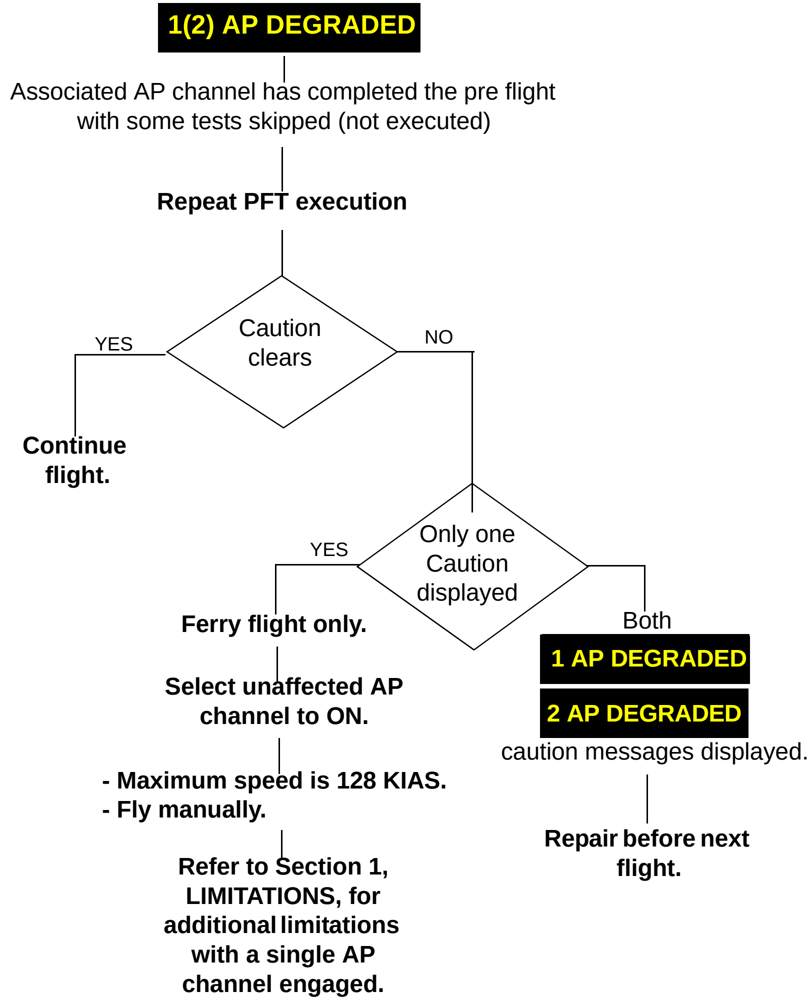
caution messages displayed.

Repair before next flight.

Note

VFR flight with one autopilot inoperative may be initiated only to fly manually to a repair facility.

AFCS PFT PARTIALLY COMPLETED



Note

VFR flight with one autopilot inoperative may be initiated only to fly manually to a repair facility.

AP CHANNEL FAILURE

1(2) AP MAINT

Caution displayed on ground only.
Any failure on associated AP channel

Only one caution message displayed

YES

Ferry flight only.

Select unaffected AP channel to ON.

- Maximum speed is 128 KIAS.
- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

NO

Both

1 AP MAINT

2 AP MAINT

caution messages displayed.

Repair before next flight.

Note

VFR flight with one autopilot inoperative may be initiated only to fly manually to a repair facility.

APMS PANEL FAIL

APMS PNL FAIL

Failure of upper mode and FD mode pushbuttons on APMS panel.

Upper modes and FD modes unavailable.

Continue flight.

MISTRIM

MISTRIM

Series actuators not centered.

- **Manual centering of series actuators is required.**
- **Manual centering is achieved by operating the cyclic FTR and pedal switches, when appropriate.**
- **Continue flight being attentive to AFCS functioning.**

PITCH TRIM FAIL

P TRIM FAIL

AFCS has lost trim actuator drive capability in pitch.

Continue flight being aware that the trim function is unavailable in pitch.

ROLL TRIM FAIL

R TRIM FAIL

AFCS has lost trim actuator drive capability in roll.

Continue flight being aware that the trim function is unavailable in roll.

YAW TRIM FAIL

Y TRIM FAIL

AFCS has lost trim actuator drive capability in yaw.

Continue flight being aware that the trim function is unavailable in yaw.

COLLECTIVE TRIM FAIL

C TRIM FAIL

AFCS has lost trim actuator drive capability in collective.

Continue flight being aware that AFCS collective modes are not available.

AUTOPILOT OFF

1(2) AP OFF + Audio Tone and Voice Message "AUTOPILOT AUTOPILOT"

Associated AP channel not engaged.

Engage AP channel on APMS panel.

Caution clears

YES

Continue flight.

NO

Failure of associated AP channel.

- Maximum speed is 128 KIAS.
Above 500 ft AGL:
- Fly attentive.
At or below 500 ft AGL:
- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

If subsequent:

1 AP OFF

2 AP OFF

- Continue Flying manually not exceeding 128 KIAS. (100 KIAS in turbulence)

Note

VFR flight with one autopilot inoperative may be initiated flying manually to a repair facility.

AUTOPILOT FAIL

1(2) AP FAIL + Audio Tone and Voice Message "AUTOPILOT AUTOPILOT"

Failure of associated AP channel.

Do NOT attempt to re-engage the affected AP channel.

- Maximum speed is 128 KIAS.

Above *500* ft AGL:

- Fly attentive.

At or below *500* ft AGL:

- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

If subsequent:

1 AP FAIL

2 AP FAIL

- Continue Flying manually not exceeding 128 KIAS. (100 KIAS in turbulence)

Note

VFR flight with one autopilot inoperative may be initiated flying manually to a repair facility.

AUTOPILOT HOT

1(2) AP HOT

Associated FCC temperature is above limit.

Continue flight.

Be attentive for possible AP channel disengagement.

IF

1 AP HOT + **2 AP HOT**

Disengage one channel of the AFCS using the pushbutton AP 1(2) on the APMS.

- Maximum speed is 128 KIAS.
Above 500 ft AGL:
- Fly attentive.
At or below 500 ft AGL:
- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

PITCH AUTOPILOT FAIL

1(2) AP P FAIL + Audio Tone and Voice Message "AUTOPILOT AUTOPILOT"

Failure affecting a single series actuator in the pitch axis.

- Maximum speed is 128 KIAS.
Above 500 ft AGL:
- Fly attentive.
At or below 500 ft AGL:
- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

ROLL AUTOPILOT FAIL

1(2) AP R FAIL + Audio Tone and Voice Message "AUTOPILOT AUTOPILOT"

Failure affecting a single series actuator in the roll axis.

- Maximum speed is 128 KIAS.
Above 500 ft AGL:
- Fly attentive.
At or below 500 ft AGL:
- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

YAW AUTOPILOT FAIL

1(2) AP Y FAIL + Audio Tone and Voice Message
"AUTOPILOT AUTOPILOT"

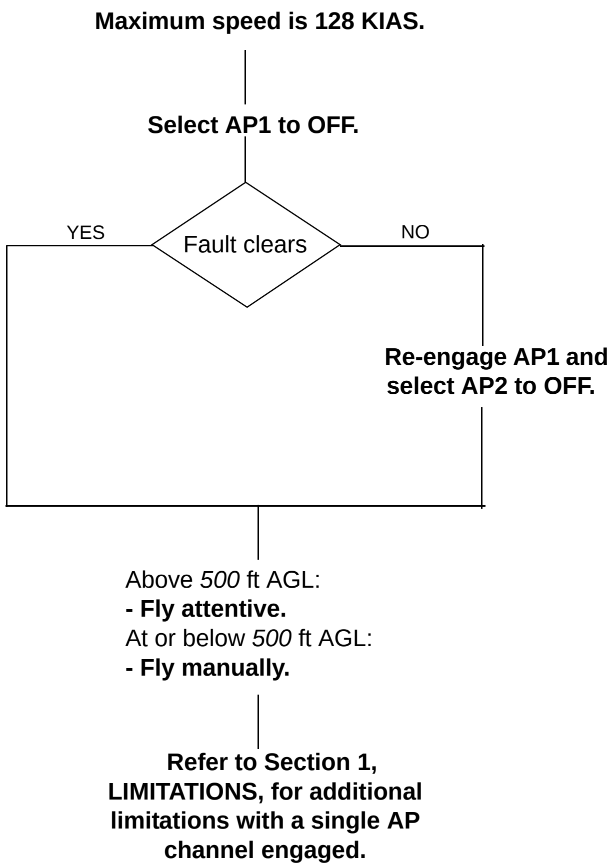
Failure affecting a single series actuator in the yaw axis.

- Maximum speed is 128 KIAS.
Above 500 ft AGL:
- Fly attentive.
At or below 500 ft AGL:
- Fly manually.

Refer to Section 1, LIMITATIONS, for additional limitations with a single AP channel engaged.

AFCS OSCILLATORY MALFUNCTION

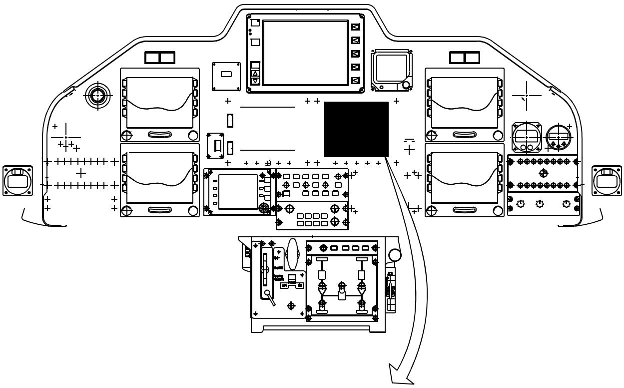
Repeated disturbances in one or more axes.



INTEGRATED DISPLAY SYSTEM FAILURE

EDU1 FAILURE

ELECTRONIC DISPLAY UNIT 1 COMPLETE FAILURE



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Complete loss of EDU1 display

IDS displayed on EDU2.

EDU2 automatically switched to reversionary mode.

Right pilot in control of helicopter

YES

dual pilot

YES

When required left pilot take control of helicopter.

Continue flight.

NO

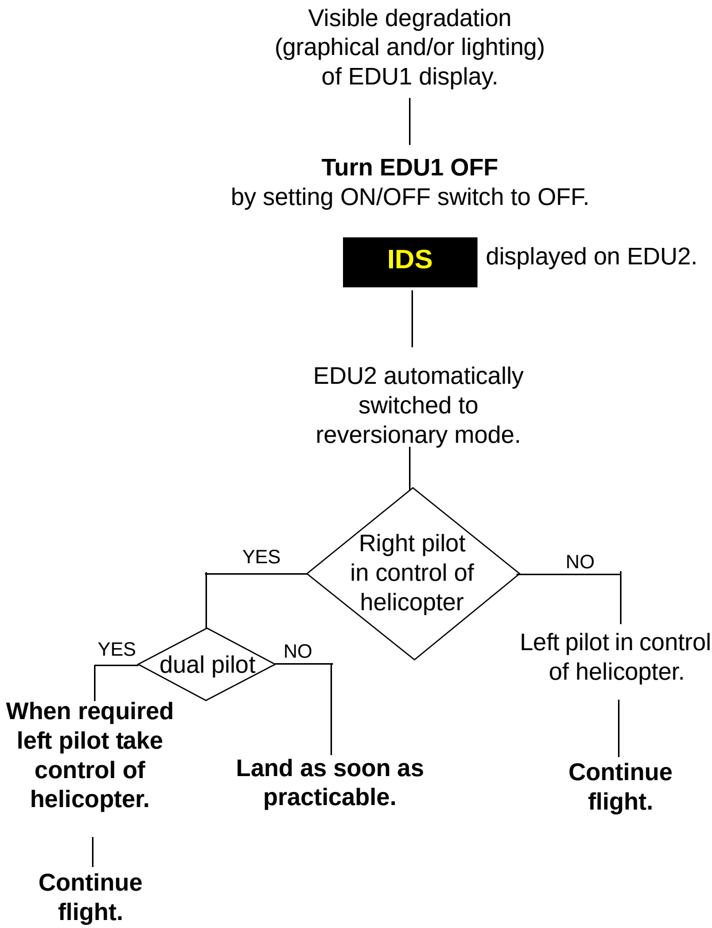
Land as soon as practicable.

NO

Left pilot in control of helicopter.

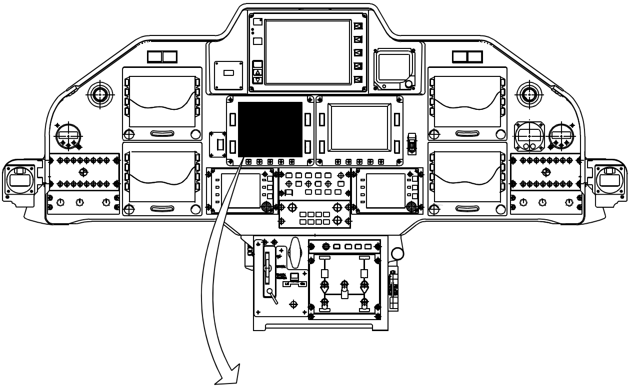
Continue flight.

ELECTRONIC DISPLAY UNIT 1 DEGRADATION



EDU2 FAILURE

ELECTRONIC DISPLAY UNIT 2 COMPLETE FAILURE



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Complete loss of EDU2

IDS displayed on EDU1.

EDU1 automatically switched to reversionary mode.

Right pilot in control of helicopter

YES

No action required. Right pilot must remain in control of helicopter.

Continue flight.

NO

Left pilot in control of helicopter.

When convenient right pilot take control of helicopter.

Continue flight.

ELECTRONIC DISPLAY UNIT 2 DEGRADATION

Visible degradation
(graphical and/or lighting)
of EDU2 display.

Turn EDU2 OFF
by setting ON/OFF switch to OFF.

IDS displayed on EDU1.

EDU1 automatically switched to
reversionary mode.

Right pilot
in control of
helicopter

YES

**No action required.
Right pilot must
remain in control of
helicopter.**

**Continue
flight.**

NO

Left pilot in control
of helicopter.

**When convenient right
pilot take control of
helicopter.**

**Continue
flight.**

MISCOMPARE OF DAU PRIMARY PARAMETERS

DAU MISCMP-P

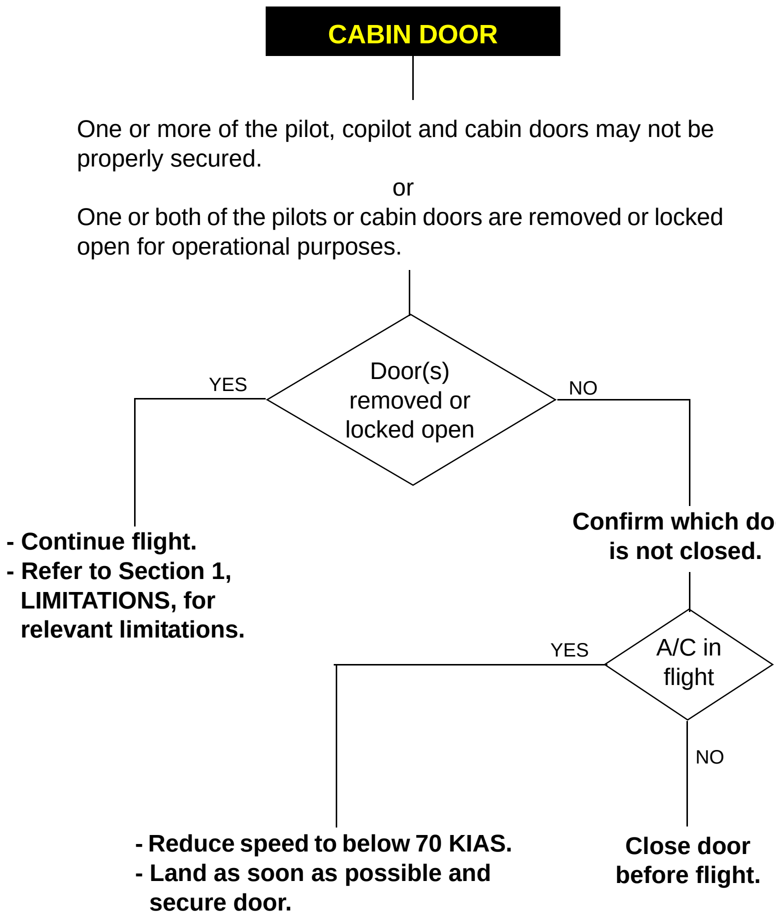
Miscompare of DAU primary parameters.
Some primary data from one or both channels of Data Acquisition Unit (DAU) are invalid.
Possible degradation in system functions.

- **Access MENU 2/3 page on EDU1 and check DAU channels status.**
- **Select DAU channel indicated by yellow legend and check for data discrepancy.**
- **Deselect affected channel to return to normal operation mode.**

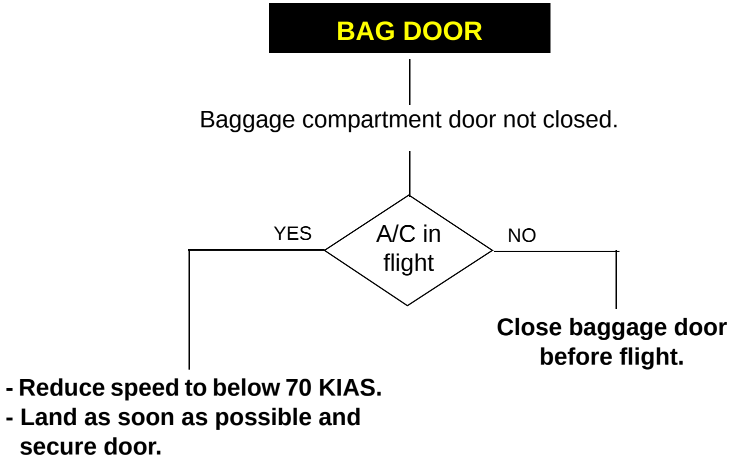
Continue flight.

MISCELLANEOUS

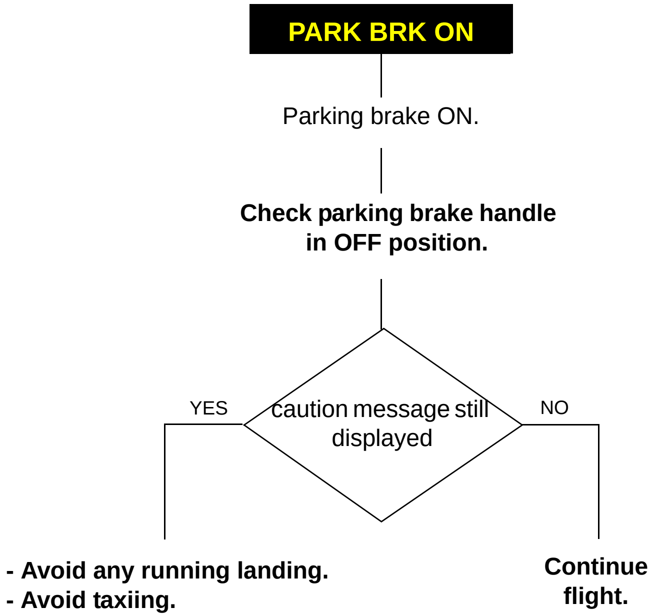
DOOR OPEN



BAGGAGE COMPARTMENT DOOR OPEN



PARKING BRAKE ON



PITOT HEAT FAILED

PITOT 1(2) FAIL

Associated pitot heat failed.

Avoid flight in visible moisture.

Continue flight.

COCKPIT FAN FAILED

COCKPIT FAN

Cockpit fan failed.

VENT CKPT switch: OFF

EFIS FANS FAILED

EFIS FAN

One or both EFIS FAN failed.

OAT
greater than 45 °C

NO

**Proceed
with flight.**

YES

Front ventilation ports : Open
VENT CKPT switch : Set to HIGH

If installed :
ECS switch : ON

LANDING GEAR RETRACTED

LANDING GEAR + Audio Tone and Voice Message
"LANDING GEAR"

Landing gear retracted when
height is less than 200 ft AGL.

Extend landing gear.

LANDING GEAR FAILS TO RAISE

Landing gear selected up but one or more green lights or the red light remains illuminated.

Confirm landing gear circuit breakers (2) not tripped.

Check EMER / NORMAL selector NOT in EMER position.

- Cycle landing gear from UP to DOWN.
- Check green lights illuminated.
- Select UP

If one or more green lights or the red light remains illuminated:
Select landing gear DOWN.

Continue flight.
Maximum speed = V_{LE} (140 KIAS).

CAUTION

The pilot should have due regard to the performance implications of prolonged flight with the landing gear extended.

AIR DATA UNIT FAILURE

ADC 1(2)

Associated Air Data Unit failed.

On the RCP, set the ADU knob to the unaffected ADU system.

Continue flight cross-checking ADU data with the ESIS data.

Note

In case of loss of ADC on side in command, Upper Modes and Flight Director Modes disengage. The modes can be re-engaged as necessary 2 minutes after system reconfiguration (advisory caption VSI BARO clears).

ATTITUDE AND HEADING REFERENCE SYSTEM FAILURE

AHRS 1(2)

Associated AHRS failed.

Also

AP AHRS1(2) FAIL

The AFCS is not receiving information from the associated AHRS.

On the RCP, set the AHRS knob to the unaffected AHRS system.

Continue flight cross-checking AHRS data with the standby instruments.

Refer to AUTOPILOT AHRS FAIL procedure. Page 3-137

Note

In case of loss of AHRS on side in command, Upper Modes and Flight Director Modes disengage. The modes can be re-engaged as required after system reconfiguration.

VNE EXCEEDED

Audio Tone and Voice Message
“AIRSPEED”.

Vne exceeded.

Reduce airspeed below Vne.

LOW HEIGHT

Audio Tone and Voice Message
“ALTITUDE ALTITUDE”.

Excessive deviation of altitude / height with respect to the pre-selected value.

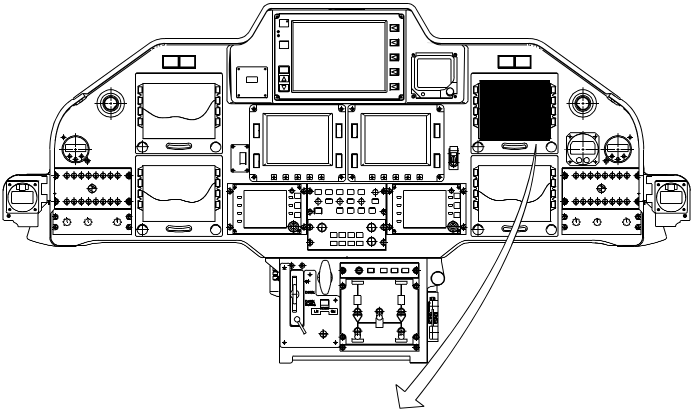
Check height and adjust as required.

CHELTON EFIS DISPLAYS FAILURE

An EFIS display failure can affect separately or simultaneously each of the four system displays (two PFDs and two MFDs).

The failure can be caused by an internal system failure. In the case of an internal system failure, the affected display (PFD or MFD) becomes blank.

FAILURE OF PILOT'S PFD



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CHECK IDU1 Displayed on Pilot's MFD.

Right pilot in control of helicopter

YES

ADC 2 Momentarily displayed on EDU2

ALT or ALTA modes disengage (if in use)

Flight Director modes disengage (if in use)

Pilot's MFD automatically reverts to PFD format.

Re-engage ALT, ALTA or Flight Director modes as necessary

Continue flight.

NO

Pilot's MFD automatically reverts to PFD format.

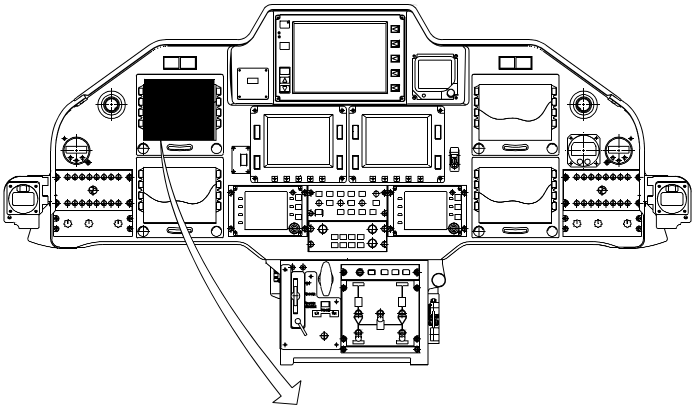
Continue flight.

<u>For EFIS with software version A109 7.0D</u>

CAUTION

Before re-engaging Flight Director modes confirm the selected navigation source has not been changed. Do not continue a Non Precision Approach.

FAILURE OF COPILOT'S PFD



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CHECK IDU1 Displayed on Copilot's MFD.

Left pilot in control of helicopter

YES

ADC 2 Momentarily displayed on EDU2

ALT or ALTA modes disengage (if in use)
Flight Director modes disengage (if in use)

Copilot's MFD automatically reverts to PFD format.

Re-engage ALT, ALTA or Flight Director modes as necessary

Continue flight.

NO

Pilot's MFD automatically reverts to PFD format.

Continue flight.

For EFIS with software version A109 7.0D

CAUTION

Before re-engaging Flight Director modes confirm the selected navigation source has not been changed. Do not continue a Non Precision Approach.

ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS) ALERTING SYSTEM

The Malfunctions and Discrepancies described in this section are indicated by the illumination of the yellow caution captions on the EFIS CAS window.

Table 3-3 Caution Annunciation Captions

Flag	Aural Annun.	Page	Malfunction
ADC1(2) FAIL	Alert Tone	3-170	ADC #1(2) failure.
AHRS1(2) FAIL	Alert Tone	3-171	AHRS #1(2) Failure.
ALT MISCOMP	Alert Tone	3-172	Altitude Miscompare.
ATT MISCOMP	Alert Tone	3-172	Attitude Miscompare.
CHECK IDU1(2)	Alert Tone	3-167 3-168	IDU1(2) fail the internal check
EFIS COOL	"EFIS Cooling"	3-173	IDU Overtemperature.
GPS LOI	Alert Tone	3-177	GPS/Loss of Integrity.
GPS LON	Alert Tone	3-176	GPS/ Loss of Navigation.
GPS1 FAIL	Alert Tone	3-175	GPS #1 Failure.
GPS2 FAIL	Alert Tone	3-174	GPS #2 Failure.
GPS MISCOMP	Alert Tone	3-175	GPS Miscompare.
GS MISCOMP	Alert Tone	3-178	Glideslope Miscompare.
HDG MISCOMP	Alert Tone	3-179	Heading Miscompare.
IAS MISCOMP	Alert Tone	3-180	Airspeed Miscompare.
IDU MISCOMP	Alert Tone	3-181	PFD vs MFD Miscompare
IDU POWER	Alert Tone	3-180	Power Supply Fail.
LOC MISCOMP	Alert Tone	3-181	Localizer Miscompare.
NO HEADING	Alert Tone	3-171	Heading Failure.
NO GPS	"GPS Failure"	3-174	GPS Failure.
OAT SENSOR	Alert Tone	3-183	OAT Sensor Failed.
RADALT FAIL	Alert Tone	3-184	Radar Altimeter Failure.
SCC FAIL	Alert Tone	3-185	SCC Card Failed.

LOSS OF DATA FROM AIR DATA COMPUTER

ADC 1(2) FAIL

displayed on copilot's (pilot's) MFD/PFD.

ADC 1(2)

also displayed on EDU1.

Associated ADC failed.

Loss of helicopter air data (altimeter, airspeed and vertical speed) related to affected ADC.

Refer to
AIR DATA UNIT FAILURE
procedure Page 3-163 .

LOSS OF DATA FROM ATTITUDE HEADING REFERENCE SYSTEM

AHRS 1(2) FAIL and/or **NO HEADING**

displayed on copilot's (pilot's) MFD/PFD.

AHRS 1(2) + **AP AHRS 1(2) FAIL**

also displayed on EDU1.

Associated AHRS failed.

Loss of helicopter attitude and heading data related to affected AHRS.

Refer to ATTITUDE AND HEADING REFERENCE SYSTEM FAILURE procedure Page 3-164.

ALTITUDE DATA DISCREPANCY

ALT MISCOMPARE

displayed on pilot's and copilot's MFD/PFD.

Altitude data discrepancy between
the two ADC.
ALT and ALTA modes disengage.

**Compare PFD's data with the
ESIS data.**

**On the RCP, set the ADU knob to the ADU system
displaying similar data to the ESIS data.**

**Continue flight cross-checking PFD altitude data to
the ESIS data.**

Re-engage AFCS modes as necessary

PITCH AND ROLL ATTITUDE DATA DISCREPANCY

ATT MISCOMPARE

displayed on pilot's and copilot's MFD/PFD.

Pitch and roll attitude data discrepancy
between the two AHRS's.

**Compare PFD's data with the
ESIS data.**

**On the RCP, set the AHRS knob to the AHRS system
displaying similar data to the ESIS data.**

**Continue flight cross-checking PFD pitch and roll
attitude data to the ESIS data.**

IDU OVERTEMPERATURE

EFIS COOL

displayed on pilot's and copilot's MFD/PFD.

IDU core temperature above 95°C

Possible display failure if temperature reaches 98°C

In case of display failure
Refer to EFIS DISPLAYS FAILURE procedure Page 3-166.

GPS2 SYSTEM FAILURE/MALFUNCTION

GPS LON

After 5 seconds

GPS 2 FAIL and **NO GPS**

+

~~**TERRAIN**~~

displayed on pilot's MFD/PFD.

Pilot's GPS system failed with loss of GPS navigation. HOV mode may disengage. VOR, VAPP, LOC/NLOC and GS/NGS modes may be degraded. TERRAIN and MAP functions not available.

On the RCP, set the GPS knob to the GPS1 position.

Continue flight.

Whenever required revert to other navigational sources.

GPS1 SYSTEM FAILURE/MALFUNCTION

GPS 1 FAIL

displayed on copilot's MFD/PFD.

Copilot's GPS system failed with loss of GPS navigation, TERRAIN and MAP functions on copilot side.

Continue flight.

GPS LOSS OF NAVIGATION

GPS LON + ~~TERRAIN~~

displayed on pilot's MFD/PFD.

Loss of GPS 2 navigation.
HOV mode may disengage. VOR, VAPP, LOC/NLOC and GS/NGS modes may be degraded. TERRAIN and MAP functions not available.

On the RCP, set the GPS knob to the GPS1 position.

SAME NAV + **SAME GPS**

displayed on pilot's MFD/PFD.

Continue flight.

Whenever required revert to other navigational sources.

GPS LOSS OF INTEGRITY

GPS LOI

displayed on pilot's MFD/PFD.

Loss of GPS 2 integrity.

On the RCP, set the GPS knob to the GPS1 position.

SAME NAV **+** **SAME GPS**

displayed on pilot's MFD/PFD.

Continue flight.

CAUTION

If GPS LOI persists revert to other navigational sources.

GPS DATA DISCREPANCY

GPS MISCOMPARE

displayed on pilot's and copilot's MFD/PFD.

Excessive discrepancy between the two GPS units data.

On the RCP, set the GPS knob to the GPS system displaying valid data.

Continue flight.

EXCESSIVE GLIDE SLOPE DISCREPANCY

GS MISCOMPARE

displayed on pilot's and copilot's MFD/PFD.

Excessive discrepancy between the two glide slope's data.

Confirm that both ILS receivers are tuned to the same frequency.

If both receivers are tuned to same frequency:
Abort ILS approach.

HEADING DATA DISCREPANCY

HDG MISCOMPARE

displayed on pilot's and copilot's MFD/PFD.

Heading data discrepancy betweenthe two AHRS.
The HDG, HOV, LOC/NLOC and GS/NGS modes may disengage.

Compare PFD's data with
Stand-by magnetic compass data.

On the RCP, set the AHRS knob to the AHRS system displaying similar data to the Stand-by magnetic compass data.

Re-engage AFCS modes as necessary

Continue flight cross-checking PFD heading data to the Stand-by magnetic compass data.

CAUTION

Do not use the SRCH LT during heading comparison.

AIRSPEED DATA DISCREPANCY

IAS MISCOMPARE

displayed on pilot's and copilot's MFD/PFD.

Airspeed data discrepancy betweenthe two ADCs.
Upper Modes and Flight Director Modes will disengage.

Compare PFD's data with ESIS data.

On the RCP, set the ADU knob to the ADU system displaying similar data to ESIS data.

Continue flight cross-checking PFD airspeed data to the ESIS data.

Re-engage AFCS Modes as necessary.

LOSS OF IDU POWER REDUNDANCY

IDU POWER

affected display has lost power redundancy.

Continue flight.

IDU DISCREPANCY

IDU MISCOMP

Displayed on pilot's MFD/PFD (right side).

|

Excessive discrepancy between critical parameter processed by PFD and MFD.

|

Disengage HOV, ALT, ALTA and Flight Director modes

|

Switch MFD to PFD format.

|

Cross check the following parameters between the 2 displays:
- Pressure Altitude
- Indicated Airspeed
- Pitch
- Roll
- Heading
- Glideslope
- Localizer

|

When a difference is detected, cross check the displays data with the ESIS data. The one displaying different data is the failed display.

|

Dim the failed display to minimum brightness and continue flight using the functioning display in PFD format.

|

If the functioning display is PFD (upper display)**, re-engage HOV, ALT, ALTA and Flight Director modes as necessary.**

Note

In case the caution is displayed on copilot's MFD/PFD while the left pilot is in control of the helicopter, continue flight with the right pilot in control of helicopter.

EXCESSIVE LOCALISER DISCREPANCY

LOC MISCOMPARE

displayed on pilot's and copilot's MFD/PFD.

Excessive discrepancy between
the two localiser's data.

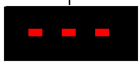
Confirm that both ILS receivers are tuned to the same frequency.

If both receivers are tuned to the same frequency:
Abort ILS approach.

LOSS OF OAT DATA

OAT SENSOR

displayed on pilot’s and/or copilot’s MFD/PFD.



also displayed on EDU2.

Loss of OAT data.

On the RCP, set the ADU knob to the ADU system displaying valid OAT data.

Note

When there is a loss of OAT data, the “AIRSPEED” voice caution will be activated at the lowest value of V_{NE} for the current pressure altitude.

RADIO ALTIMETER DATA LOSS

RADALT FAIL

displayed on pilot's and copilot's MFD/PFD.

Loss of radio altimeter height data on a MFD/PFD.
RHT mode disengage and LOC/NLOC, GS/NGS modes may be degraded (if in use).

Refer to barometric altimeter

Continue flight.

CAUTION

Below 2500 ft AGL, height data can still be provided by GPS systems (height data followed by a "G") or by the ADC systems (height data followed by a "B").

SCC CARD FAILURE

SCC FAIL

displayed on pilot's and copilot's MFD/PFD.

SCC card cannot be read upon power up

Do not take off
Shut down Engines

Correct problem before flight..

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SECTION 4

PERFORMANCE DATA

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SECTION 4

PERFORMANCE DATA

GENERAL

The performance data presented herein is derived from the engine-manufacturer's specification power for the engine less installation losses. This data is applicable to the basic helicopter without any optional equipment which would appreciably affect lift, drag, or power available.The performance shown is based on (1) adherence to the operational limitations shown in Section 1 of this manual, (2) the use of the procedures described in Sections 2 and 3 of this manual and (3) periodic engine power checks to assure minimum specification engine power output. The performance at a weight lower than the lowest weight shown on any particular graph should be assumed to be no better than that at the lowest weight shown. Performance extrapolation beyond the ranges shown in the graphs is not permitted.

USING THE GRAPHS

The performance information is presented graphically and in the sequence in which it would ordinarily be used. The introduction to each performance item contains an example of graph usage. Additionally, each graph contains a pictogram showing the correct sequence of steps in graph usage. As aircraft performance can be substantially affected by many factors (weight, altitude, temperature, power levels, etc.), careful attention should be paid to the explanatory text accompanying each graph to assure that the correct set of factors is being used.

When quick estimates of performance are required (no interpolation between curves), it is recommended that the most critical curves be used for the estimates (higher weight, higher density altitude, etc.).

DENSITY ALTITUDE CHART

(Figure 4-1)

Enter the graph at the desired value of Outside Air Temperature (OAT), on the horizontal axis. Proceed vertically until intercepting the desired value of Pressure Altitude. Proceed left to read the resulting Density Altitude (in feet). Proceed right to read the corresponding value of the reciprocal of the square root of the density ratio, which is used to calculate True Airspeed.

Example:

At a Pressure Altitude of 5000 ft and an OAT of +25°C, determine the corresponding Density Altitude.

Solution:

Enter Figure 4-1 at an OAT of +25°C, and proceed vertically until intersecting a Pressure Altitude of 5000 ft (diagonal curves). From this point, proceed left to read a Density Altitude of 7300 ft.

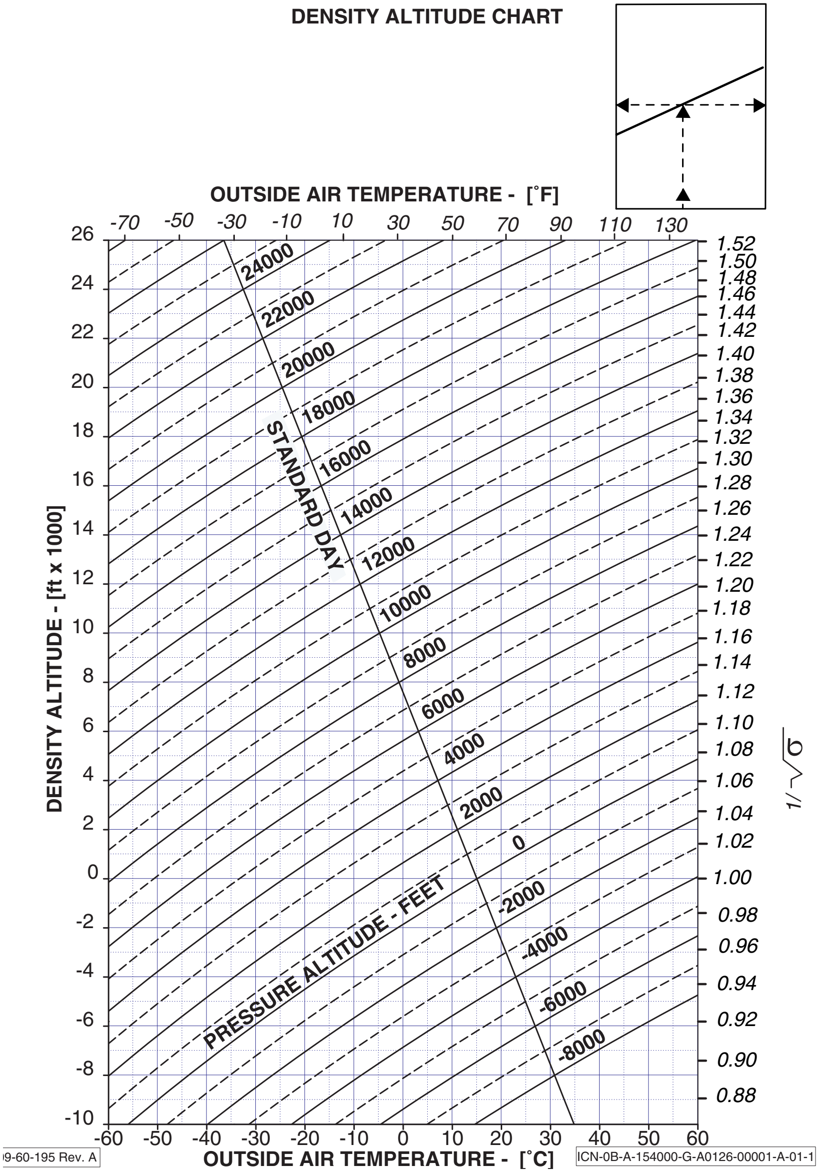

Figure 4-1 Density / Altitude Chart

CONVERSION CHART

(Figure 4-2)

The table in Figure 4-2 is self-explanatory. The conversion from Metric System values to Imperial System values, and vice-versa, is given for a number of useful flight parameters.

CONVERSION CHART

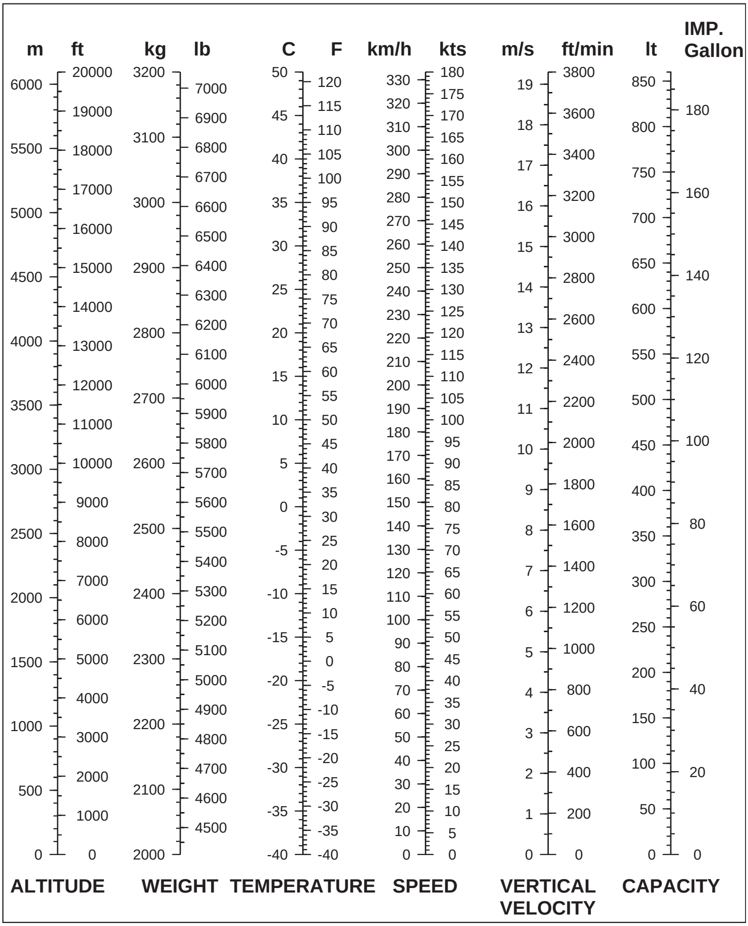

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Figure 4-2 Conversion Chart

AIRSPEED CALIBRATION CURVE

(Figures 4-3 to 4-5)

Calibration curves are presented for the pilot's and co-pilot's Air Data System. The Indicated Airspeed shown on the cockpit instruments is produced by pressure readings at the aircraft pitot and static ports and, in general, these pressure readings contain errors which must be corrected in order to obtain Calibrated Airspeed (that airspeed which would be shown on the cockpit instruments if there were no errors). Calibrated airspeed must be used in order to calculate correct values of the True Airspeed.

The magnitude of the error depends on the Indicated Airspeed. Enter the appropriate Airspeed Calibration Curve graph (Figure 4-3 to Figure 4-5) with the desired value of Indicated Airspeed, and proceed vertically until intercepting the curve. Proceed to the left to read the corresponding value of Calibrated Airspeed. Use this value to calculate the corresponding True Airspeed, as required.

Example:

On the pilot's Air Data System Indication, determine the Calibrated Airspeed corresponding to an Indicated Airspeed of 140 knots.

Solution:

Use Figure 4-3 .
Enter the graph at an Indicated Airspeed of 140 knots and proceed vertically until intersecting the curve. Proceed left and read the corresponding Calibrated Airspeed of 127 knots.

AIRSPEED CALIBRATION
PILOT INDICATOR

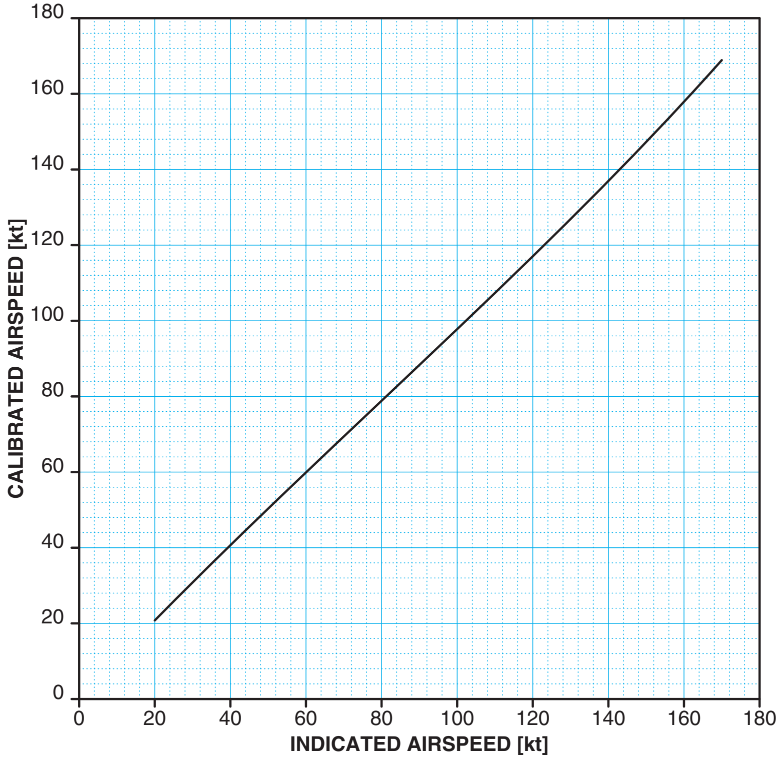

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Figure 4-3 Airspeed Calibration Curve - Pilot

AIRSPEED CALIBRATION
CO-PILOT INDICATOR

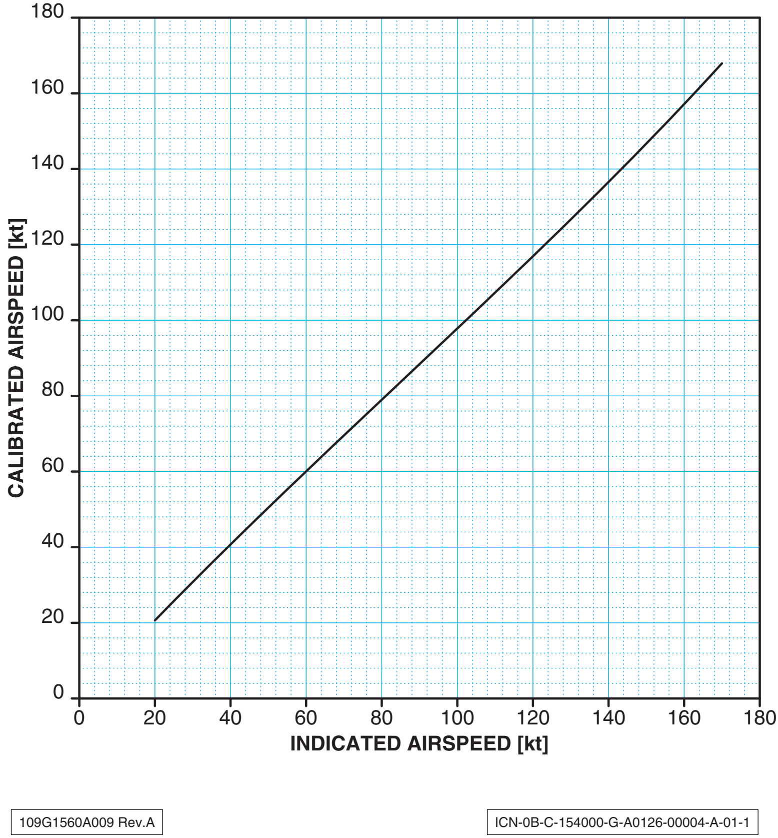

Figure 4-4 Airspeed Calibration Curve - Co-pilot

AIRSPEED CALIBRATION
ESIS INDICATOR

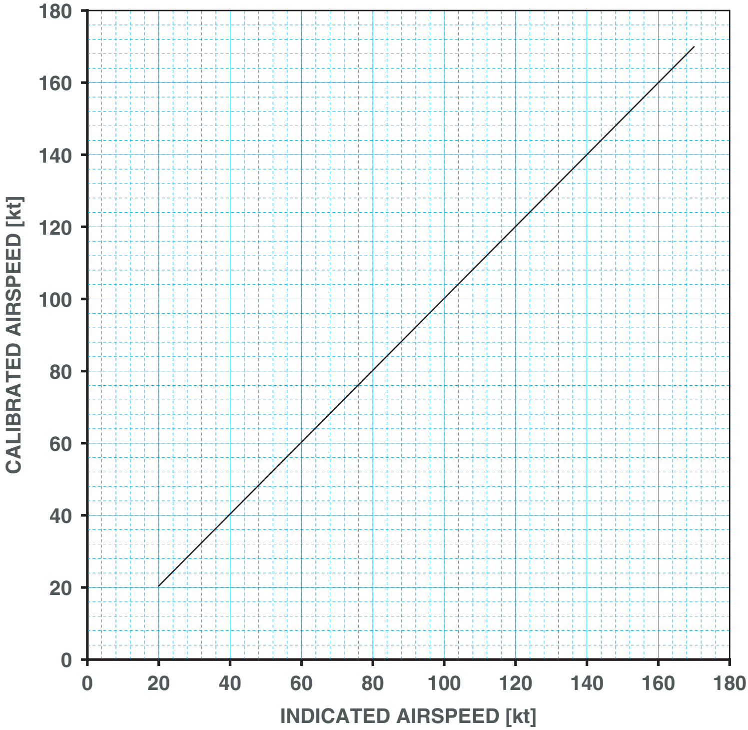

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Figure 4-5 Airspeed Calibration Curve - ESIS

ENGINE POWER ASSURANCE CHECKS

(Figures 4-6 and 4-7)

The purpose of the Engine Power Assurance Checks is to provide a means of monitoring engine health on an on-going basis. Two check types are provided for this purpose:

- a Ground/Hover Power Assurance Check
- a Level Flight Power Assurance Check.

A Ground/Hover Power Assurance Check procedure should be used to check if the engine power available, before take-off, is within the limits to guarantee the performance presented in this Section.

Another procedure is provided, to enable the operator to check the engine power in level flight. The procedure should be used to check if the power available is within the limits to guarantee the performance presented in this Section and maintain a trend chart to monitor the deterioration of the engine with operating hours.

POWER CHECK INTERVALS

Either ground or inflight power check shall be accomplished:

— at intervals not exceeding 50 flying hours or monthly, whichever comes first, for non Category A operations.

— whenever abnormal engine function is suspected.

GROUND/HOVER POWER ASSURANCE CHECK

The following procedure is used to check engine performance:

Note

A power assurance check conducted in visible moisture may be unreliable.

1. Position the helicopter into the prevailing wind to minimize hot gas ingestion.
2. Record date, helicopter serial number, aircraft hours, engine serial number and engine hours.

3. Set the barometric pressure to 1013 mb (29.92 inches of Hg).

4. Confirm that the heater/ECS is set to OFF.

5. Set the ENG MODE 1/2 switch of the test engine to FLT position and the remaining engine to IDLE position.

6. With the rotor speed at 102%, set the collective (until condition “light on wheels” or hovering 3 feet) so as to obtain a percent torque value within the range shown on the left graph of Figure 4-5 depending on the airfield elevation.
   Do not exceed: 840 °C TOT or 97.1% N1 or 125% TRQ.

7. Maintain a fixed collective for one minute and, then, record the following data:

   — TRQ (Engine Torque)
   — Pressure Altitude
   — OAT
   — TOT
   — N1

8. When recording is completed, lower collective to MPOG, set the ENG MODE 1/2 switch of test engine to IDLE position and plot readings on Figure 4-6:

   a) Enter the left graph on Figure 4-6 with the recorded engine torque value. Drop vertically until intercepting the recorded Pressure Altitude value (interpolate between the curves, if necessary).
      Move horizontally to the right to intercept the recorded OAT value to obtain the maximum allowable TOT and N1 values for the test condition.

   b) Compare the maximum allowable TOT and N1 values to the recorded TOT and N1 values respectively.

c) If the recorded TOT or N1 value is less than the maximum allowable TOT or N1 value, engine condition is acceptable for flight.
The difference between the maximum allowable TOT/N1 and the recorded TOT/N1 is called the Power Assurance Margin (PAM).

d) If the recorded TOT or N1 value is greater than the maximum allowable TOT or N1, the performance in this Section may not be achieved. Refer to Engine Maintenance Manual (EMM) for required actions.

Repeat steps 1. through 8., as necessary, for the other engine.

Example:

A Ground/Hover Power Assurance Check on Number 1 engine produced the following recorded results:

— TRQ = 92%

— Pressure Altitude = 3000 ft

— OAT = 0 °C

— TOT = 705 °C

— N1 = 90%

Solution:

Entering the left graph of Figure 4-6 at 92% engine torque, drop down to the Pressure Altitude curve for 3000 ft. Move right to the 0 °C OAT curve to obtain the maximum allowable TOT and the maximum allowable N1. From the 0 °C OAT curves move vertically up to obtain a TOT value of 735 °C and an N1 value of 91.4%.

The recorded TOT value of 705 °C is less than the maximum allowable TOT (735 °C) and the recorded N1 value of 90% is less than the maximum allowable N1 (91.4%), and therefore the engine is acceptable for flight.

LEVEL FLIGHT POWER ASSURANCE CHECK

The following procedure is used to check engine performance in-flight:

Note

A power assurance check conducted in visible moisture may be unreliable.

1. Record date, helicopter serial number, aircraft hours, engine serial number and engine hours.
2. Set the barometric pressure to 1013 mb (29.92 inches of Hg).
3. Confirm that the heater/ECS is set to OFF.
4. Establish level flight at 100 KIAS above 1000 ft AGL.

Note

For best results, it is recommended that the POWER CHECK procedure be carried out in calm conditions.

5. Set ENG GOV switch to MANUAL position for the engine to be tested, and leave the ENG GOV switch to AUTO for the other engine. Set NR to 100%.
6. On test engine, set the torque by using the TRIM switch on collective grip, observing the following constraints:
   a) on test engine (in MANUAL mode control), do not exceed: 840 °C TOT or 97.1% N1 or 100% TRQ;
   b) on the other engine (in AUTO mode control), do not allow TRQ to decrease below 20%.

7. Maintain a fixed collective setting (once set, do not adjust collective again) for one minute. It is acceptable for the airspeed to vary from the datum by ±10 KIAS to maintain constant altitude.
   Record the following data for the test engine:

   — TRQ (Engine Torque)

   — Pressure Altitude

   — OAT

   — TOT

   — N1

8. When recording is completed, set the ENG GOV switch to AUTO for the test engine and plot readings on Figure 4-7:

   a) Enter the left graph of Figure 4-7 with the recorded engine torque value. Drop vertically until intercepting the recorded Pressure Altitude value (interpolate between the curves, if necessary).
      Move horizontally to the right to intercept the recorded OAT value to obtain the maximum allowable TOT and N1 values for the test condition.

   b) Compare the maximum allowable TOT and N1 values to the recorded TOT and N1 values respectively.

   c) If the recorded TOT or N1 value is less than the maximum allowable TOT or N1 value, engine condition is acceptable for flight.
      The difference between the maximum allowable TOT/N1 and the recorded TOT/N1 is called the Power Assurance Margin (PAM).

   d) If the recorded TOT or N1 value is greater than the maximum allowable TOT or N1, the performance in this Section may not be achieved, carry out a GROUND/HOVER POWER ASSURANCE CHECK procedure.

9. Repeat steps 1. through 8., as necessary, for the other engine.

Example:

Use the same method as for the GROUND/HOVER POWER ASSURANCE CHECK.

* HEATER/ECS OFF
* GENERATOR LOAD TO MINIMUM
* SET NR to 102%

POWER ASSURANCE CHECK PW207C ENGINES HOVER

* TEST ENGINE MODE SWITCH: FLIGHT
* OTHER ENGINE MODE SWITCH: IDLE

* INCREASE COLLECTIVE UNTIL LIGHT ON WHEELS OR HOVERING AT 3 FEET, NOSE INTO THE WIND. DO NOT EXCEED 840 C TOT OR 97.1% N1 OR 125 % TORQUE.
* STABILIZE POWER ONE MINUTE, THEN RECORD OAT, PRESSURE ALTITUDE, ENGINE TORQUE, TOT AND N1.
* ENTER CHART AT INDICATED TORQUE, MOVE DOWN TO INTERSECT PRESSURE ALTITUDE, PROCEED TO THE RIGHT TO INTERSECT OAT, THEN MOVE UP TO READ VALUES FOR MAXIMUM ALLOWABLE TOT AND GAS PRODUCER RPM (N1).
* IF INDICATED TOT OR N1 EXCEEDS MAXIMUM ALLOWABLE VALUE, REPEAT CHECK, STABILIZING POWER FOR TWO MINUTES.
* REPEAT CHECK USING OTHER ENGINE
*IF EITHER ENGINE EXCEEDS ALLOWABLE TOT OR N1, PUBLISHED PERFORMANCE MAY NOT BE ACHIEVABLE. REFER TO EMM.

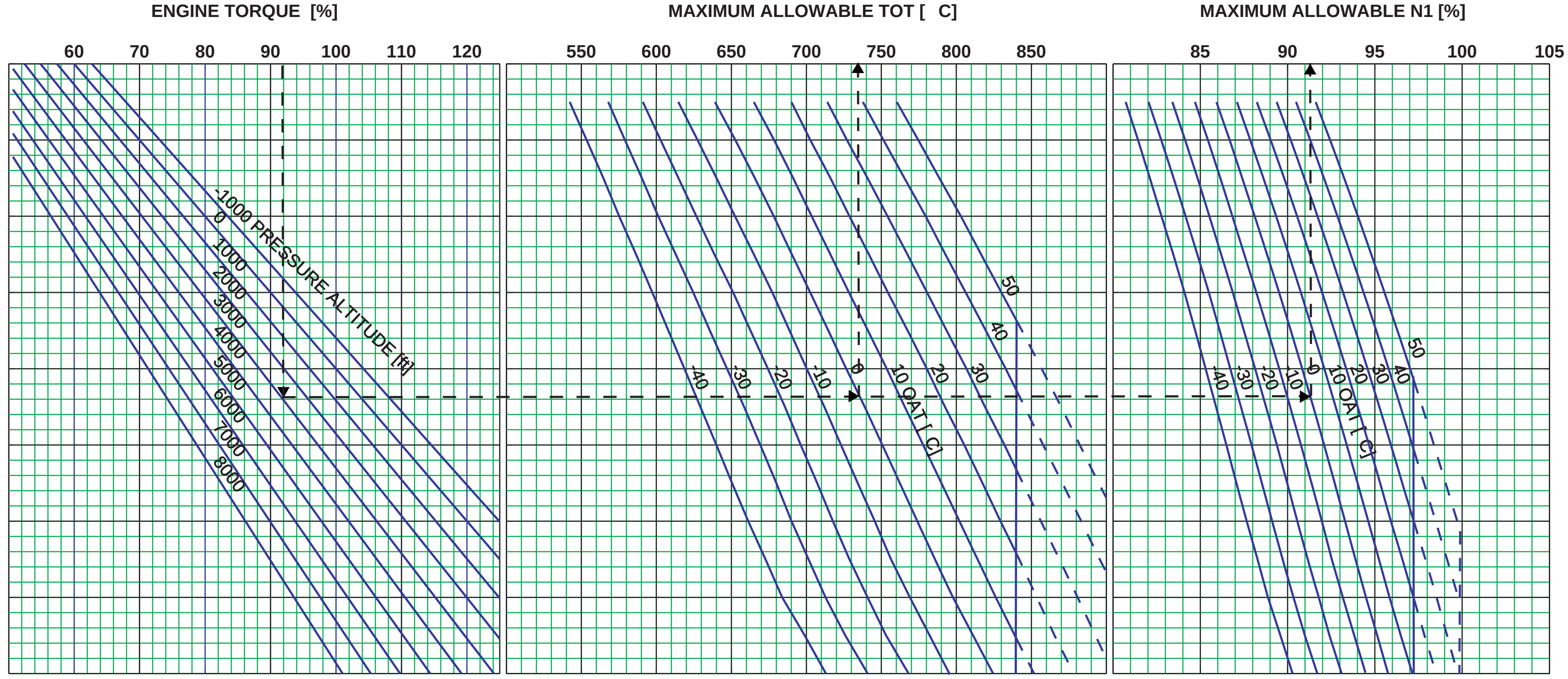

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Figure 4-6 Power Assurance Check in Hover

* HEATER/ECS OFF
* GENERATOR LOAD TO MINIMUM
* SET NR to 100%

POWER ASSURANCE CHECK PW207C ENGINES IN FLIGHT

* TEST ENGINE: SET ENG. GOV. SWITCH TO MANUAL
* OTHER ENGINE : LEAVE ENG. GOV. SWITCH TO AUTO

* ESTABLISH LEVEL FLIGHT AT LEAST 1000 FT AGL
* AIRSPEED=100 ±10 KIAS

* OPERATE TEST ENG TRIM TO INCREASE POWER UNTIL ENGINE TORQUE IS WITHIN TEST RANGE. DO NOT EXCEED 840 C TOT OR 97.1% N1 OR 100% TORQUE.
* STABILIZE POWER ONE MINUTE, THEN RECORD OAT, PRESSURE ALTITUDE, ENGINE TORQUE, TOT AND N1.
* ENTER CHART AT INDICATED TORQUE, MOVE DOWN TO INTERSECT PRESSURE ALTITUDE, PROCEED TO THE RIGHT TO INTERSECT OAT, THEN MOVE UP TO READ VALUES FOR MAXIMUM ALLOWABLE TOT AND GAS PRODUCER RPM (N1).
* IF INDICATED TOT OR N1 EXCEEDS MAXIMUM ALLOWABLE VALUE, REPEAT CHECK, STABILIZING POWER TWO MINUTES.
* REPEAT CHECK USING OTHER ENGINE.
* IF EITHER ENGINE EXCEEDS ALLOWABLE TOT OR N1, AFTER STABILIZING TWO MINUTES, CARRY OUT A POWER ASSURANCE CHECK IN HOVER.

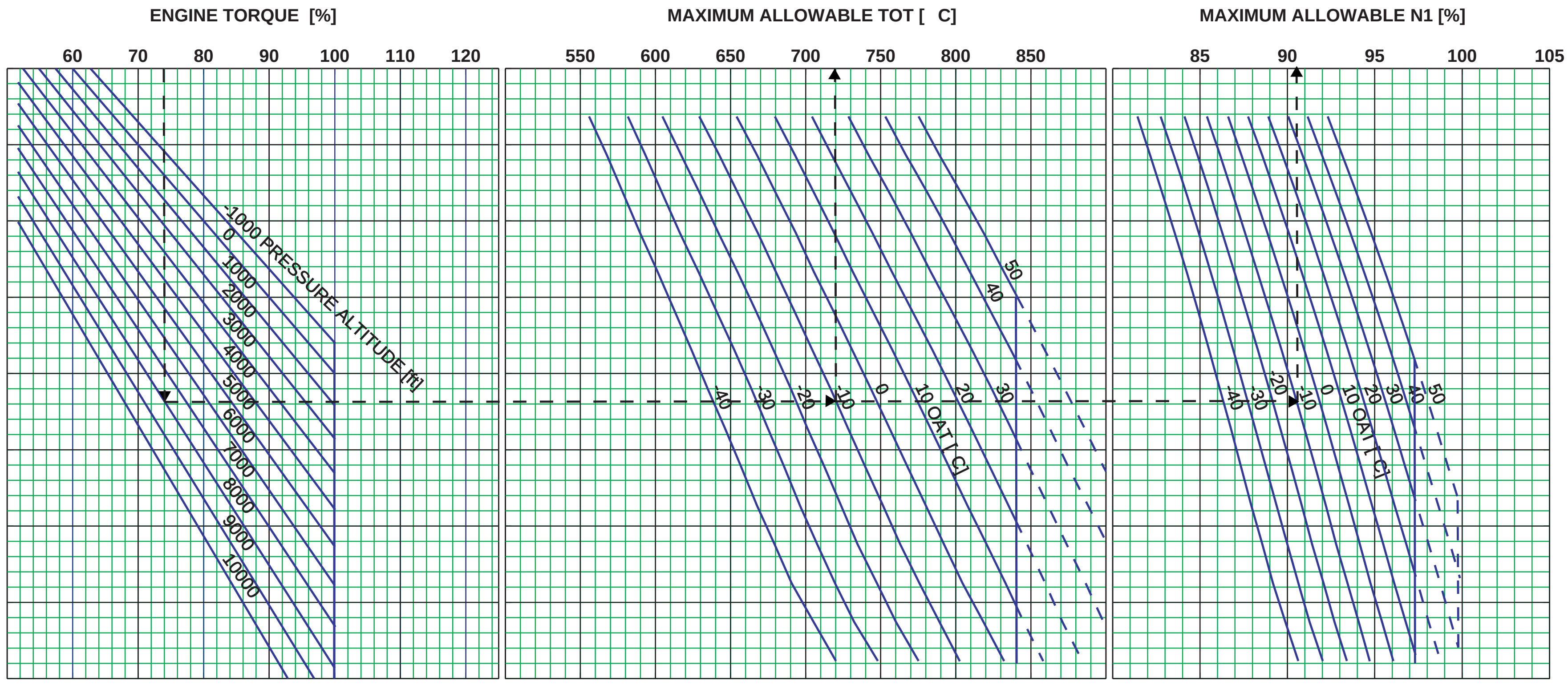

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Figure 4-7 Power Assurance Check in Flight

HOVER CEILING

(Figures 4-8 to 4-13)

The Hover Ceiling charts define the maximum weights at which an IGE hover (at 3 ft wheel height) or an OGE hover (60 ft AGL) is possible for varying combinations of Pressure Altitude and OAT with main rotor speed (NR) at 102% and zero wind conditions.

For hover in sidewind condition, refer to OPERATION VS ALLOWABLE WIND procedure in this Section or of the applicable Optional Equipment.

Information is presented for All Engines Operating (AEO) at Take-Off Power (TOP) and Maximum Continuous Power (MCP).

Note

For OAT range refer to Section 1, Figure 1-4 “Altitude / OAT Envelope”.

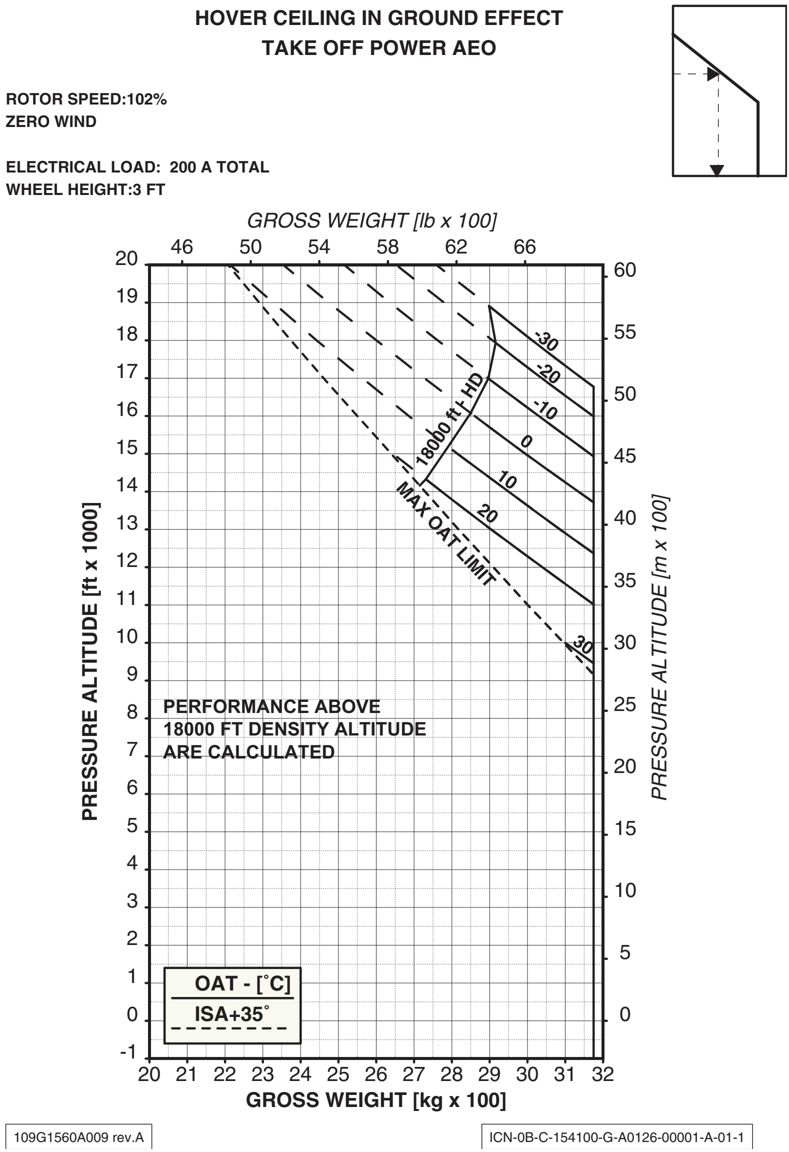

Figure 4-8 Hover Ceiling - IGE at Take-Off Power - AEO

HOVER CEILING IN GROUND EFFECT
MAXIMUM CONTINUOUS POWER AEO

ROTOR SPEED:102% ELECTRICAL LOAD: 200 A TOTAL
ZERO WIND WHEEL HEIGHT:3 FT

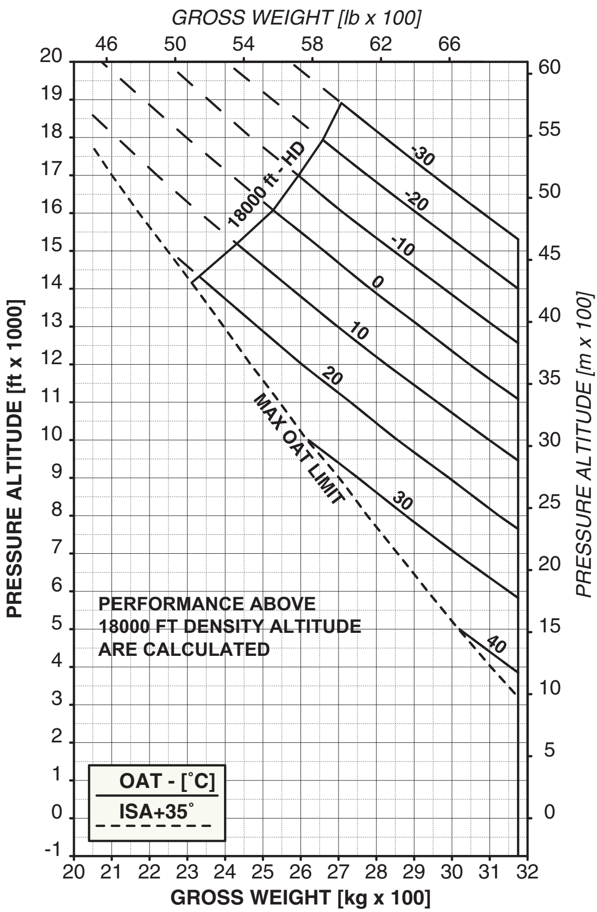

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Figure 4-9 Hover Ceiling - IGE at Maximum Continuous Power - AEO

HOVER CEILING OUT OF GROUND EFFECT
TAKE OFF POWER AEO

ROTOR SPEED:102% ELECTRICAL LOAD: 200 A TOTAL
ZERO WIND

CAUTION: HOGE OPERATION MAY RESULT IN VIOLATION OF H-V

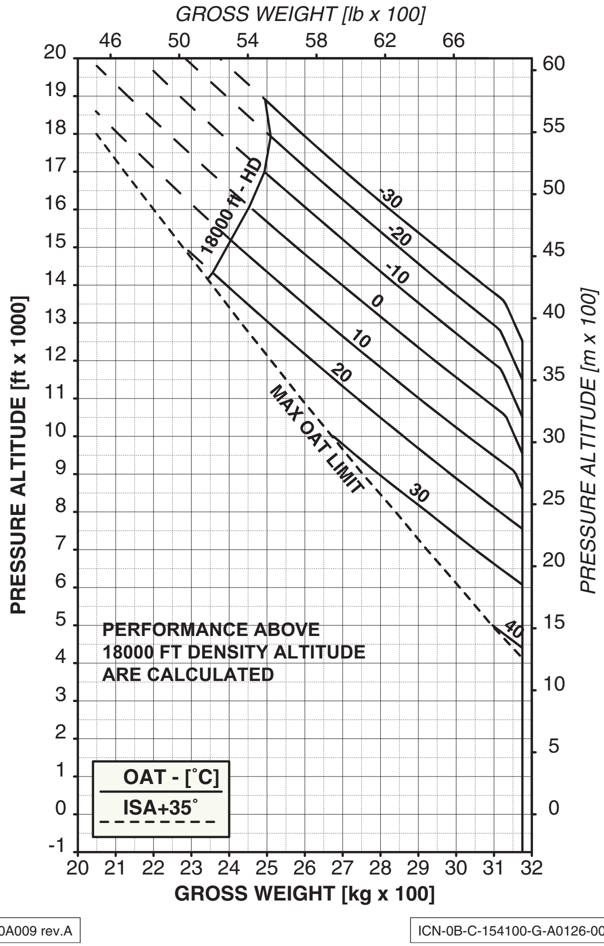

Figure 4-10 Hover Ceiling - OGE at Take-Off Power - AEO

HOVER CEILING OUT OF GROUND EFFECT
MAXIMUM CONTINUOUS POWER AEO

ROTOR SPEED:102%
ZERO WIND

ELECTRICAL LOAD: 200 A TOTAL

CAUTION: HOGE OPERATION MAY RESULT IN VIOLATION OF H-V

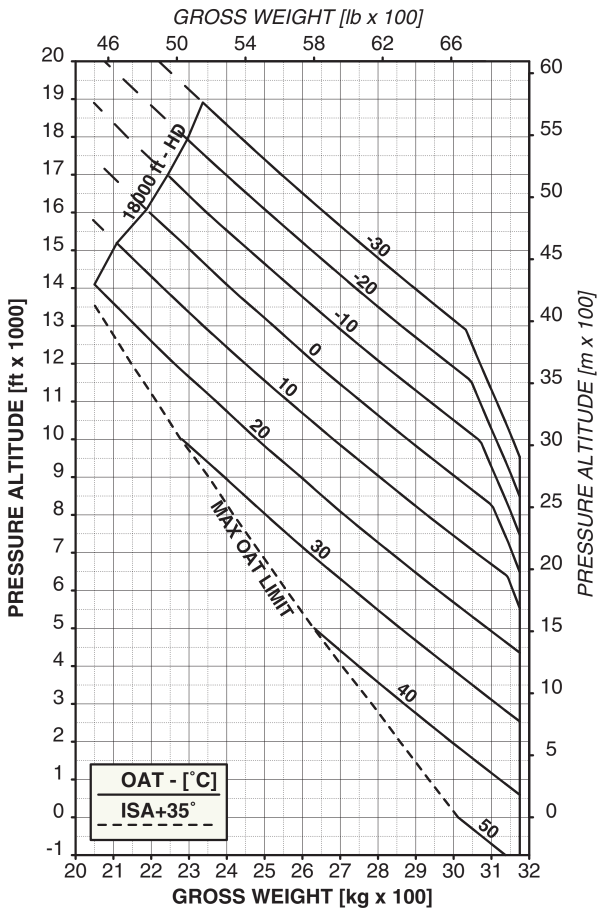

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Figure 4-11 Hover Ceiling - OGE at Maximum Continuous Power - AEO

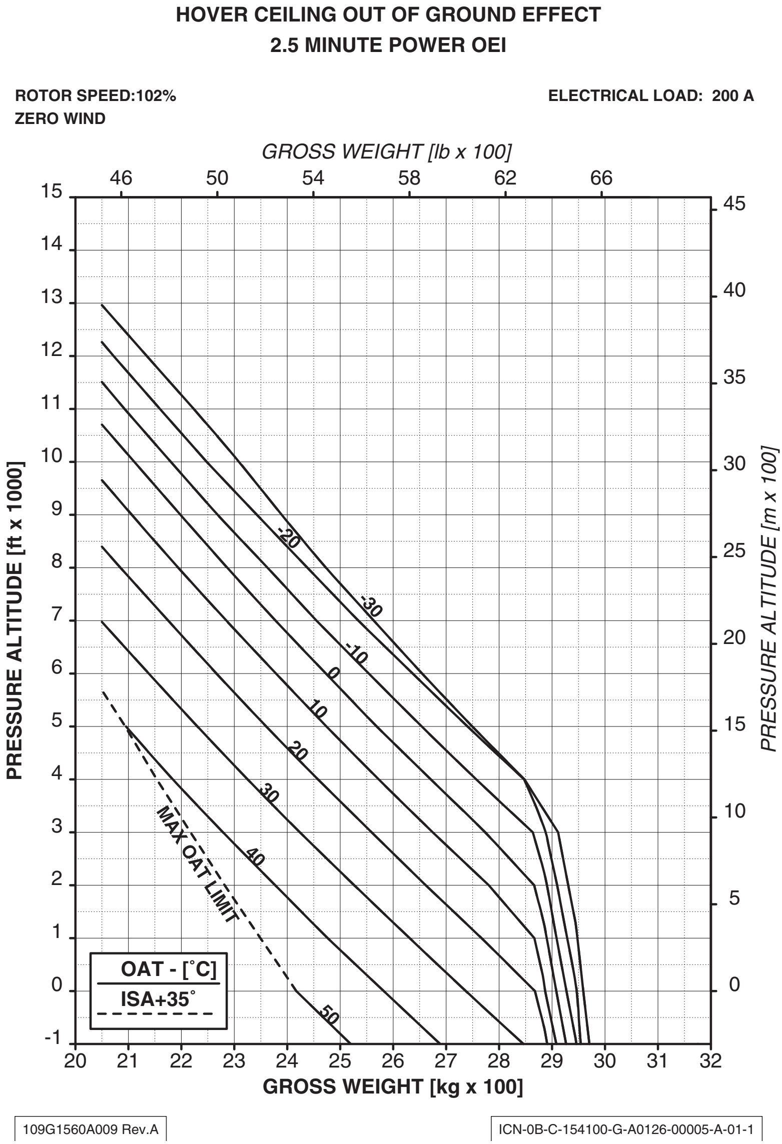


Figure 4-12 Hover Ceiling - OGE OEI, 2.5 Minute Power

HOVER CEILING OUT OF GROUND EFFECT
MAXIMUM CONTINUOUS POWER OEI

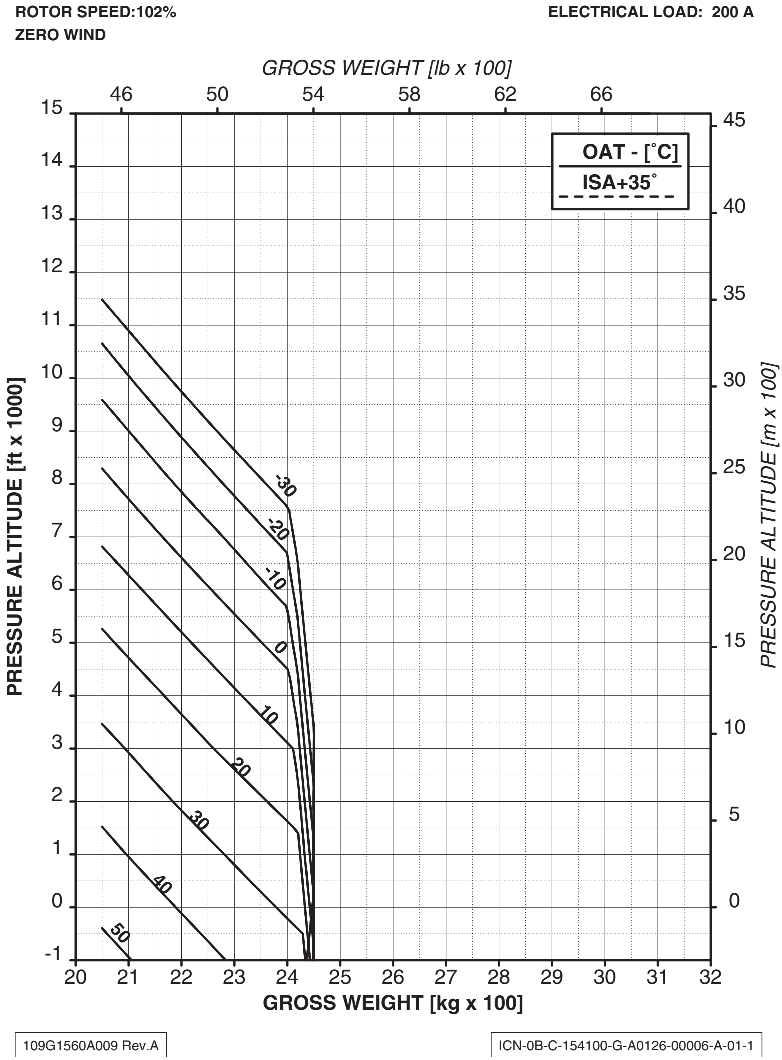


Figure 4-13 Hover Ceiling - OGE at Maximum Continuous Power - OEI

OPERATIONS VS ALLOWABLE WIND

(Figures 4-14 and 4-15)

Note

Before using Chart A and/or Chart B of this paragraph, the appropriate helicopter maximum gross weight for Hover (IGE/OGE) must be determined using the pertinent Hover performance charts.

The “Wind Ground Speed Azimuth Envelope IGE/OGE” charts define, at each altitude/OAT combination, the maximum weight and maximum allowable wind for which satisfactory stability and control has been demonstrated.
Chart A (Figure 4-14) is valid up to a density altitude of 10000 ft Hd.
Maximum relative wind speed in shaded area of Chart A (from 60 to 135 degrees) is determined by Area 2 of Chart B.
Maximum relative wind speed in all azimuths between 10 and 350 degrees, above 10000 ft Hd is determined by Area 1 of Chart B.

Table 4-1 gives a simplified method to know the maximum wind allowed in the critical azimuths at several helicopter weights and density altitude. For more accurate evaluation refer to Figures 4-14 and 4-15.

EXAMPLES

Determine the gross weight at which a specific controllability is guaranteed.

Data 1 (Wind limiting)

Pressure Altitude = 13000 ft

OAT = –10 °C

Wind Speed = 20 kts

Solution 1

Using the hover OGE at Maximum Continuous Power performance chart Figure 4-11, starting on the Pressure Altitude axes from 13000 ft, move to the right to intercept the - 10 °C OAT line, then move down to read the gross weight of 2680 kg.

Enter Chart B (Figure 4-15) with a Pressure Altitude of 13000 ft, move up to intercept the –10 °C OAT line and then to right to intercept the maximum gross weight (2700 kg) line for Hover OGE. The intersection point falls beyond the 20 kts relative wind line: therefore, the helicopter maximum gross weight must be further limited to 2640 kg (intersection point with the 20 kts wind line).

In this case, the gross weight of 2640 kg, guarantees the ability to conduct a Take-off/Landing or hovering operations with relative wind speed of 20 kts from any azimuth and hover OGE at Maximum Continuous Power.

Data 2 (Density Altitude limiting)

Pressure Altitude = 3500 ft

OAT = 40 °C

Wind Speed = 20 kts

Solution 2

Using the hover OGE at Maximum Continuous Power performance chart Figure 4-11, starting on the Pressure Altitude axes from 3500 ft, move to the right to intercept the 40 °C OAT line, then move down to read the gross weight of 2810 kg.

Enter Chart B (Figure 4-15) with a Pressure Altitude of 3500 ft, move up to intercept the 40 °C OAT line, and then to right to intercept the maximum gross weight (2810 kg) line for Hover OGE.
The intersection point gives the maximum allowed relative wind speed (30 kts) from the critical azimuths to conduct a safe Take-off/Landing or hovering operations.
Whenever the intersection point falls within Area 2 of Chart B, refer to Chart A for other azimuths.

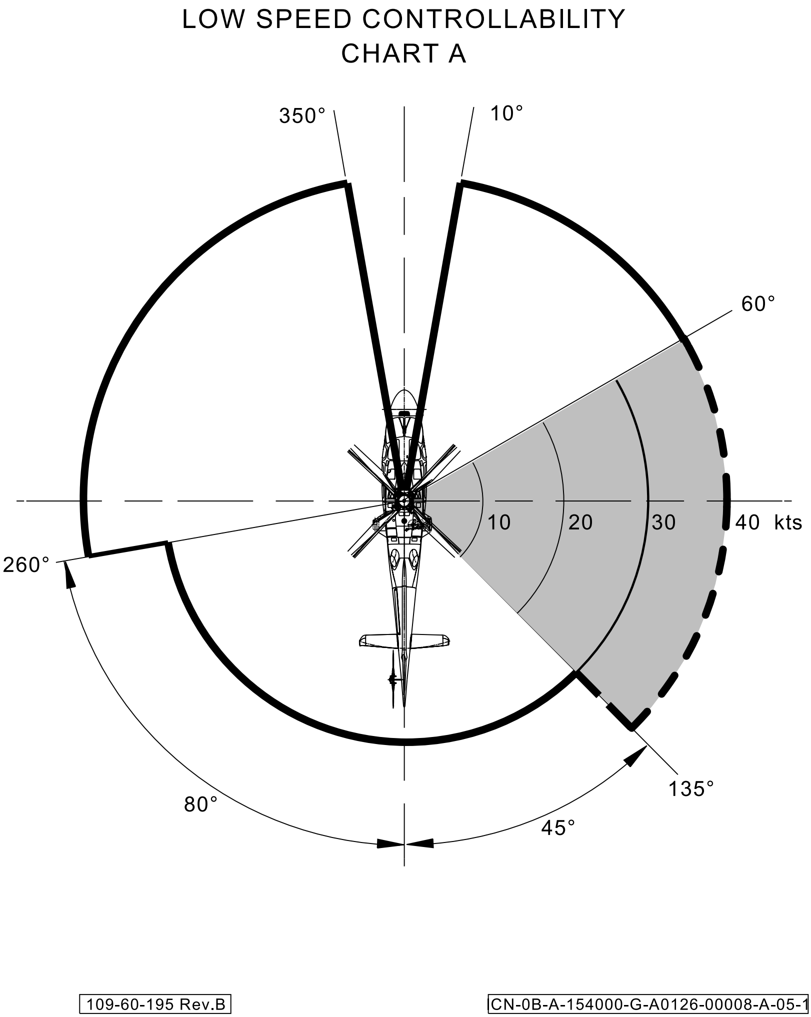
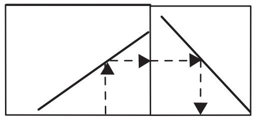


Figure 4-14 Wind/Ground Speed Azimuth Envelope IGE/OGE - Chart A



WEIGHT-ALTITUDE-TEMPERATURE
FOR LOW SPEED CONTROLLABILITY
CHART B

NOTE: ALLOWABLE GROSS WEIGHTS OBTAINED FROM THIS CHART MAY EXCEED CONTINUOUS HOVER CAPABILITY UNDER CERTAIN AMBIENT CONDITIONS. REFER TO HOVER CEILING CHARTS.

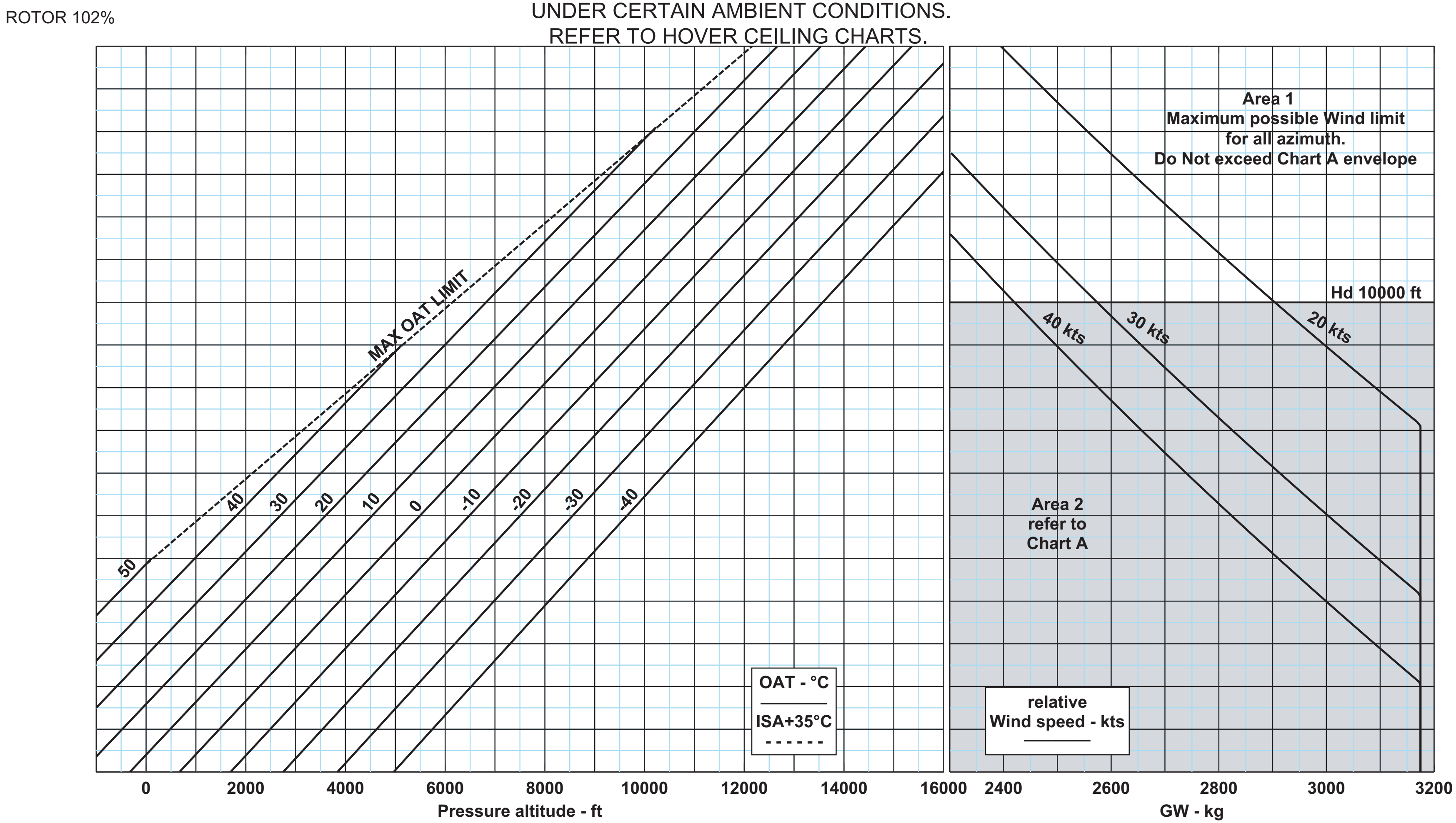


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Figure 4-15 Wind/Ground Speed Azimuth Envelope IGE/OGE - Chart B

Table 4-1. Simplified table

MAXIMUM LATERAL ALLOWABLE WIND ENVELOPE
ALL AZIMUTH (Simplified Table) *

Altitude Hd \ Weight	2400 kg	2600 kg	2800 kg	3000 kg	3175 kg
Up to 3000 ft	30	30	30	30	30
Up to 5000 ft	30	30	30	30	25
Up to 7000 ft	30	30	30	25	20
Up to 9000 ft	30	30	25	20	-
Up to 10000 ft	30	28	20	-	-
Up to 12000 ft	30	20	-	-	-
Up to 14000 ft	25	-	-	-	-
Up to 16000 ft	20	-	-	-	-

*: For a complete Allowable Wind envelope information refer to applicable RFM chart

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HEIGHT-VELOCITY DIAGRAM

(Figures 4-16 to 4-18)

The Height-Velocity diagram is used to establish if, in the event of a single engine failure during take-off, landing or other operations near the surface, a combination of airspeed and height above ground exists for which a safe single engine landing on a smooth, level and hard surface cannot be assured (dangerous zone).

The Height-Velocity diagram is split in three charts (Charts A, B and C).

Chart A

Chart A (Figure 4-16) defines a transfer value, as a function of weight, altitude and temperature, and defines the use of Chart B (Figure 4-17) or Chart C, (Figure 4-18) , if entering, respectively, in Area 1 or Area 2.

Charts B and C

Charts B and C define the shape of the H-V curve. The H-V shape is obtained by joining the high hover point (**HH**), determined by entering the appropriate chart with the pressure altitude, OAT and transfer value, with the fixed points **KN** = 40 ft/30 kts and **LH** = 10 ft/0 kts.
Chart B is applicable to Hd below 7000 ft (Area 1 of Chart A).
Chart C is applicable to Hd from 7000 ft to the altitude limit, which is 14000 Hp or 14000 Hd whichever comes first. (Area 2 of Chart A).

Yellow Area of Chart B and C

Combination of weight/altitude/temperature at which the shape of the H-V curve is fixed by **LH** = 10 ft/0 kts, **KN** = 40 ft/30 kts, $\mathbf{HH_1}$ = 80 ft/0 kts.

NO H-V AREA (Green area of Chart B and C)

Combination of weight/altitude/temperature at which no H-V curve exists.

Note

The height-velocity diagram does not define the conditions which assure continued flight following an engine failure nor the conditions from which a safe power-off landing can be made.

EXAMPLES:

H-V Curve Determination

Data 1

Pressure Altitude = 2000 ft

OAT = 20 °C

GW = 2600 kg

Solution 1

Chart A: Enter chart with Pressure Altitude of 2000 ft and move up to intersect the 20 °C OAT line.
Move to the right to intersect the 2600 kg GW line, then move down to read a transfer value of 2.75.

Chart B: Enter chart with Pressure Altitude of 2000 ft, move up to intersect the 20 °C OAT line and then move right to intersect the 2.75 transfer value line. Since the resulting point falls in the green area, the H-V curve does not exist for this flight condition.

Data 2

Pressure Altitude = 5000 ft

OAT = 40 °C

GW = 3175 kg

Solution 2

Chart A: Enter chart with Pressure Altitude of 5000 ft and move up to intersect the 40 °C OAT line.
Move right to intersect the 3175 kg GW line, then move down to read a transfer value of 4.

Chart C: Enter chart with Pressure Altitude of 5000 ft, move up to intersect the 40 °C OAT line, move right to intersect the transfer value line of 4. Move up to read a height above ground of 205 ft (high hover point, point **HH**). Drawing a line between this point **HH** and point **KN,** and joining it with the **KN-LN** line, the H-V curve for this flight condition is obtained.

FLY-AWAY MANOEUVRE

Fly-away is the manoeuvre that, in case of an engine failure occurring in hovering out of ground effect at **High Hover point (HH) plus 20 ft,** allows enough height to gain a suitable speed for a single engine flight condition with **a ground clearance of 10 ft.**

Fly-away Height Calculation

Whenever the Height-Velocity diagram exists, it is possible to successfully carry out a fly-away manoeuvre by adding 20 ft to the High Hover point (HH) of the H-V curve.
In all other conditions with NO H-V AREA, it is possible to complete a fly-away manoeuvre from a minimum height of 80 ft.

EXAMPLE:

Fly-away Minimum Height Calculation

Data

Pressure Altitude = 5000 ft

OAT = 40 °C

GW = 3175 kg

Solution

Solution 2 of the examples of H-V Curve Determination on the previous page, defines, for the desired flight condition, a high hover (**HH**) point of 205 ft.
Adding 20 ft to **HH** (205 ft), the required minimum height to assure a fly-away manoeuvre is found to be 225 ft.

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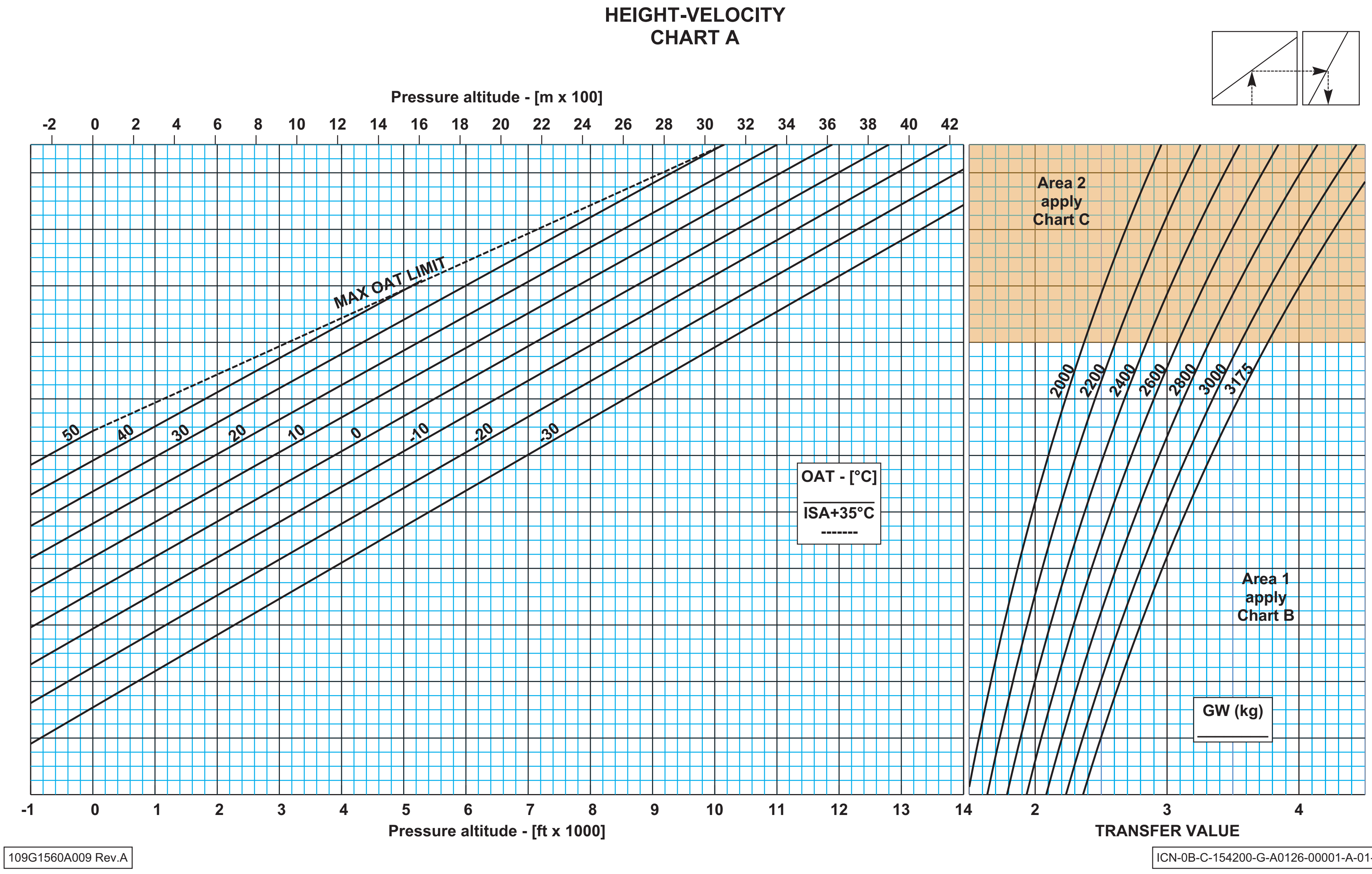

Figure 4-16 Height - Velocity Diagram - Chart A

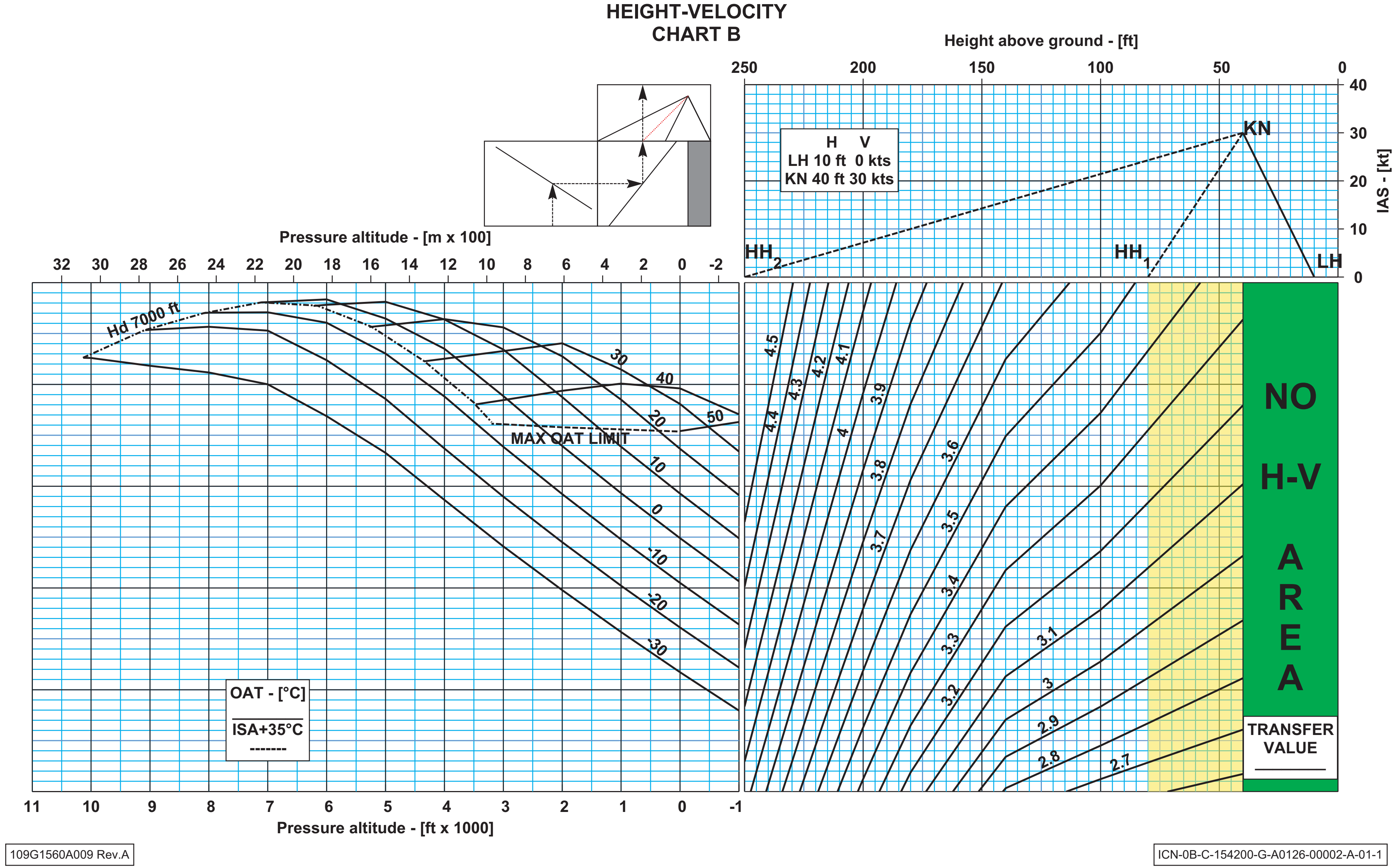

Figure 4-17 Height - Velocity Diagram - Chart B

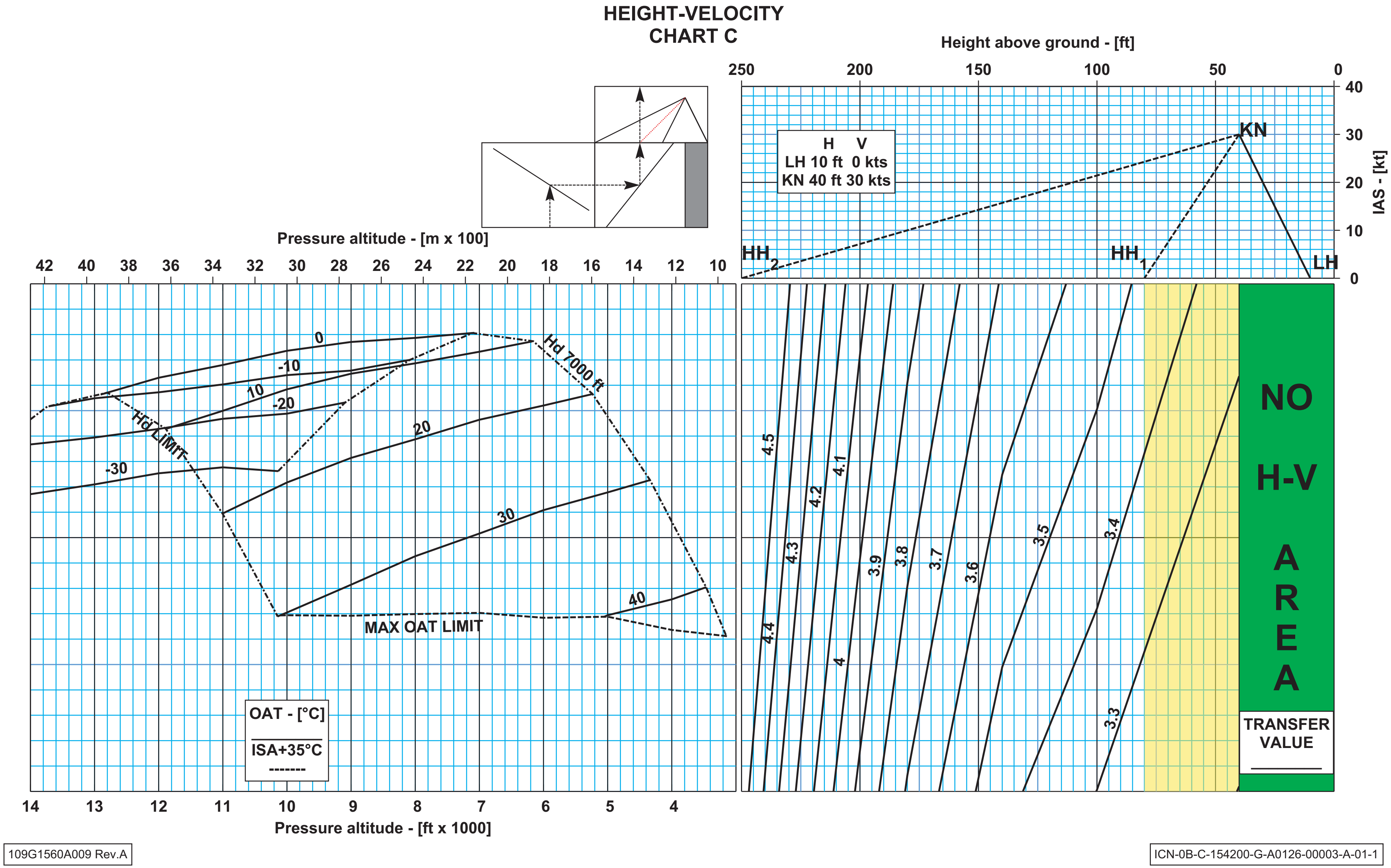

Figure 4-18 Height - Velocity Diagram - Chart C

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RATE OF CLIMB

(Figures 4-19 to 4-34)

The climb performance shown are based on flight test results and cover the range of gross weights from 2300 kg to the Maximum Gross Weight of 3175 kg. Rate of Climb (ROC) is given for the All Engines Operating (AEO) case and for the One Engine Inoperative (OEI) case.

For the AEO case, rate of climb is given with the engines operating at the Take-Off Power (TOP) rating limit and at the Maximum Continuous Power (MCP) rating limits.

For the OEI case, rate of climb is given with the engine operating at the 2.5 minute Power rating limit and at the Maximum Continuous Power rating limits. The datum speed for all Rate of Climb graphs is Vy, which is 80 KIAS up to 10000 ft and then decreases by 2 knots every 1000 ft to become 60 KIAS at 20000 ft.

Each graph presents the information for a specific Gross Weight. On each graph, additional information is provided relative to the applicable engine power rating. Care should be taken to ensure that the correct graph is utilized in obtaining Rate of Climb data.

Examples

Example Number 1:

Determine the All Engine Operating (AEO) Rate of Climb at a Gross Weight of 2600 kg. The engines are operated at the TOP rating limits. The Pressure Altitude is 14000 ft and the OAT is -10 °C.

Solution:

Figure 4-20 presents the required information.
Enter the graph at a Pressure Altitude of 14000 ft and proceed to the right until intersecting the -10 °C curve. From this point, drop vertically to the horizontal scale and read the Rate of Climb of 1870 ft/min.

Example Number 2:

Determine the One Engine Inoperative (OEI) Rate of Climb at a Gross Weight of 2600 kg. The live engine is operating at the 2.5 minute Power rating limit. The Pressure Altitude is 9000 ft and the OAT is +5 °C.

Solution:

Figure 4-28 presents the required information.
Enter the graph at a Pressure Altitude of 9000 ft and proceed to the right and interpolate between the 0 °C and +10 °C curves to find the point at +5 °C. From this point, drop vertically to the horizontal scale and read the Rate of Climb of 850 ft/min.

Example Number 3:

Determine the All Engine Operating (AEO) Rate of Climb at an OAT of +10 °C, at an helicopter gross weight of 2750 kg and a Pressure Altitude of 10000 ft.

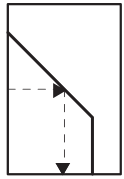
The engines are operated at the MCP rating limit.

Solution:

From Figure 4-24, using the previous procedure, the Rate of Climb at 2600 kg is 1520 ft/min and from Figure 4-25 the Rate of Climb at 2900 kg is 1150 ft/min.
The Rate of Climb at 2750 kg is therefore:

$$\Delta ROC = \frac{GW_{ACTUAL} - GW_{2600}}{GW_{2900} - GW_{2600}} \times (ROC_{2900} - ROC_{2600}) = \frac{150}{300} \times (1150 - 1520) = -185$$

$$ROC_{2750} = ROC_{2600} + \Delta ROC = 1520 - 185 = 1335 \text{ ft/min.}$$



RATE OF CLIMB
TAKE-OFF POWER AEO

ROTOR SPEED: 100%
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

ELECTRICAL LOAD: 200 A TOTAL

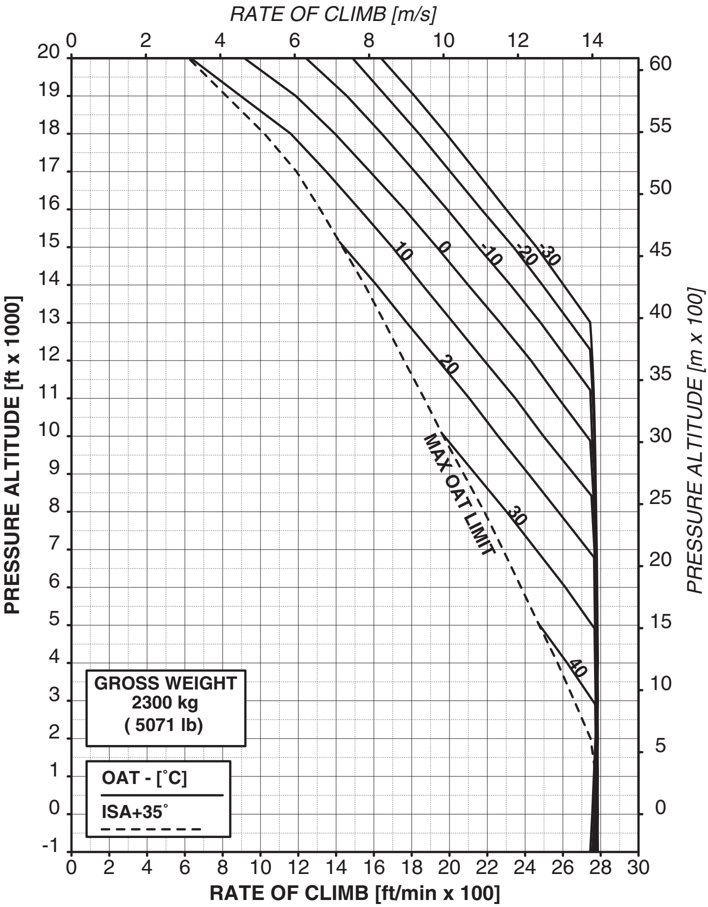


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Figure 4-19 Rate of Climb at Take-Off Power - AEO - Gross Weight 2300 kg

RATE OF CLIMB
TAKE-OFF POWER AEO

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A TOTAL**

80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

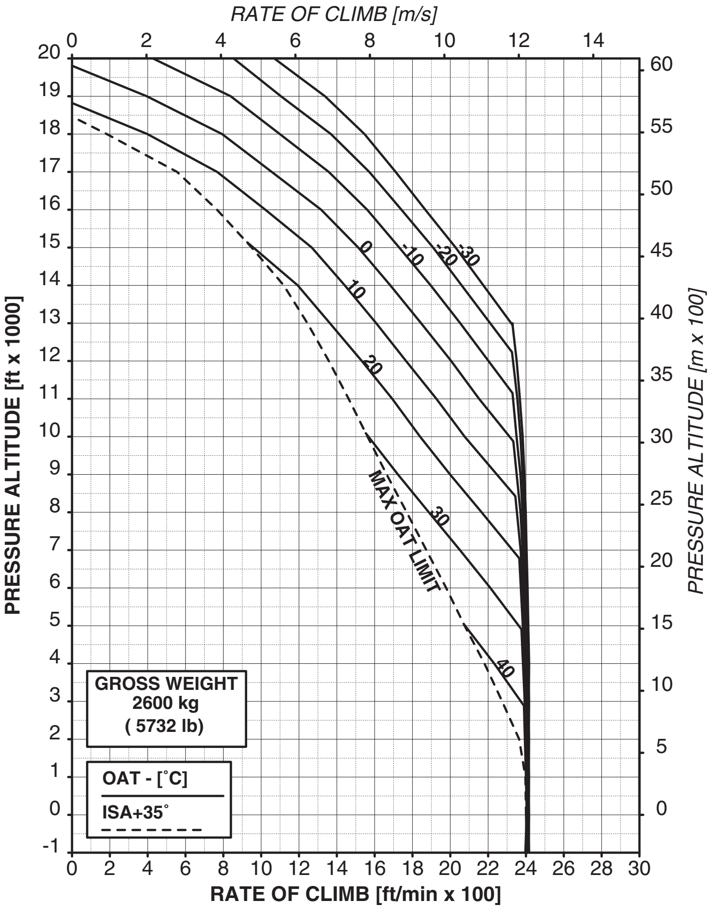

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Figure 4-20 Rate of Climb at Take-Off Power - AEO - Gross Weight 2600 kg

RATE OF CLIMB
TAKE-OFF POWER AEO

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A TOTAL**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

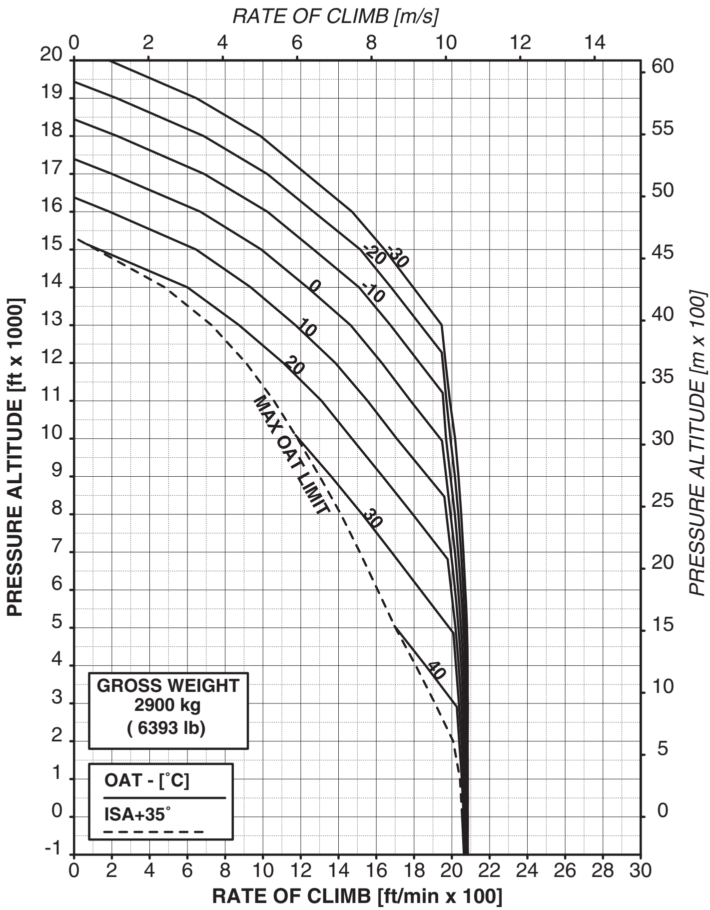

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Figure 4-21 Rate of Climb at Take-Off Power - AEO - Gross Weight 2900 kg

RATE OF CLIMB
TAKE-OFF POWER AEO

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A TOTAL**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

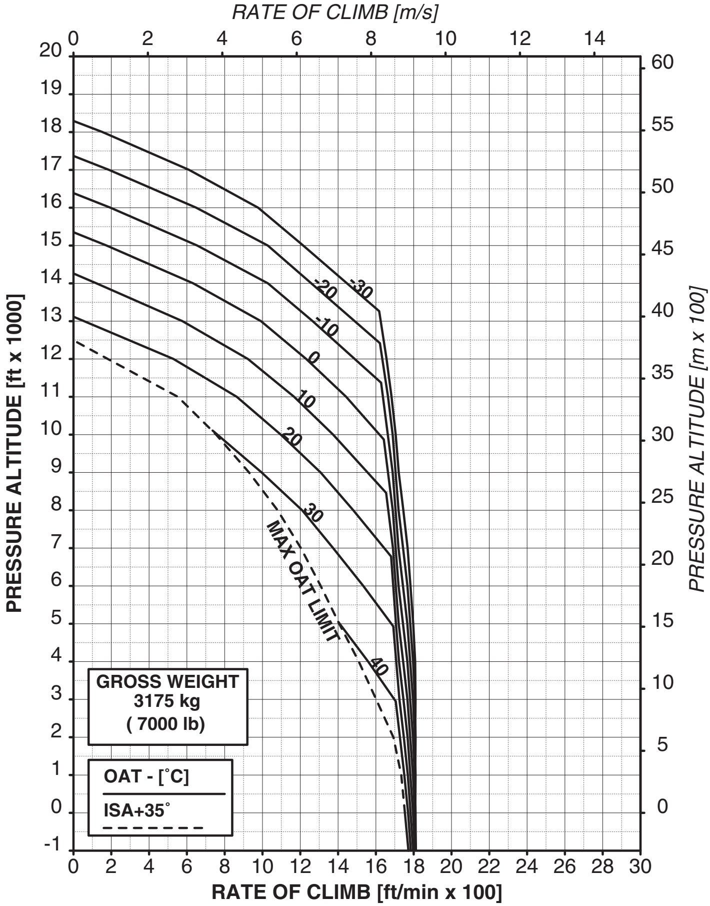

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Figure 4-22 Rate of Climb at Take-Off Power - AEO - Gross Weight 3175 kg

RATE OF CLIMB
MAXIMUM CONTINUOUS POWER AEO

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A TOTAL**

80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

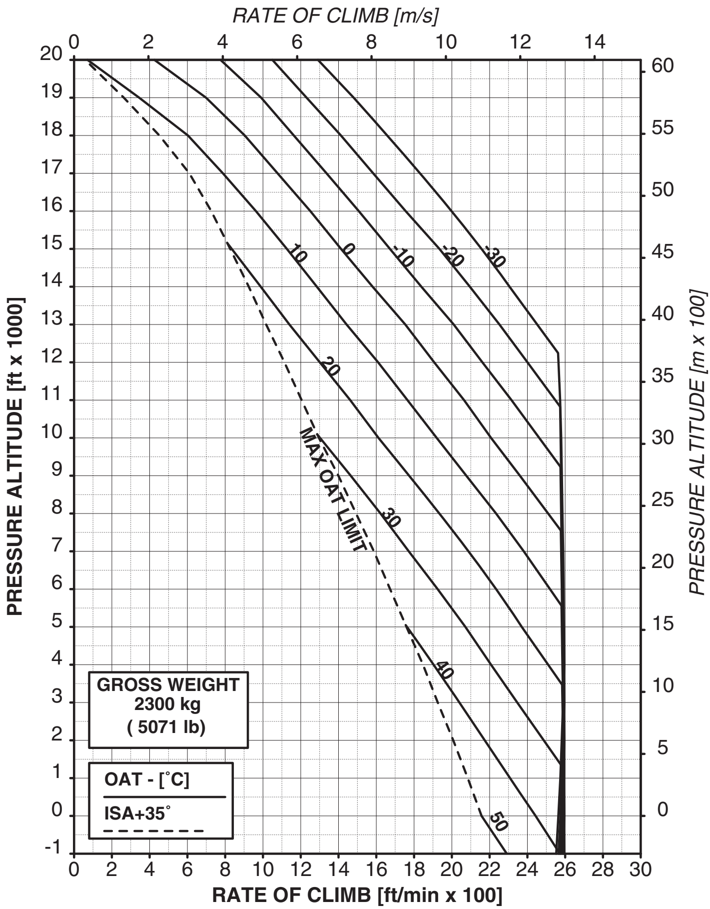

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Figure 4-23 Rate of Climb at Maximum Continuous Power - AEO - Gross Weight 2300 kg

RATE OF CLIMB
MAXIMUM CONTINUOUS POWER AEO

ROTOR SPEED: 100% ELECTRICAL LOAD: 200 A TOTAL
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

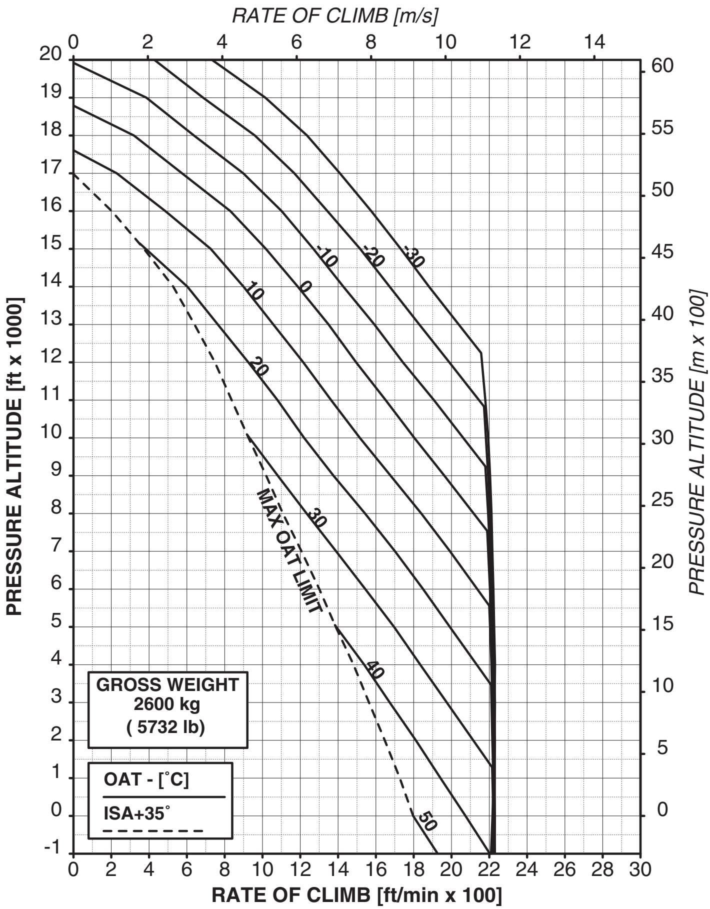

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Figure 4-24 Rate of Climb at Maximum Continuous Power - AEO - Gross Weight 2600 kg

RATE OF CLIMB
MAXIMUM CONTINUOUS POWER AEO

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A TOTAL**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

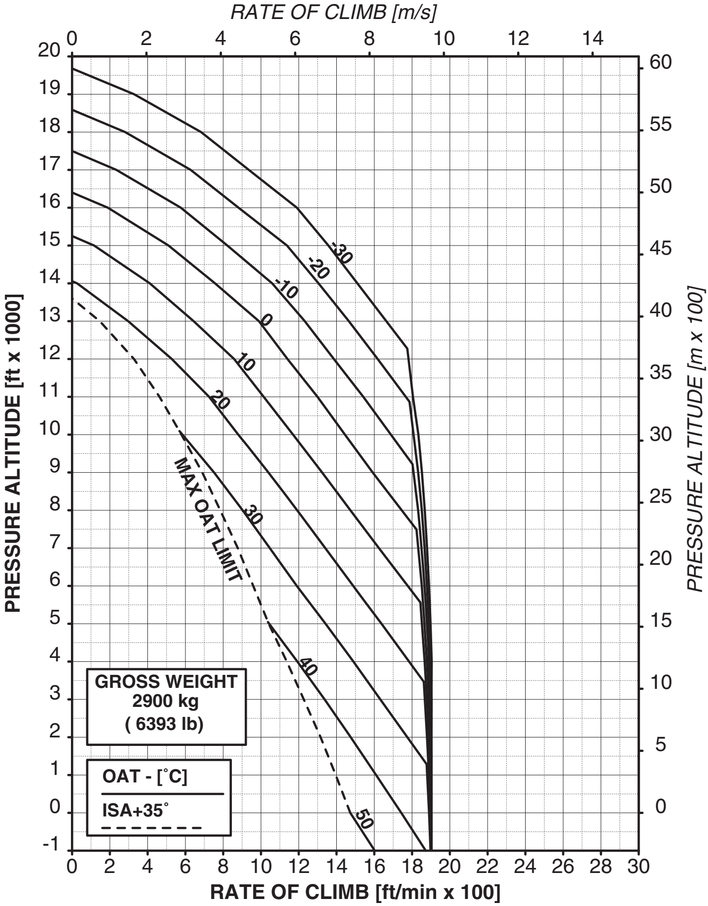

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Figure 4-25 Rate of Climb at Maximum Continuous Power - AEO - Gross Weight 2900 kg

RATE OF CLIMB
MAXIMUM CONTINUOUS POWER AEO

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A TOTAL**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

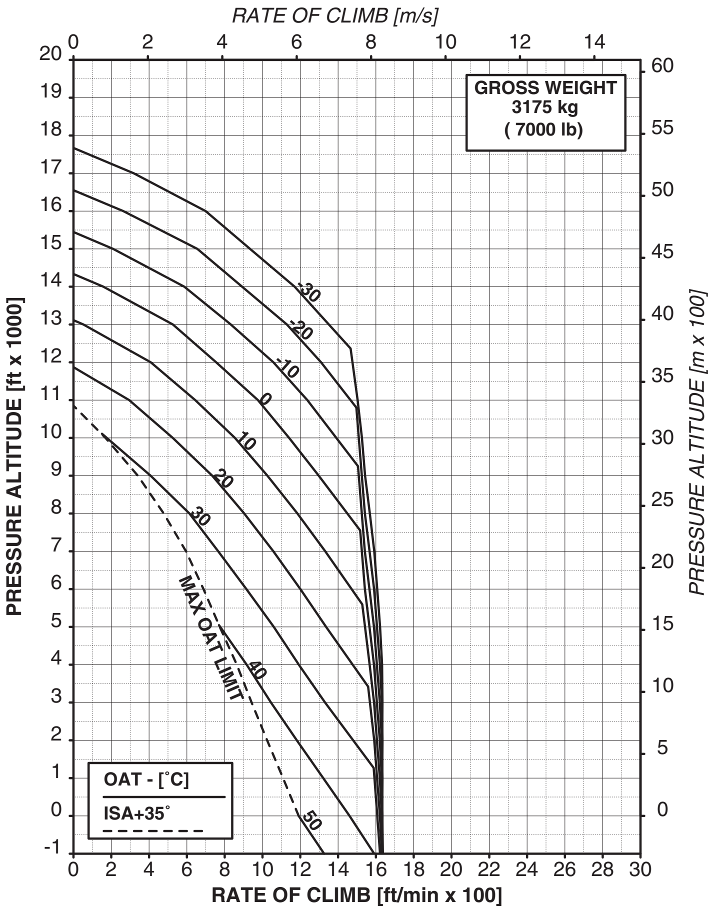

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Figure 4-26 Rate of Climb at Maximum Continuous Power - AEO - Gross Weight 3175 kg

RATE OF CLIMB
2.5 MINUTE POWER OEI

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

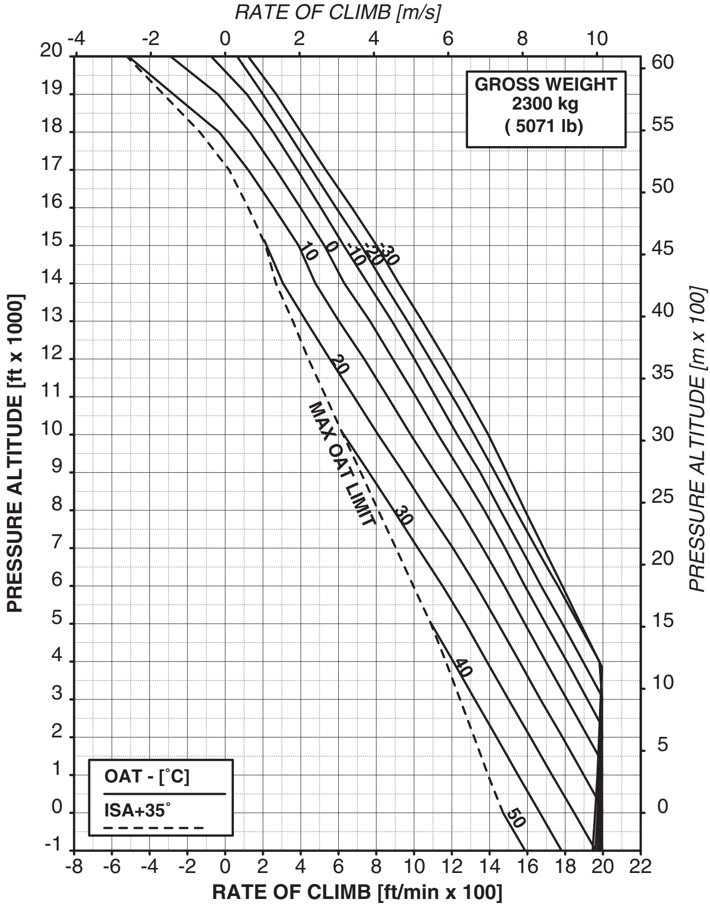

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Figure 4-27 Rate of Climb at 2.5 Minute Power - OEI - Gross Weight 2300 kg

RATE OF CLIMB
2.5 MINUTE POWER OEI

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

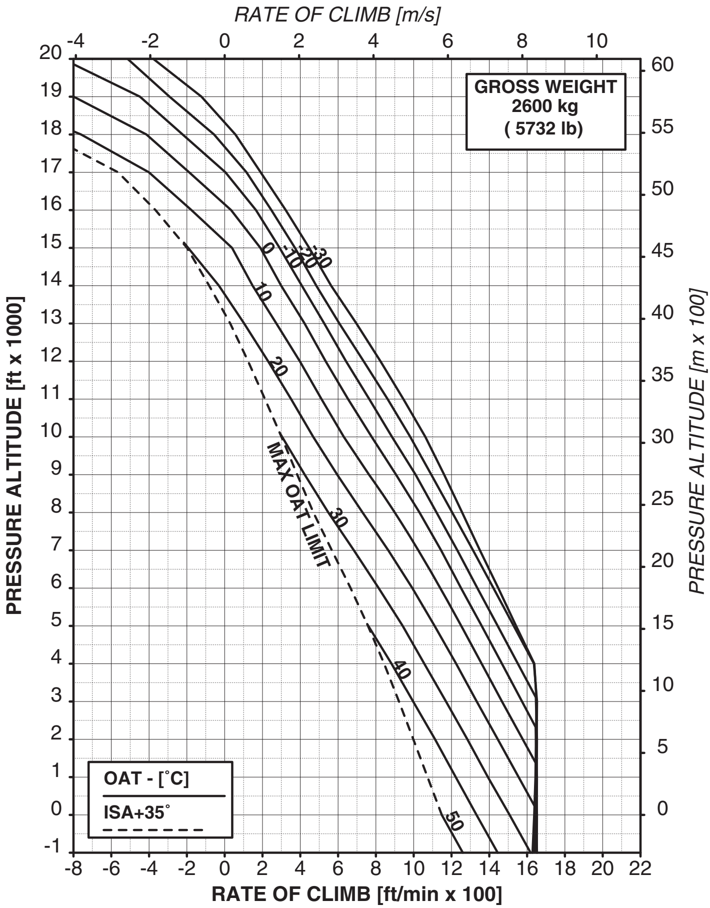

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Figure 4-28 Rate of Climb at 2.5 Minute Power - OEI - Gross Weight 2600 kg

RATE OF CLIMB
2.5 MINUTE POWER OEI

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A**

80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

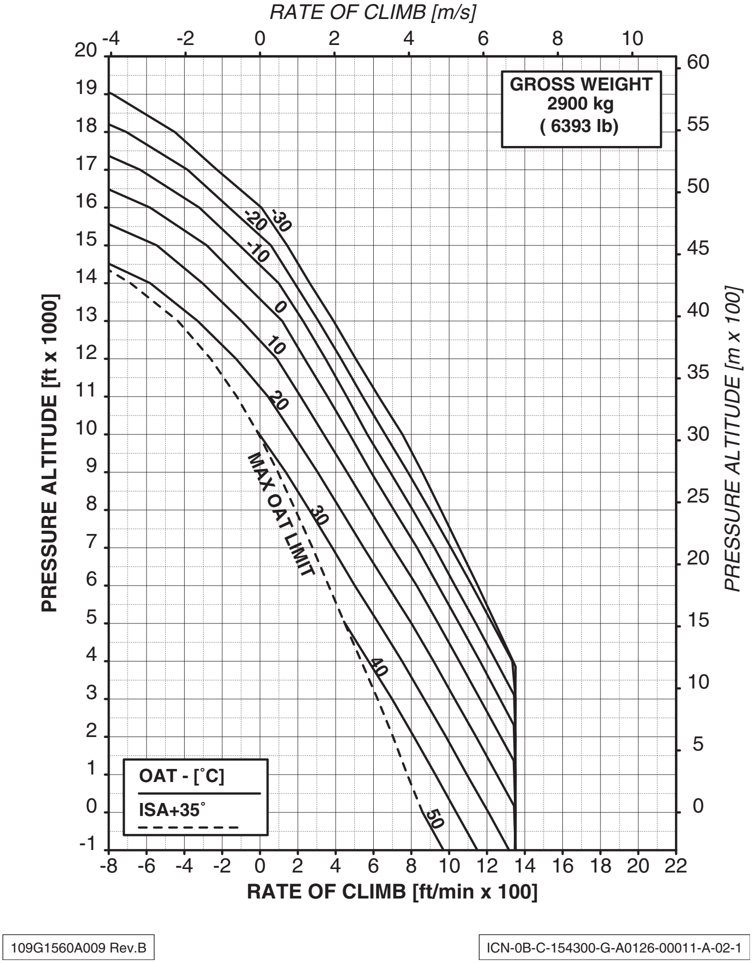

Figure 4-29 Rate of Climb at 2.5 Minute Power - OEI - Gross Weight 2900 kg

RATE OF CLIMB
2.5 MINUTE POWER OEI

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

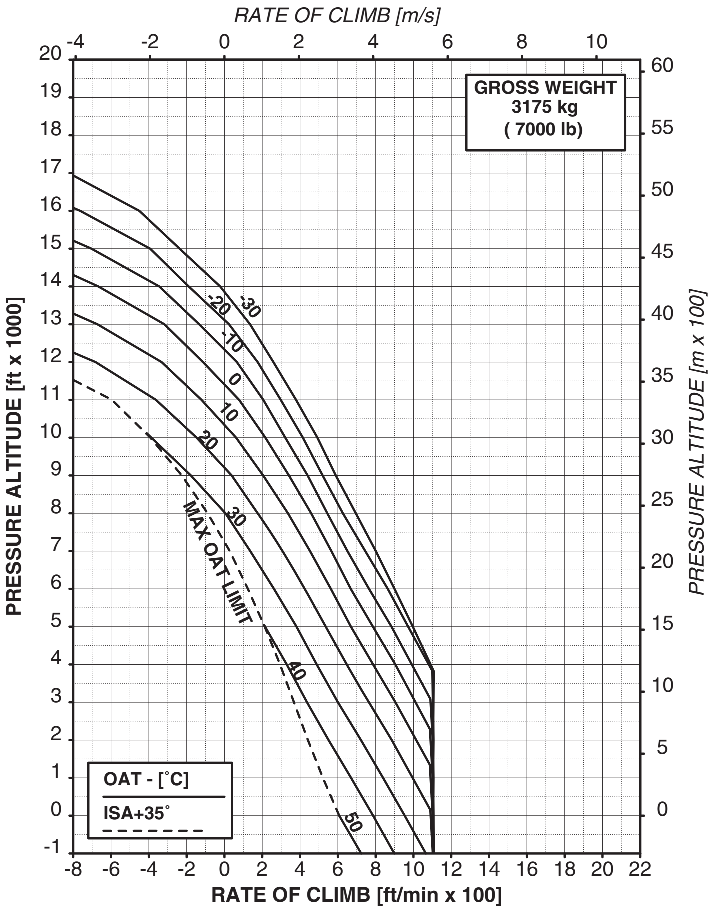

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Figure 4-30 Rate of Climb at 2.5 Minute Power - OEI - Gross Weight 3175 kg

RATE OF CLIMB
MAXIMUM CONTINUOUS POWER OEI

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

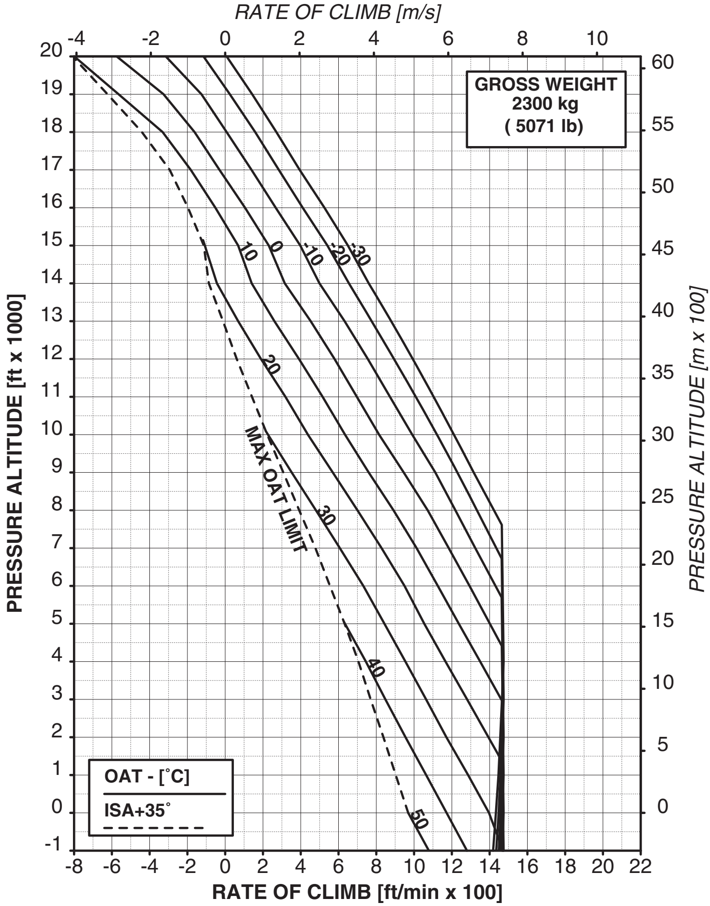

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Figure 4-31 Rate of Climb at Maximum Continuous Power - OEI - Gross Weight 2300 kg

RATE OF CLIMB
MAXIMUM CONTINUOUS POWER OEI

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A**

80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

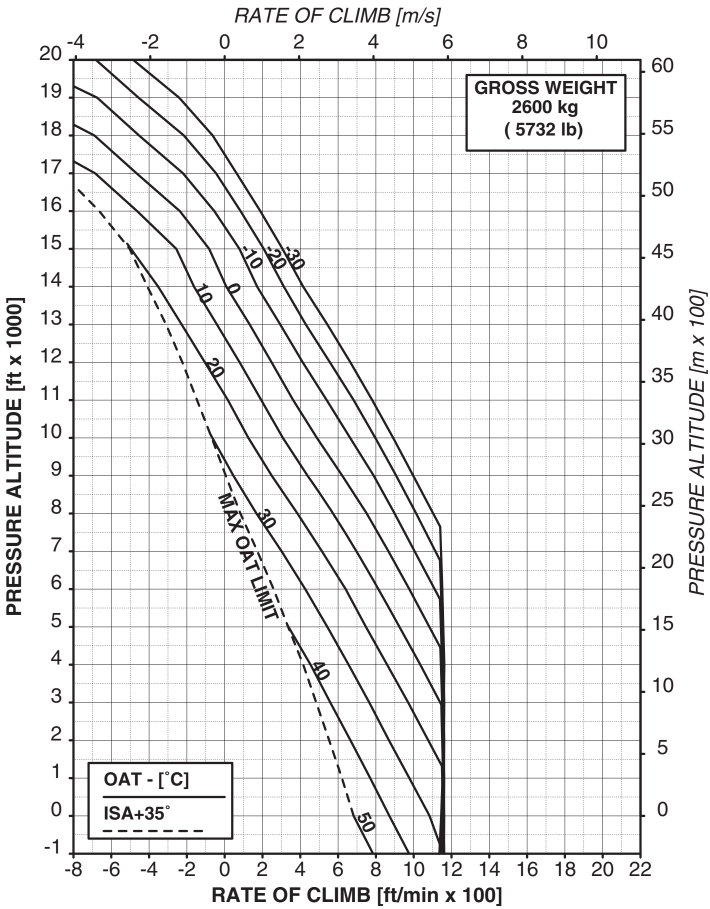

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Figure 4-32 Rate of Climb at Maximum Continuous Power - OEI - Gross Weight 2600 kg

RATE OF CLIMB
MAXIMUM CONTINUOUS POWER OEI

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

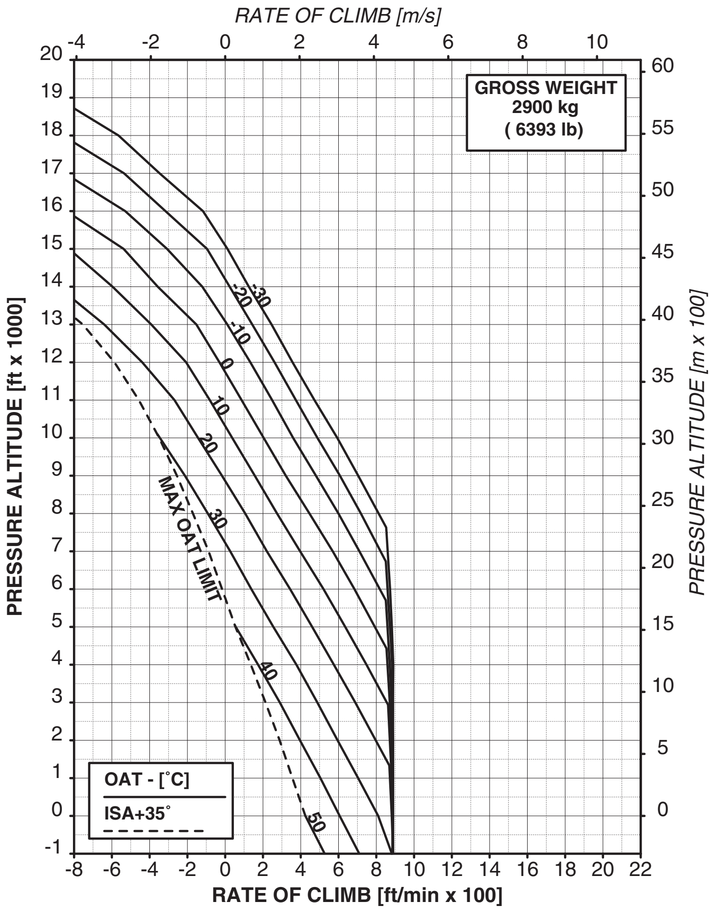

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Figure 4-33 Rate of Climb at Maximum Continuous Power - OEI - Gross Weight 2900 kg

RATE OF CLIMB
MAXIMUM CONTINUOUS POWER OEI

ROTOR SPEED: 100% **ELECTRICAL LOAD: 200 A**
80 KIAS DECREASE 2 kt EACH 1000 ft ABOVE Hp 10000 ft

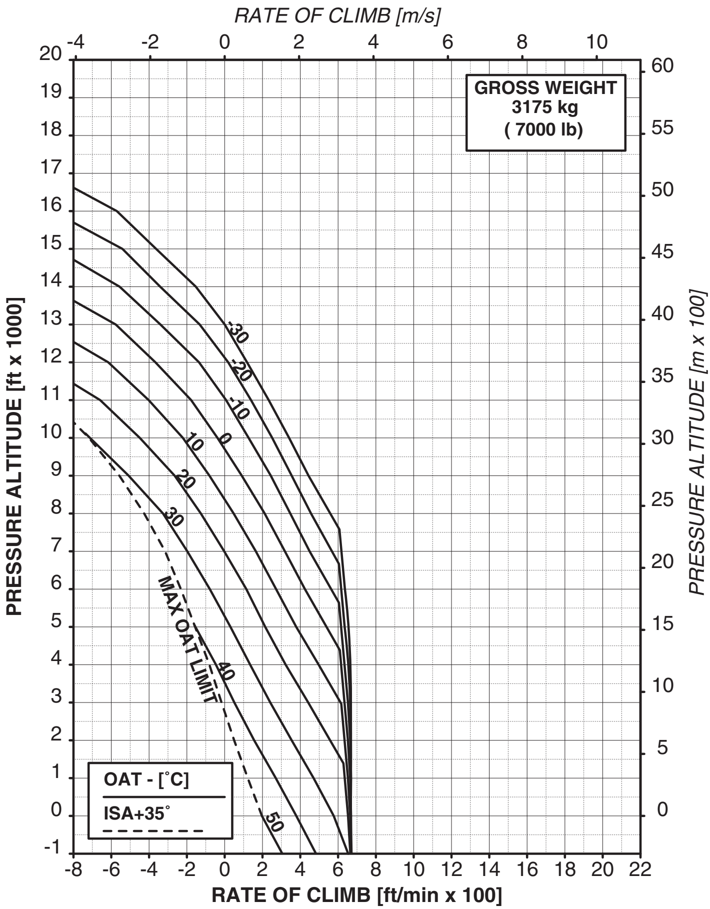

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Figure 4-34 Rate of Climb at Maximum Continuous Power - OEI - Gross Weight 3175 kg

NOISE CHARACTERISTICS

The following noise levels comply with ICAO Annex 16, Chapter 8, 4th Edition:

Model: AW109SP Engine: Pratt & Whitney 207C Maximum Gross Weight: 3175 kg			
Configuration	**Level Fly-over EPNL (EPNdB)**	**Take-off EPNL (EPNdB)**	**Approach EPNL (EPNdB)**
Clean aircraft. No external kit installed.	88.4	90.5	91.2

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SECTION 5

OPTIONAL EQUIPMENT SUPPLEMENTS

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SECTION 5

OPTIONAL EQUIPMENT SUPPLEMENTS

GENERAL

The Supplements contained in the Document No. 109G0040A019 and relative to the optional equipments installed on the helicopter shall be attached to the present Rotorcraft Flight Manual.

Note

Refer to Rotorcraft Flight Manual AW109SP - Optional Equipment Supplements, Document No. 109G0040A019, for the approved list of applicable Supplements.

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PART II - MANUFACTURER’S DATA

LIST OF REVISIONS

REVISION NO.	SUBJECT	DATE
–	Issue A	25 May 2009
–	Issue B	11 December 2009
1	Not affected	–
2	Revised pages C-1, 7-i thru 7-xviii, 7-159, 7-163, 7-164, 7-174, 7-223, 7-228 thru 7-234, 7-240 thru 7-244, 7-255, 7-258, 7-320, 7-332 thru 7-334, 7-340, 7-353, 7-362, 7-421 and 7-424. Added pages 7-164A, 7164B, 7-190A, 7-190B, 7-202A thru 7-202D, 7-234A thru 7-234D, 7-283A thru 7-283J, 7-340A, 7-340B, 7-364A, 7-364B and 7-425 thru 7-428.	26 July 2010
3	Revised pages C-1, 7-i thru 7-xviii, 7-203, 7-219 thru 7-221, 7-239 and 7-338 thru 7-340. Added pages 7-78A thru 7-78H.	8 November 2010
4	Revised pages C-1, 7-i thru 7-xviii, 7-31, 7-33 thru 7-38, 7-55, 7-78H, 7-93, 7-234D, 7-284, 7-287, 7-294, 7-295, 7-297, 7-301, 7-308, 7-309, 7-314, 7-316, 7-321 and 7-322. Added pages 7-78I thru 7-78P, 7-124A thru 7-124D, 7-234E thru 7-234H and 7-400A thru 7-400F.	11 February 2011

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SECTION 6

WEIGHT AND BALANCE

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SECTION 6

WEIGHT AND BALANCE

Note

In accordance with EASA procedures, the detail weight and balance data of this section are not subject to EASA approval.
The loading instructions of this section, however, have been accepted by EASA as satisfying all requirements for instructions on loading of the rotorcraft within approved limits of weight and center of gravity, and on maintaining the loading within such limits.

GENERAL

This section provides information for the weight and balance computation of the AW109SP helicopter.

It is the pilot's responsibility to ensure that the helicopter is properly loaded so as to maintain for the duration of the entire flight the Center of Gravity (CG) within the limitations defined in Paragraph WEIGHT AND CENTER OF GRAVITY of SECTION 1 - LIMITATIONS.

WARNING

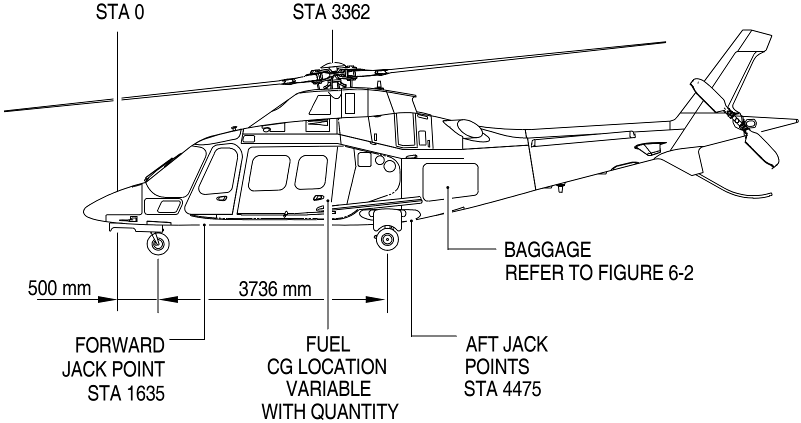
Operation outside of prescribed weight and balance limitations could result in an accident and serious or fatal injury.

Figures, charts and examples are provided to assist the pilot in computing the proper loading conditions.

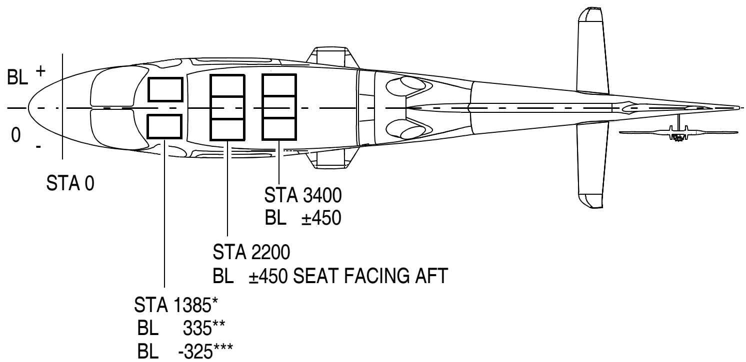
DEFINITIONS

Datum Lines	All measurements required to calculate weight and balance are made with respect to datum lines established by the manufacturer. Two datum lines are utilized, a longitudinal datum line and a lateral datum line.
Longitudinal Datum Line	The longitudinal datum line (Station 0) is located 1635 mm forward of the front jackpoint.
Lateral Datum Line	The lateral datum line (butt line 0) is the line extending down the centerline of the helicopter (helicopter viewed from the top) measurements made to the starboard side of the helicopter are positive numbers. Measurements made to the port side of the helicopter are negative.
Arm	Arm is the term used for the measured value, in mm, from a datum line to a particular item of interest.
Moment	Moment is the product of the weight of an item multiplied by its arm. Moments are expressed in units of kg mm.
Longitudinal Center of Gravity	The longitudinal position of the mass balance point of the helicopter, expressed as an arm measured from the longitudinal datum line.
Lateral Center of Gravity	The lateral position of the mass balance point of the helicopter, expressed as an arm measured from the lateral datum line.
Optional Equipment	Optional equipment is that equipment which is installed to configure the helicopter for a particular configuration.
Basic Weight	The Basic Weight is the empty weight (equipped) less engine oil and special mission equipment. It includes the trapped and unusable fuel and oil.
Empty Weight (equipped)	Weight of aircraft complete with all systems as configured in accordance with the model detail specification with full operating fluids, including engine oil. The helicopter is prepared for service.
Operating Weight	The Operating Weight consists of the equipped empty weight plus crew weight.

Dry Weight	The Dry Weight consists of the Operating Weight plus Payload (passengers, baggage/cargo).
Gross Weight	The Gross Weight consists of the Dry Weight plus take-off or landing fuel.
Maximum Gross Weight or Maximum Take Off Weight	The Maximum Gross Weight is determined by performance considerations and must be calculated considering expected mission variables (see Chart E - Weight & Balance Computation Form).



STANDARD VERSION
PILOTS AND PASSENGERS STATION



* ADJUSTABLE FROM 1328 TO 1404 mm
** PILOT
*** COPILOT / PASSENGER

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Figure 6-1 Helicopter Stations Diagram

DOORS OPEN OR REMOVED

Opening or removing doors results in center of gravity changes. Door removal or opening is limited to passenger doors.

LONGITUDINAL MOMENTS

The table below lists weight and longitudinal moment adjustment which shall be made in determining the gross weight and longitudinal CG when doors are opened or removed.

Door configuration	Weight change (kg)	STA (mm)	Moment change (kg mm)
Right sliding door removed	-15.6	2647	-41293.2
Left sliding door removed	-15.6	2647	-41293.2
Right sliding door open	–	4026	21512.4
Left sliding door open	–	4026	21512.4

LATERAL MOMENTS

The table below lists weight and lateral moment adjustment which shall be made in determining the gross weight and lateral CG when doors are removed.

Door configuration	Weight change (kg)	BL (mm)	Moment change (kg mm)
Right sliding door removed	-15.6	752	-11731.2
Left sliding door removed	-15.6	-752	11731.2

WEIGHTS, ARMS AND MOMENTS

LONGITUDINAL MOMENTS

PILOTS AND PASSENGERS			
Weight (kg)	Pilot Copilot or Passenger (STA 1385 mm)(*) Moment (kg mm)	Passengers 3 places central seat aft facing (STA 2200 mm) Moment (kg mm)	Passengers 3 places aft seat (STA 3400 mm) Moment (kg mm)
60	83100	132000	204000
65	90025	143000	221000
70	96950	154000	238000
75	103875	165000	255000
80	110800	176000	272000
85	117725	187000	289000
90	124650	198000	306000
95	131575	209000	323000
100	138500	220000	340000
120	166200	264000	408000
140	193900	308000	476000
160	221600	352000	544000
180	249300	396000	612000
200	277000	440000	680000
220		484000	748000
240		528000	816000
260		572000	884000
280		616000	952000
300		660000	1020000
320		704000	1088000
(*) Adjustable from 1328 to 1404 mm			

USABLE FUEL - MAIN FUEL TANKS			
Weight (kg)	**Capacity (l) (0.8 kg/l)**	**STA (mm)**	**Moment (kg mm)**
20	25	3761	75220
40	50	3765	150580
60	75	3771	226260
80	100	3779	302320
100	125	3732	373200
120	150	3663	439560
140	175	3613	505820
160	200	3578	572480
180	225	3561	640980
200	250	3599	719800
220	275	3657	804540
240	300	3705	889200
260	325	3745	973700
280	350	3780	1058400
300	375	3810	1143000
320	400	3837	1227840
340	425	3860	1312400
360	450	3881	1397160
380	475	3899	1481620
400	500	3916	1566400
420	525	3931	1651020
440	550	3945	1735800
449.6	562	3951	1776370

UNUSABLE FUEL - MAIN FUEL TANKS			
Weight (kg)	**Capacity (l) (0.8 kg/l)**	**STA (mm)**	**Moment (kg mm)**
10	12.5	3761	37610

ENGINE OIL (STA 4280 mm)		
Weight (kg)	**Capacity (l)**	**Moment (kg mm)**
10	10.52	42800

UNDRAINABLE ENGINE OIL (STA 4280 mm)		
Weight (kg)	**Capacity (l)**	**Moment (kg mm)**
1.8	1.9	7704

MAIN TRANSMISSION OIL (STA 3602 mm)		
Weight (kg)	**Capacity (l)**	**Moment (kg mm)**
10.5	11	37821

LATERAL ARMS

Item	Butt line (BL) (mm)
Pilot	+ 335
Copilot	– 325
Passengers	See Figure 6-1
Baggage	0

ALLOWABLE BAGGAGE LOAD

The baggage compartment is divided in five zones. In Figure 6-2, the maximum baggage load for each zone is presented.

The maximum allowable baggage load is 120 kg.

The following table defines the longitudinal baggage moments for each zone.
The maximum allowable longitudinal baggage moment is 585600 kg mm.
An example of baggage loading is provided in Figure 6-2.
The total baggage longitudinal moment shall be less than 585600 kg mm.

Baggage load (kg)	ZONE 1 STA 4880 mm	ZONE 2 STA 5240 mm	ZONE 3 STA 5560 mm	ZONE 4 STA 5960 mm	ZONE 5 STA 6430 mm
	Baggage moment (kg mm)				
10	48800	52400	55600	59600	64300
20	97600	104800	111200	119200	128600
30	146400	157200	166800	178800	192900
40	195200	209600	222400	238400	257200
50	244000	262000	278000	298000	321500
60	292800	314400	333600	357600	385800
70	341600	366800	389200	417200	450100
75	366000	393000	417000	447000	482250
78	380640	408720	433680	464880	501540
80	390400	419200	444800	476800	514400
84	409920	440160	467040	500640	540120
85	414800	445400	472600	506600	
90	439200	471600	500400	536400	
93	453840	487320	517080	554280	
100	488000	524000	556000		
102	497760	534480	567120		
105	512400	550200			
110	536800	576400			
120	585600				

ZONE	WEIGHT (kg)	MOMENT (kg mm)
1	50	244000
2	40	209600
3	23	127880
4	—	—
5	—	—
TOTAL	113	581480

The total baggage load and moment calculated above are less than the maximum allowable.

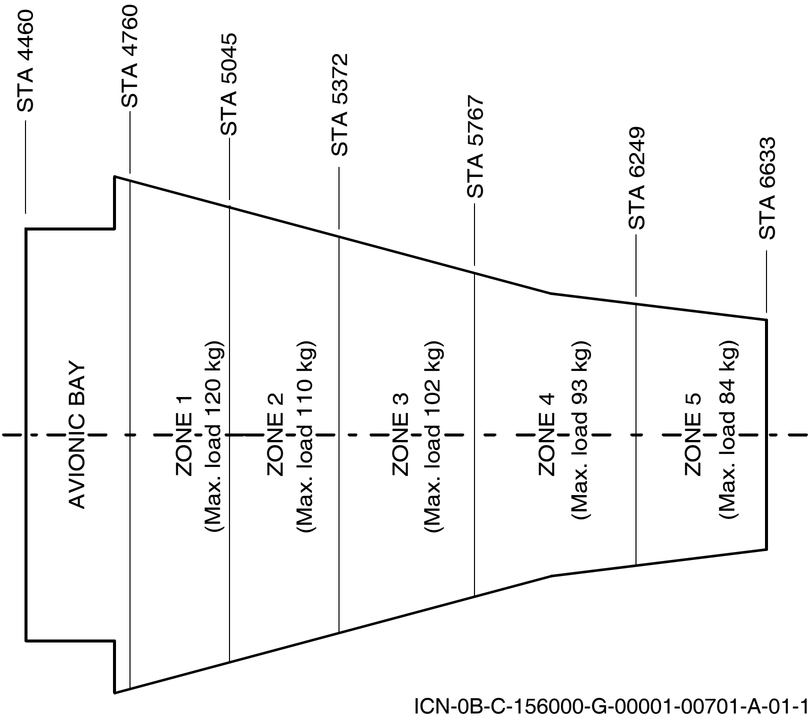

Figure 6-2 Baggage Loading Zones

COMPUTATION OF LOADING

LONGITUDINAL LOADING SAMPLES

The basic weight and moment of the helicopter can be found in Chart B - Helicopter Weighing Record. The values in Chart B were obtained by weighing the helicopter and computing the basic weight, moment and CG.

Subsequently as items of equipment are added or removed, entries should be made in Chart C - Basic Weight and Balance Record and the new basic weight and moment computed.

The basic weight includes fixed ballast, hydraulic fluid, unusable fuel, undrainable engine oil and transmission oil.

Three loading samples are shown as follows (the equipped empty weight and longitudinal CG arm are assumed to be 2181.4 kg and 3482 mm):

Example one

Item	Weight (kg)	STA (mm)	Moment (kg mm)
Equipped empty weight	**2181.4**	**3482**	**7595635**
Pilots (2)	160	1385	221600
Operating weight	**2341**	**3339**	**7817235**
Forward passengers (2)	160	2200	352000
Aft passengers (3)	240	3400	816000
Baggage	40	4880	195200
Dry weight	**2781**	**3301**	**9180435**
Fuel at take-off	360	3881	1397160
Take-off weight	**3141**	**3367**	**10577595**
Fuel at landing	100	3732	373200
Landing weight	**2881**	**3316**	**9553635**

Example two

Item	Weight (kg)	STA (mm)	Moment (kg mm)
Equipped empty weight	**2181.4**	**3482**	**7595635**
Pilot (1)	80	1385	110800
Operating weight	**2261.4**	**3408**	**7706435**
Forward passengers (3)	240	2200	528000
Aft passengers (2)	160	3400	544000
Dry weight	**2661**	**3298**	**8778435**
Fuel at take-off	449.6	3951	1776370
Take-off weight	**3111**	**3393**	**10554804**
Fuel at landing	70	3775	264250
Landing weight	**2731**	**3311**	**9042685**

LATERAL LOADING SAMPLE

The empty weight lateral CG is assumed to be at BL “10.7” unless a different entry has been made in Chart C - Basic Weight and Balance Record. With the empty weight lateral CG the approved lateral loading limits will not be exceeded except under extreme lateral loadings. An example of an asymmetric loading computation is given below.

Lateral example

Item	Weight (kg)	BL (mm)	Moment (kg mm)
Equipped empty weight	**2181.4**	**10.7**	**23341**
Pilot	80	335	26800
Copilot	80	-325	-26000
Operating weight	**2341.4**	**10.3**	**24141**
Forward passenger left	80	-450	-36000
Forward passenger right	80	450	36000
Aft passengers right	80	450	36000
Dry weight	**2581.4**	**23.3**	**60141**
Fuel at take-off	449.6	0	0
Take-off weight	**3031.0**	**19.8**	**60141**
Fuel at landing	90	0	0
Landing weight	**2671.4**	**22.5**	**60141**

WEIGHT AND BALANCE DETERMINATION

Instructions for weight and balance determination are herewith enclosed with instructions for use of charts to enable the operator to obtain all necessary data as to basic helicopter configuration, empty weight and center of gravity. These charts will also provide for continuous control of weight and balance of the helicopter.

This system of weight and balance computation requires the use of charts and forms. They are identified as follows:

Chart A - Equipment List.

Chart B - Helicopter Weighing Record.

Chart C - Basic Weight and Balance Record.

Chart D - Data for Helicopter Weight and Balance Computation

Chart E - Weight and Balance Computation

WEIGHT AND BALANCE DATA RESPONSIBILITY

The Manufacturer inserts all helicopter identifying data on the various charts. This record constitutes the basic weight and balance data of the helicopter, to which the Rotorcraft Flight Manual was assigned, for the condition shown on Chart A. The operator shall keep this data up-to-date by recording all changes made to the configuration of the helicopter.

USE OF CHARTS AND FORMS

USE OF CHART A

The Chart A gives the weight, arm and moment of all the standard and optional equipment. The Manufacturer places check marks in the “Basic Configuration” column to identify the items of equipment in the helicopter for the weighing condition. A check (V) in the columns headed “In Helicopter” indicates the presence of the item in the helicopter, and a zero (0) indicates its absence. The next columns of chart A will permit inspection of the helicopter for equipment actually installed. When making an inventory, note whether any items of equipment have been installed or removed and if so enter corresponding weight and moment change on Chart C.

Subsequent check list inventories shall be carried out in the following cases:

1. When the helicopter undergoes modification, major repair or overhaul.
2. When changes in equipment are made for a different type of operation.
3. When the helicopter is reweighed.

WEIGHING INSTRUCTIONS

Refer to Chapter 08 of the helicopter maintenance publication.

USE OF CHART B

1. Enter the actual scale readings in the first column of Sheet 1. Subtract tare, if any, from the scale readings to obtain the net weight.
2. Multiply the net weights by their respective arms.
3. Add the net weight and moments.
4. Divide the total moment by the net weight to obtain “as weighed” CG position. Transfer the “TOTAL” (as weighed) weight arm and moment to the Sheet 2.
5. Subtract the weight and moment of oil (if reservoir is full) from the “as weighed” total.
6. Subtract the total weight and moment of equipment weighed but not part of the basic helicopter (list these items in column one).

7. Add the weight and moment of unusable fuel.
8. Add the total weight and moment of the basic items not in helicopter when weighed (list these in column two). Added items shall be marked on Chart A.
9. Enter the new basic weight and moment on Chart C.

USE OF CHART C

Chart C is a continuous history of the basic weight and moment resulting from modifications and equipment is considered the current weight and balance status of the basic helicopter. Make additions or subtractions to the basic weight and moment in Chart C as follows:

1. When the helicopter undergoes modification, major repair or overhaul.
2. When changes in equipment are made for a different type of operation.
3. When the helicopter is reweighed.

Note

If any equipment is not listed on Chart A, determine its weight and arm, and list corresponding data on Charts A and C.

USE OF CHART D

The Chart D provides information necessary for weight and balance computation.

USE OF CHART E

The Chart E serves as a work sheet and records the calculations and any corrections that must be made to ensure that helicopter will be within weight and CG limits.

Note

A Chart E shall be filled prior to any flight.

1. Enter the helicopter basic weight and moment. Obtain these figures from the last entry on Chart E.

2. Enter the weight of all applicable items in the marked “Weight”. Obtain the corresponding arms from Chart D and calculate the moments.
3. Add weight and moments. Divide total moment by total weight to obtain CG arm.
4. Ascertain that CG is within allowable limits.
5. Should corrections be required, readjust ballast to return CG within allowable limits.

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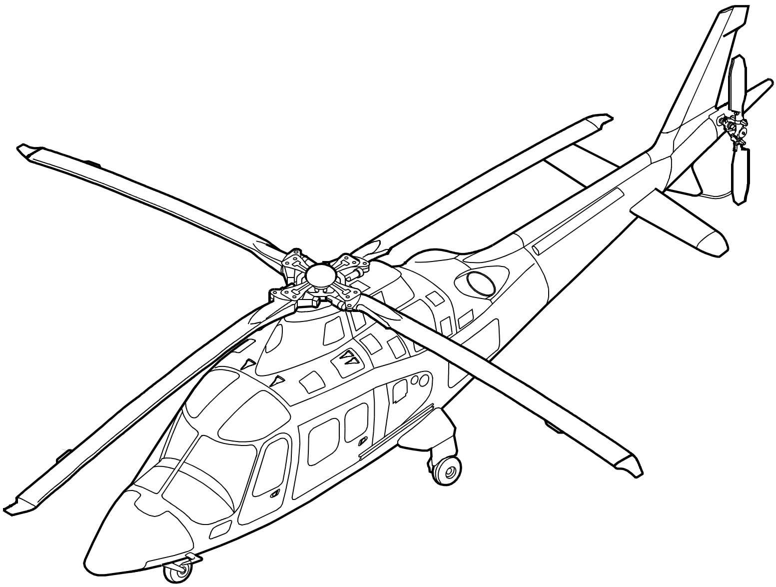
WEIGHT AND BALANCE



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HELICOPTER AW109SP

SERIAL NUMBER ....

REGISTRATION MARKS ....

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For all pages of Chart A:

— √ Means installed on the helicopter

— ○ Means not installed on the helicopter

Page 1 of 1

CHART A - EQUIPMENT LIST

Report 109G0840WXXX

RECORD OF CHECKING (Date and Signature)

MODEL: AW109SP | **REGISTRATION MARKS** | **S/N**

ITEM NUMBER	POWER PLANT SYSTEM		Q.ty	WEIGHT (kg)	ARM (mm)	MOMENT (kg mm)	Basic configuration	In helicopter	Chart C entry	In helicopter	Chart C entry	In helicopter	Chart C entry
	P/N	DENOMINATION											

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CHART B - HELICOPTER WEIGHING RECORD		Sheet 1 of 2
MODEL:	S/N:	REGISTRATION MARKS:
DATE:	PLACE:	SIGNATURE:
Reason for weighing:		
Scale type:		

JACKPOINTS	SCALE READING	TARE	NET WEIGHT	STA (1)	LONGITUDINAL MOMENT	BL (2)	LATERAL MOMENT
	(kg)	(kg)	(kg)	(mm)	(kg mm)	(mm)	(kg mm)
FORWARD				1635		0	
LH AFTERWARD				4475		-450	
RH AFTERWARD				4475		450	
TOTAL (as weighed) (to Sheet 2 of 2)							

Note 1

The forward lower central cabin is provided by two forward jack points, only one forward jack point is assured by using a proper tool which collect both jacking points.

The Station Reference Datum (STA 0) is located 1635 mm forward the forward jack points. Therefore the STA are positive.

E = Distance from the reference datum (STA 0) to the forward jack point Station of 1635 mm.

F = Distance from the reference datum (STA 0) to the left and right aft jackpoints Station of 4475 mm.

Note 2

The Butt Line Reference Datum (BL 0) is located on the fuselage center line.

Therefore the BL are negative on the left side and positive on the right side.

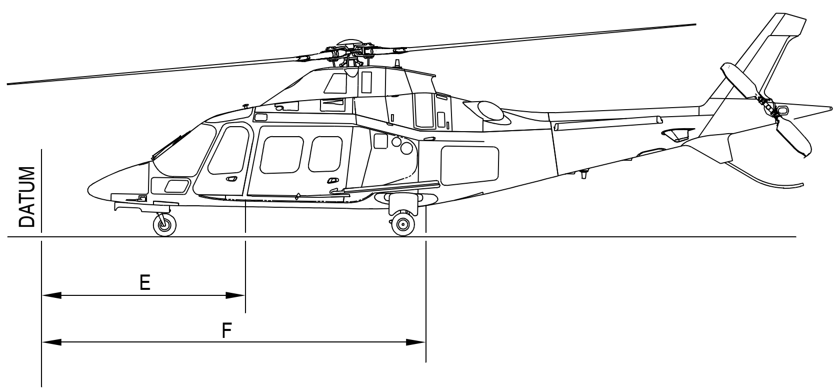

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CHART B - HELICOPTER WEIGHING RECORD					Sheet 2 of 2
DESCRIPTION	NET WEIGHT	STA	LONG. MOMENT	BL	LAT. MOMENT
	(kg)	(mm)	(kg mm)	(mm)	(kg mm)
TOTAL (as weighed) (from Sheet 1 of 2)					
SUBTRACT (from Table 1)					
ADD (from Table 2)					
BASIC AIRCRAFT (to Chart C)					

TABLE 1 ITEMS WEIGHED BUT NOT PART OF BASIC WEIGHT

DENOMINATION	WEIGHT	STA	LONG. MOMENT	BL	LAT. MOMENT
	(kg)	(mm)	(kg mm)	(mm)	(kg mm)
TOTAL					

TABLE 2 BASIC ITEMS NOT INSTALLED WHEN WEIGHED

DENOMINATION	WEIGHT	STA	LONG. MOMENT	BL	LAT. MOMENT
	(kg)	(mm)	(kg mm)	(mm)	(kg mm)
TOTAL					

Reasons of the weighing:	Type Scales:

NOTE :
Remove the weight of the mission equipment items, which are pointed out in Chart A in [ ]

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CHART C BASIC WEIGHT AND BALANCE RECORD														Page N°
MODEL				S/N			REGISTRATION MARKS							
DATE	Item n° see Chart A		DENOMINATION	LOADING CHANGE					BASIC WEIGHT, MOMENT & CG					SIGNAT.
	IN (1)	OUT (2)		WGT (3) (kg)	STA CG (mm)	LONG MOMENT (kg mm)	BL CG (mm)	LATERAL MOMENT (kg mm)	WGT (kg)	LONG MOMENT (kg mm)	STA CG (mm)	LATERAL MOMENT (kg mm)	BL CG (mm)	

Note 1 IN = installed component **Note 2** OUT = removed component
Note 3 Weight of an installed component is positive (+) Weight of a removed component is negative (-)

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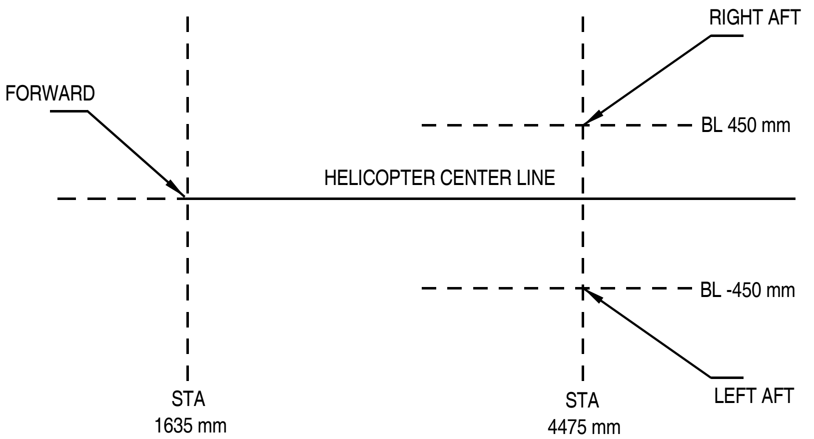
CHART D - DATA FOR HELICOPTER WEIGHT & BALANCE COMPUTATION

MODEL AW109SP

JACKPOINTS LOCATION



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Longitudinal STA 0 (datum) is 1635 mm forward of the front jackpoint.

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CHART E - WEIGHT & BALANCE COMPUTATION FORM

MODEL	S/N	REGISTRATION MARKS	DATE	PLACE	COMPUTED BY

REF.	ITEM	WEIGHT (kg)	STA (mm)	LONG. MOMENT (kg mm)	BL (mm)	LATL. MOMENT (kg mm)
1	HELICOPTER BASIC (Ref. To Chart C)					
2	PILOT					
3	COPILOT					
4	PASSENGER					
5	PASSENGER					
6	PASSENGER					
7	PASSENGER					
8	PASSENGER					
9	PASSENGER					
10	LOOSE EQUIPMENT					
11	CABIN LOAD					
12	BAGGAGE COMPARTMENT LOAD					
14						
15						
16						
17						
18						
19						
20						
21	**DRY WEIGHT**					
22	FUEL (at take-off)					
23	**GROSS WEIGHT (at take-off)**					
24	FUEL (at landing)					
25	**GROSS WEIGHT (at landing)**					
26	**BALLAST (if required)**					

LIMITATIONS	REMARKS
Refer to Section 1 of this RFM	

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SECTION 7

SYSTEM DESCRIPTION

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HELICOPTER GENERAL

(Chapter 00)

THE AW109SP HELICOPTER

(Figure 7-1)

The AW109SP is a high-speed, high performance, multipurpose helicopter powered by two Pratt & Whitney Canada PW207C engines.

The helicopter has:

– A four-bladed, fully articulated main rotor
– A low drag titanium and composite main rotor head with elastomeric bearings
– A two-bladed tail rotor
– A retractable tricycle-type landing gear
– A fuselage forward section and an aft section (tail boom).

The helicopter is designed to fulfill many roles, which include transport of personnel, equipment and cargo. It may also be used for aerial supply and rescue operations. Optional equipment may be fitted for specific missions (ambulance duties, transport of external loads, etc).

The airframe consists of two major assemblies: the fuselage and the tail boom.

The fuselage comprises the forward fuselage, the center fuselage and the aft fuselage.

The forward fuselage includes a nose bay for the installation of electric and avionic equipment and a bottom bay that accommodates the nose landing gear, the hydraulic accumulators and other hydraulic components. The forward fuselage also includes the cockpit. A hinged door on each side of the forward fuselage provides access to the cockpit.

The center fuselage includes the passenger compartment (cabin), the fuel tank bay, the landing gear bays. A sliding door is located on each side of the center fuselage for access to the passenger compartment.

The cabin is generally rigged to carry six passengers in two three-seater benches. Other configurations can be arranged for mission-specific requirements.

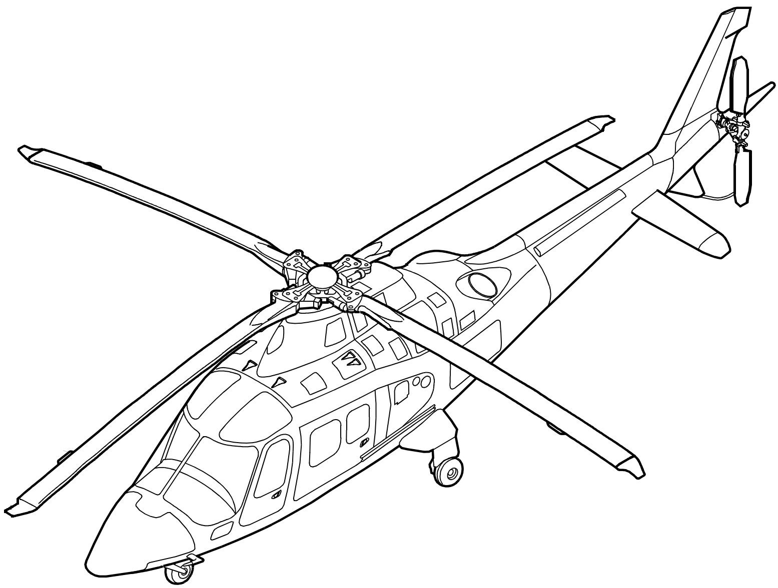

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Figure 7-1 The AW109SP Helicopter

The aft fuselage includes the avionic equipment bay, the electric equipment bay and the baggage compartment. A hinged door on the right side of the aft fuselage permit the access to the baggage compartment.

The deck located above the center fuselage and the aft fuselage accommodates the hydraulic components (reservoirs, filter groups, shut-off valves), the main transmission, the oil coolers and the engines. The upper deck is enclosed by removable fairings and cowlings.

The tail boom is attached to the aft fuselage at four points. The vertical fin is an integral part of the tail boom structure. The tail boom accomodates the Number 3 tail-rotor drive-shaft, the 90-degree gearbox, the tail rotor, the elevators and the tail skid. The tail cone is attached to the aft end of the tail boom.

HELICOPTER DIMENSIONS

Refer to Figure 7-2.

GENERAL ARRANGEMENT

Refer to Figure 7-3.

COCKPIT ARRANGEMENT

Refer to Figure 7-4.

INSTRUMENT PANEL

Refer to Figure 7-5.

FRONT CONSOLE

Refer to Figure 7-6.

INTERSEAT CONSOLE

Refer to Figure 7-7.

OVERHEAD CONSOLE

Refer to Figure 7-8.

CIRCUIT BREAKER PANEL

Refer to Figure 7-8.

ELECTRICAL CONTROL PANEL

Refer to Figure 7-8.

ENVIRONMENTAL CONTROL SYSTEM CONTROL PANEL

Refer to Figure 7-8.

ENGINE CONTROL QUADRANT

Refer to Figure 7-8 and Figure 7-9.

CYCLIC STICK HANDGRIP

Refer to Figure 7-10.

COLLECTIVE LEVER HANDGRIP

Refer to Figure 7-11.

CABIN ARRANGEMENT

Refer to Figure 7-12.

ANTENNA LOCATION

Refer to Figure 7-13.

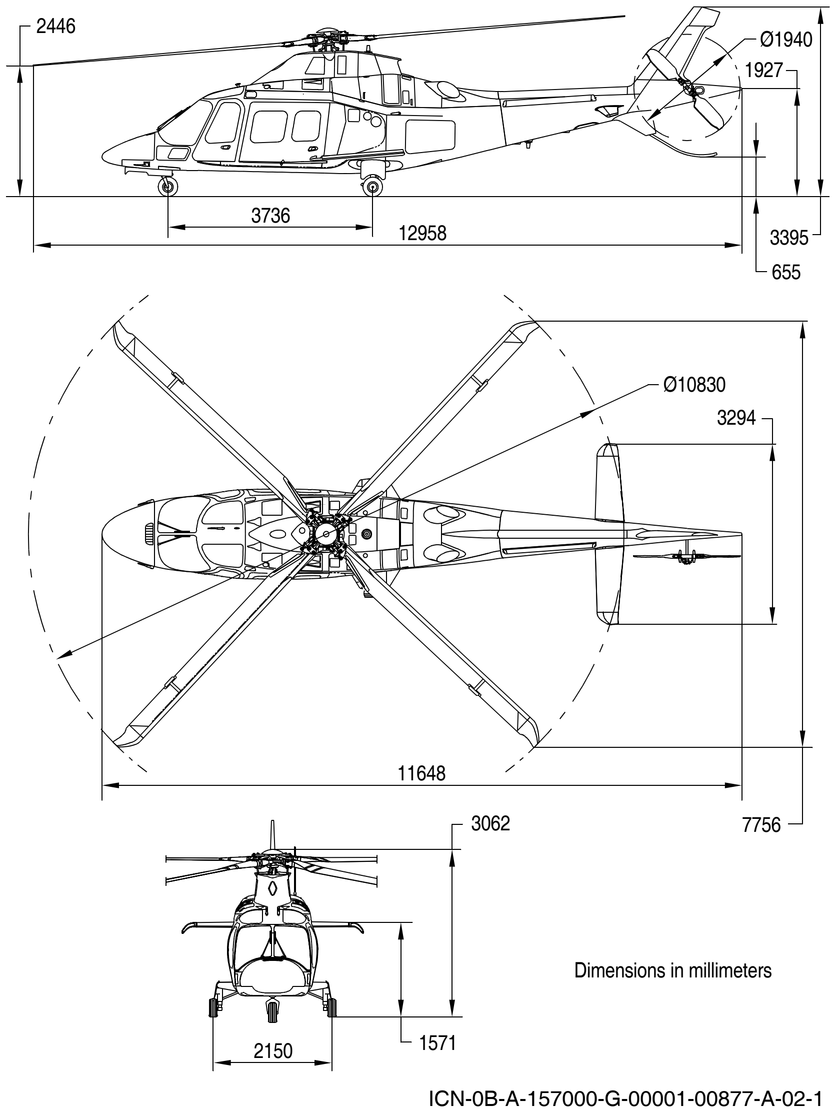

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Figure 7-2 Helicopter Dimensions

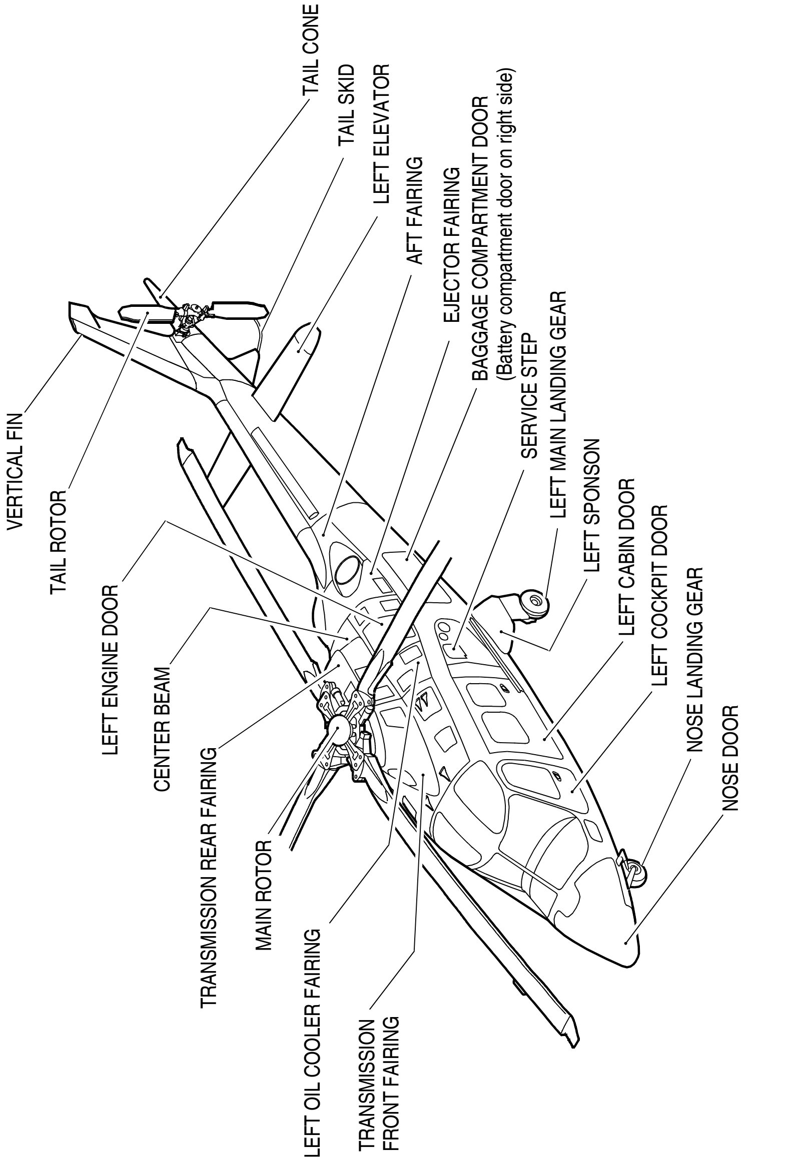

Figure 7-3 General Arrangement

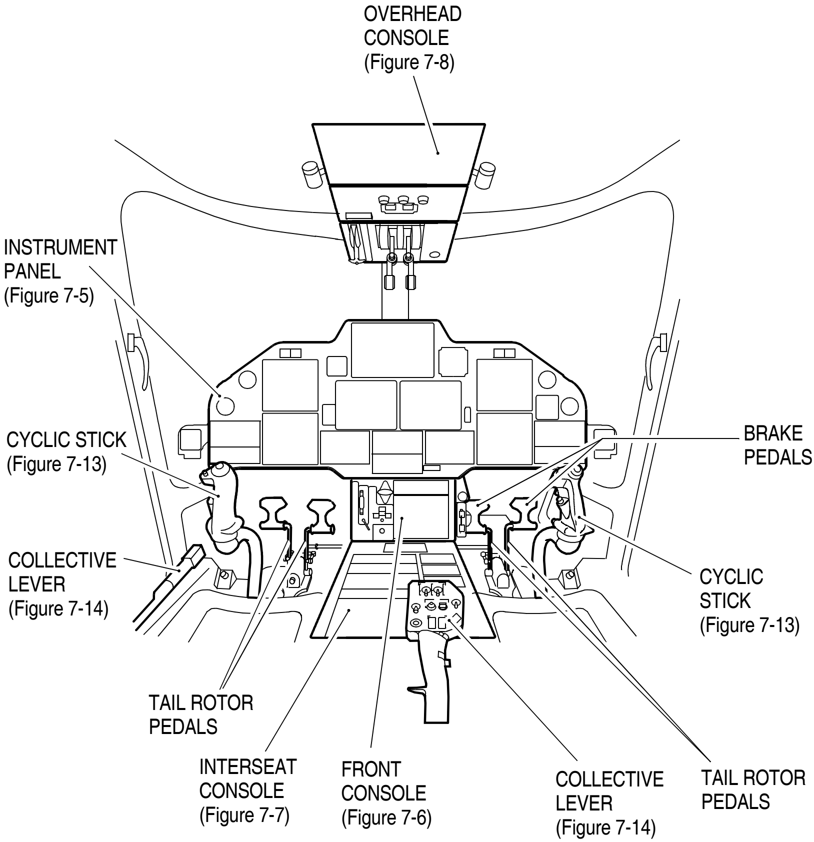

Figure 7-4 Cockpit Arrangement

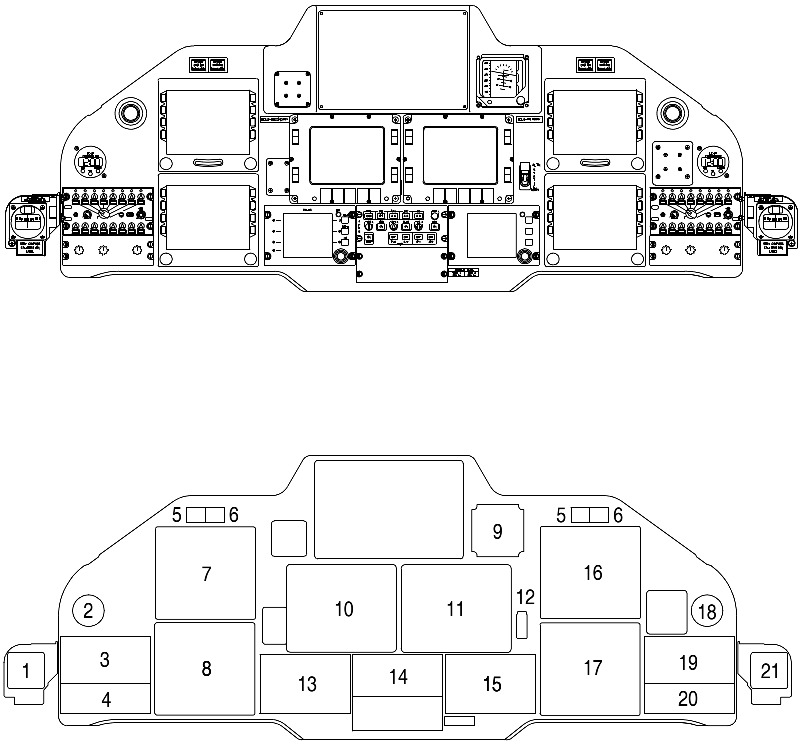

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Figure 7-5 Instrument Panel

Key to Figure 7-5

Ref	Description
1	Copilot magnetic compass
2	Copilot digital chronometer
3	Copilot audio control panel
4	Copilot remote bugs panel
5	Master caution light
6	Master warning light
7	Copilot Primary Flight Display (PFD 1)
8	Copilot Multifunction Display (MFD 1)
9	Standby electronic indicator
10	Secondary EDU (EDU 2)
11	Primary EDU (EDU 1)
12	STATIC switch
13	Copilot RTU panel (RTU 1)
14	APMS panel
15	Pilot RTU panel (RTU 2)
16	Pilot Primary Flight Display (PFD 2)
17	Pilot Multifunction Display (MFD 2)
18	Pilot digital chronometer
19	Pilot audio control panel
20	Pilot remote bugs panel
21	Pilot magnetic compass

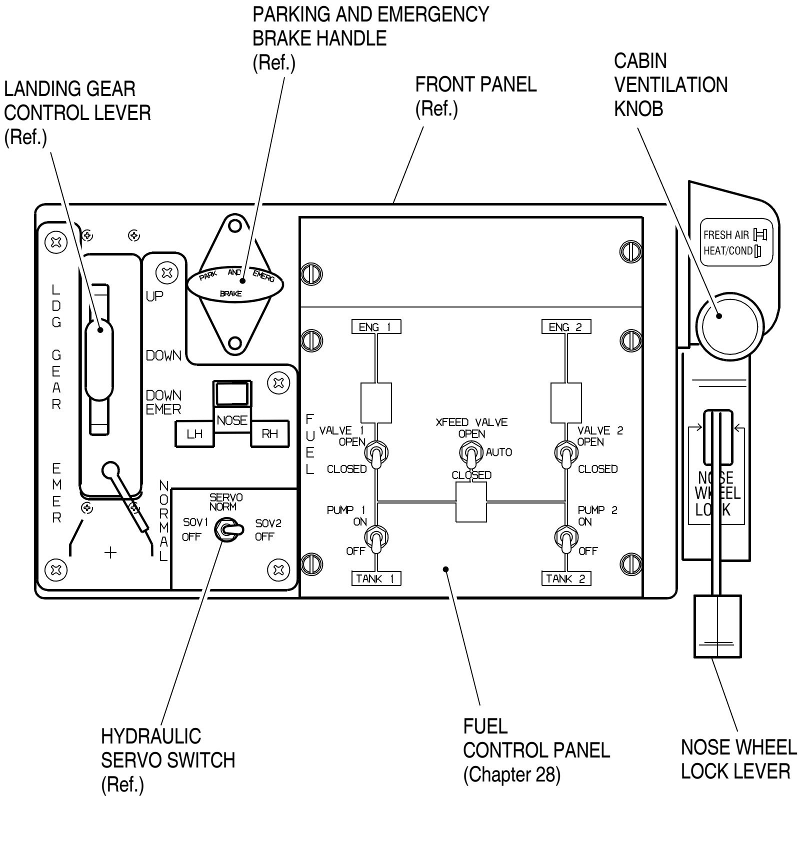

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Figure 7-6 Front Console (Typical)

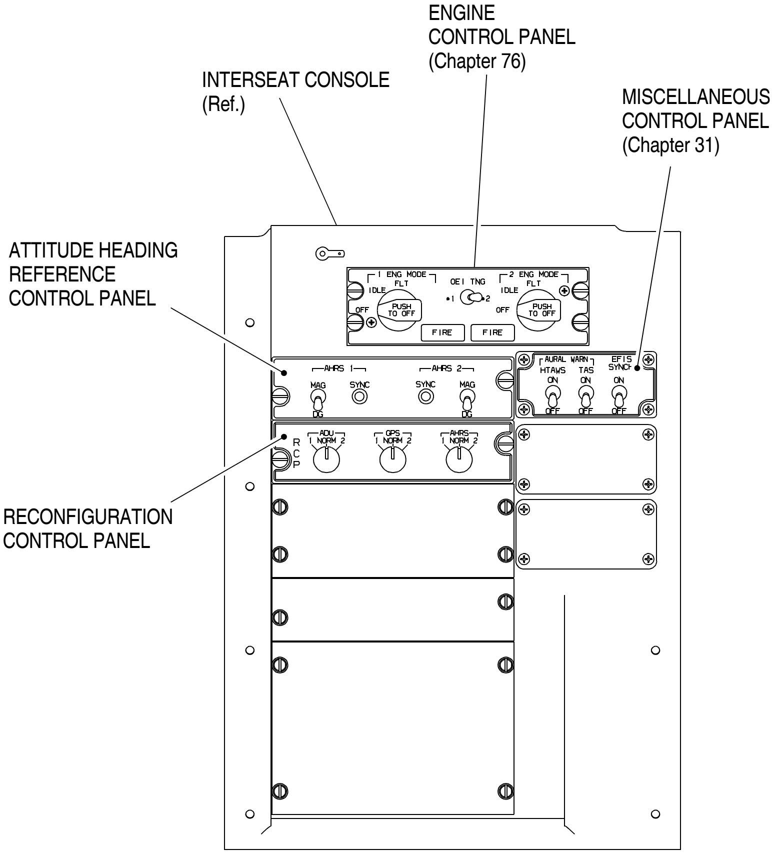

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Figure 7-7 Interseat Console

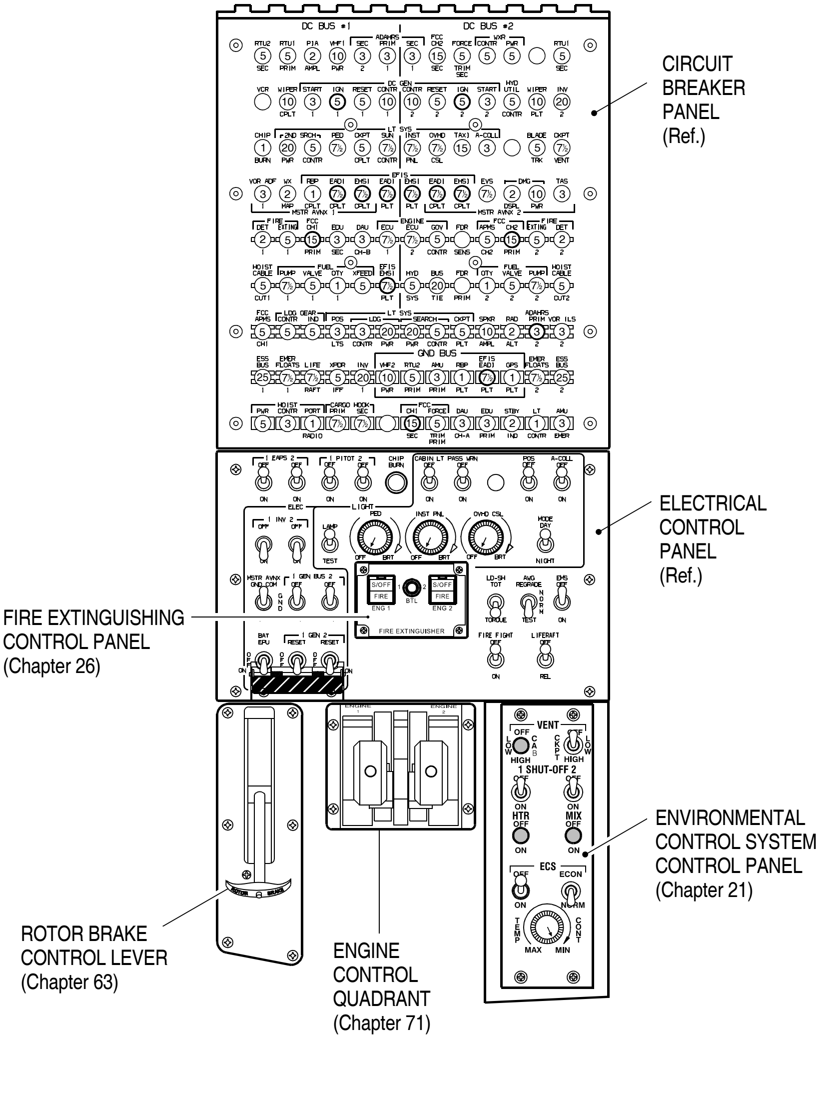

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Figure 7-8 Overhead Console

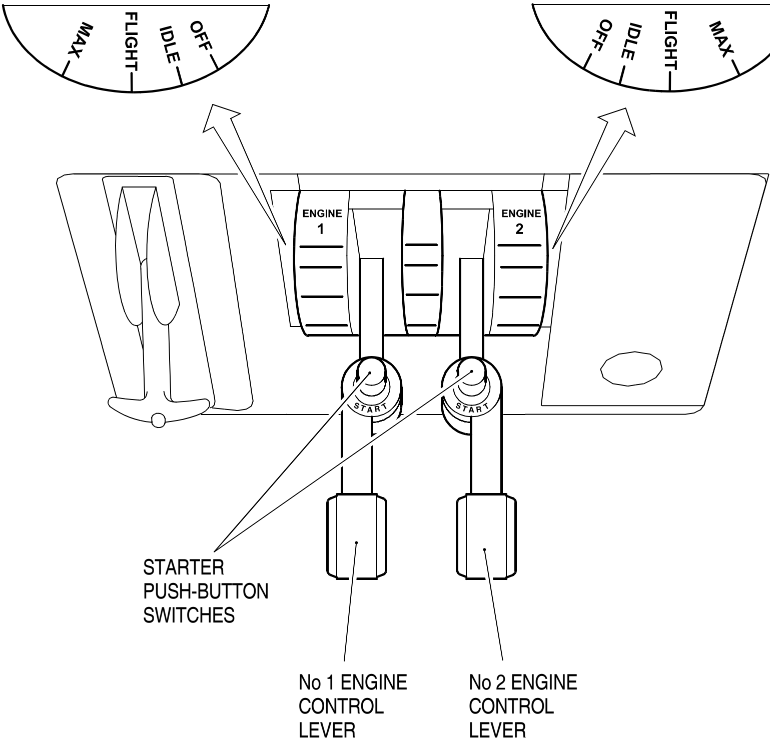

Figure 7-9 Engine Control Quadrant

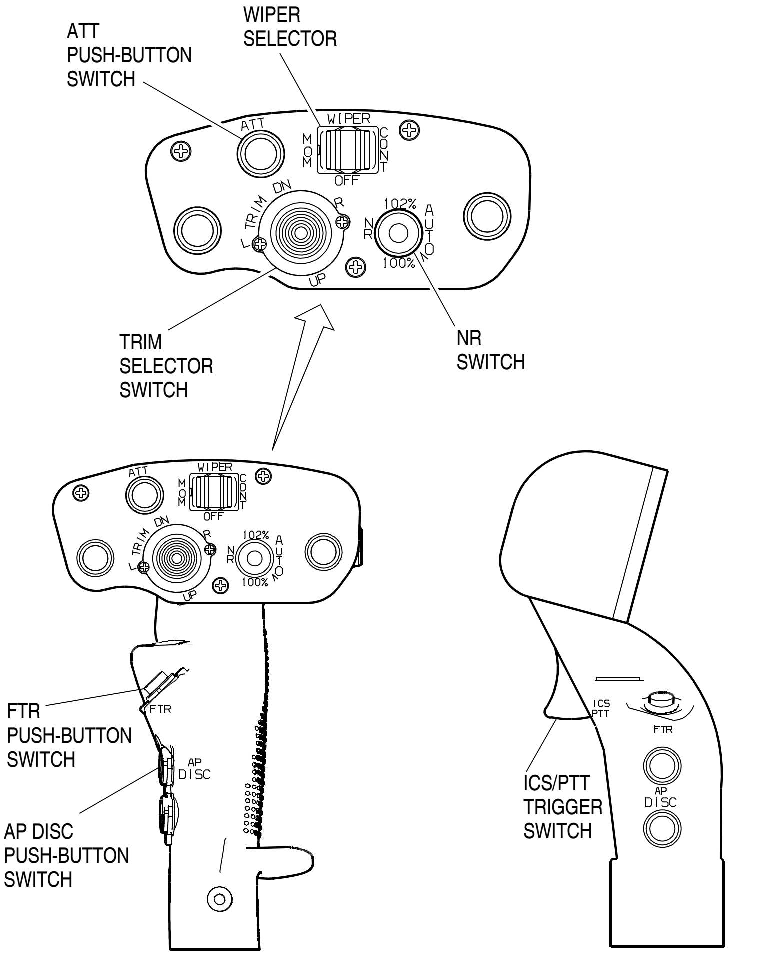

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Figure 7-10 Cyclic Stick Handgrip

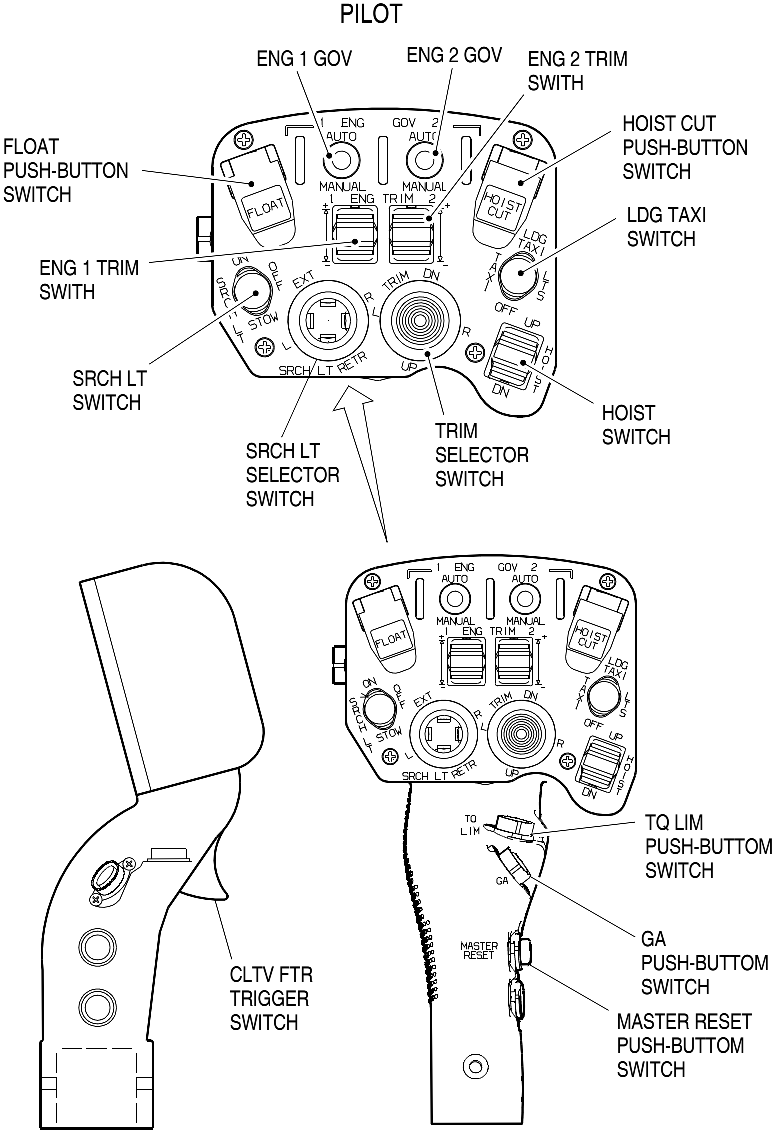

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Figure 7-11 Collective Lever Handgrip (Sheet 1 of 2)

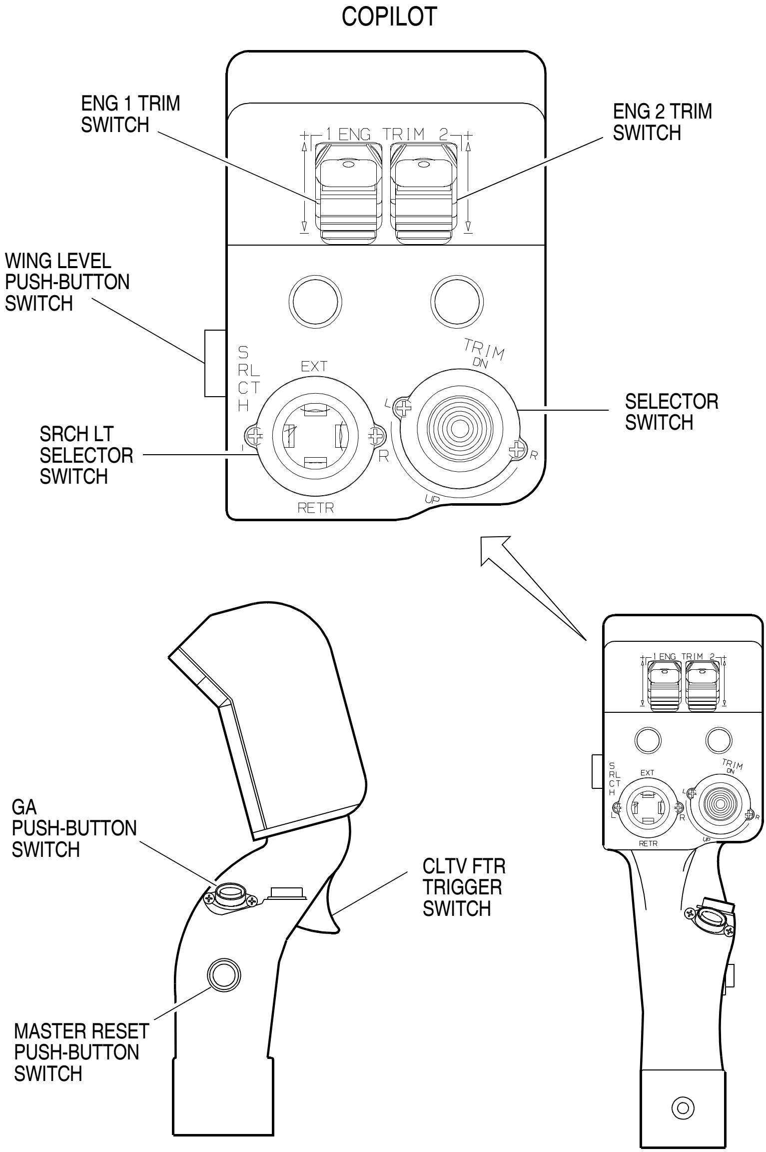

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Figure 7-11 Collective Lever Handgrip (Sheet 1 of 2)

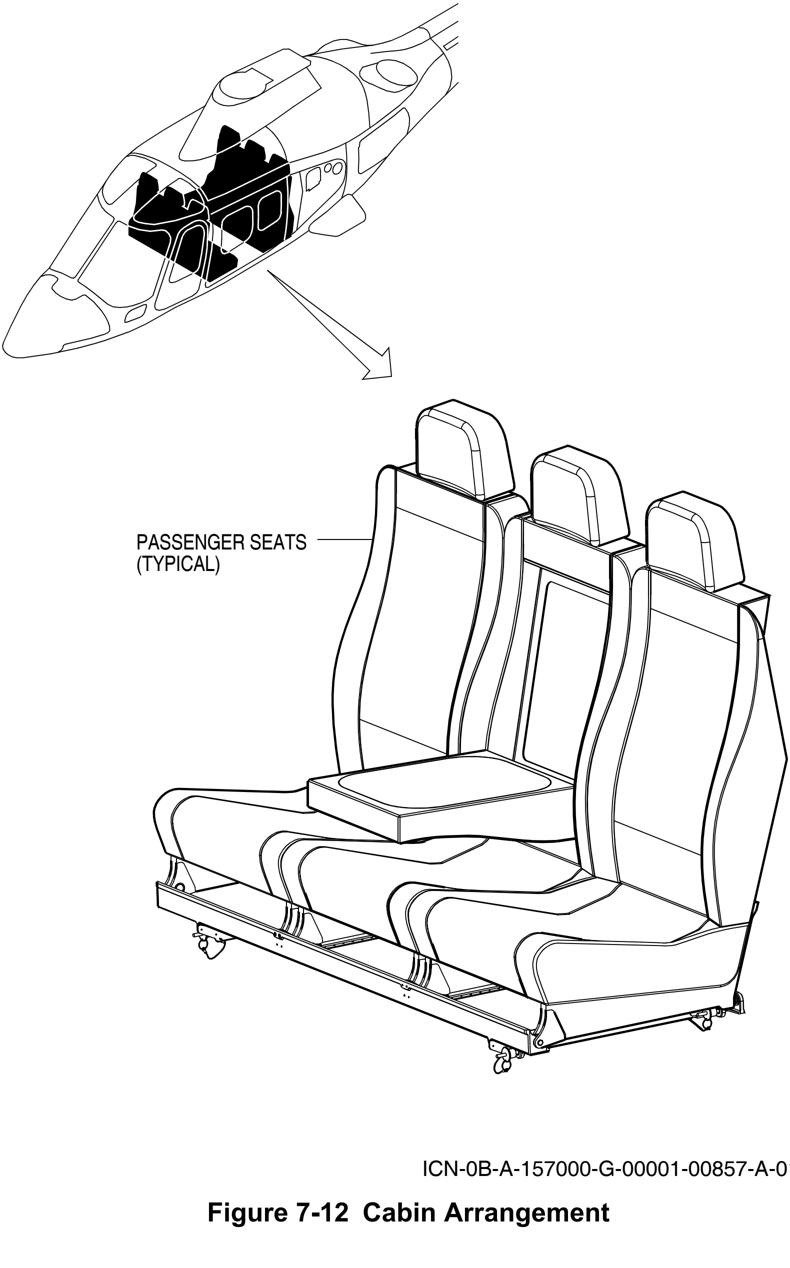

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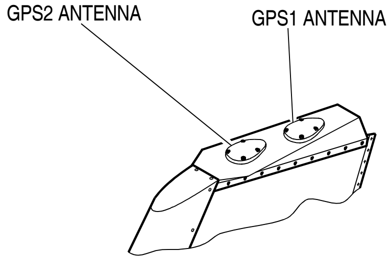
Figure 7-12 Cabin Arrangement



(Top of vertical fin)

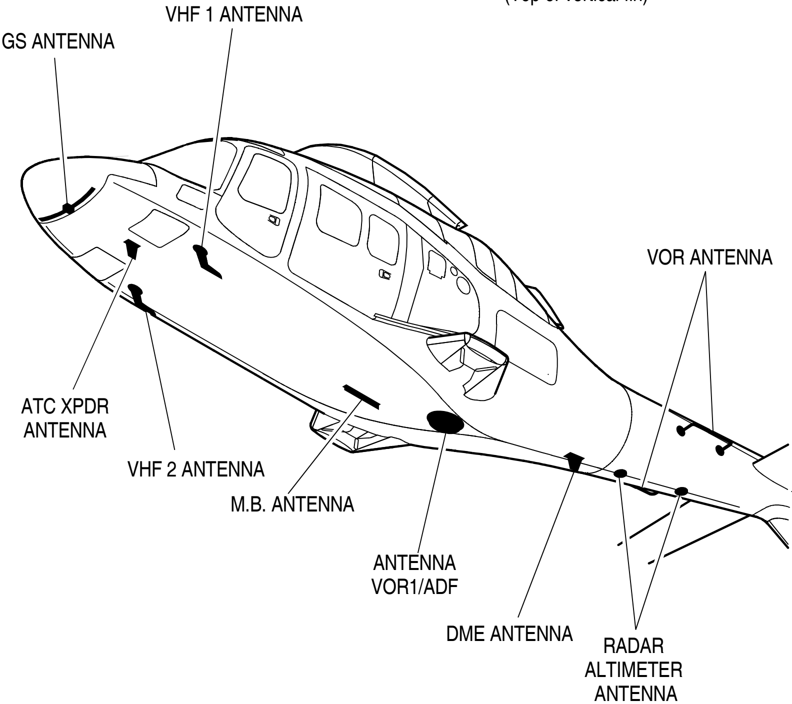


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Figure 7-13 Antenna Location

ENVIRONMENTAL CONTROL

(Chapter 21)

GENERAL

The Environmental Control System (ECS) gives a means of heating, cooling and treating the air used for the ventilation and the conditioning of the cockpit, the cabin and the avionic equipment.
The environmental control system includes:

– The ventilation system
– The avionic equipment ventilation systems
– The integrated ECS (optional) or the heating system (optional).

VENTILATION SYSTEM

The ventilation system supplies forced ducted air to the cockpit and the cabin in order to provide conditioned air to the aircrew and to the passengers.

The ventilation system has two different circuits:

– The floor ventilation circuit
– The ceiling ventilation circuit.

The main components of the ventilation system are shown in Figure 7-14.
Refer to Figure 7-15 for the ventilation system schematic diagram.
The ventilation of the cockpit and the cabin at floor level is achieved through ram air from the nose air intake conveyor and a ventilation fan located on the aft bulkhead of the nose bay.
The ventilation fan is powered through the circuit breaker that follows:

– CKPT VENT (MAIN BUS #2).

The ventilation fan is controlled by the VENT - CKPT switch located on the ECS control panel of the overhead console.
The ram air is ducted to the air mixing box and distributed through the two windshield diffusers, the two air valves on the sides of the instrument panel, the two air valves on the cockpit floor and the two air valves on the cabin floor.

The rate and direction of airflow from the air valves can be manually controlled by moving them.
The ventilation of the cockpit and the cabin at ceiling level is achieved through ram air from the air scoop located on the upper deck.
The ram air is ducted to the two cockpit diffusers and the four cabin air valves.

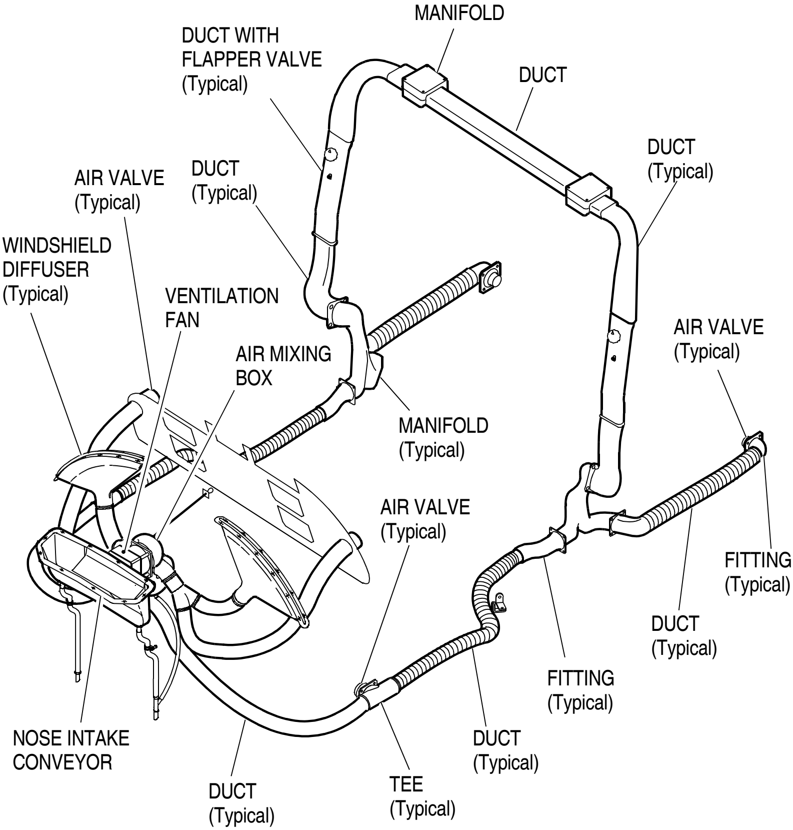

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Figure 7-14 Ventilation System - Component Location (Sheet 1 of 2)

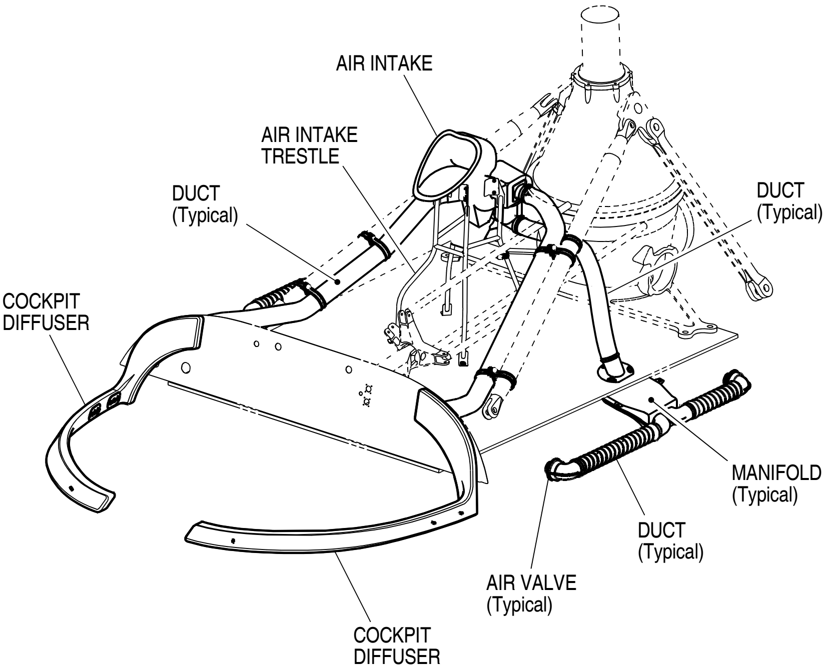

Figure 7-14 Ventilation System - Component Location (Sheet 2 of 2)

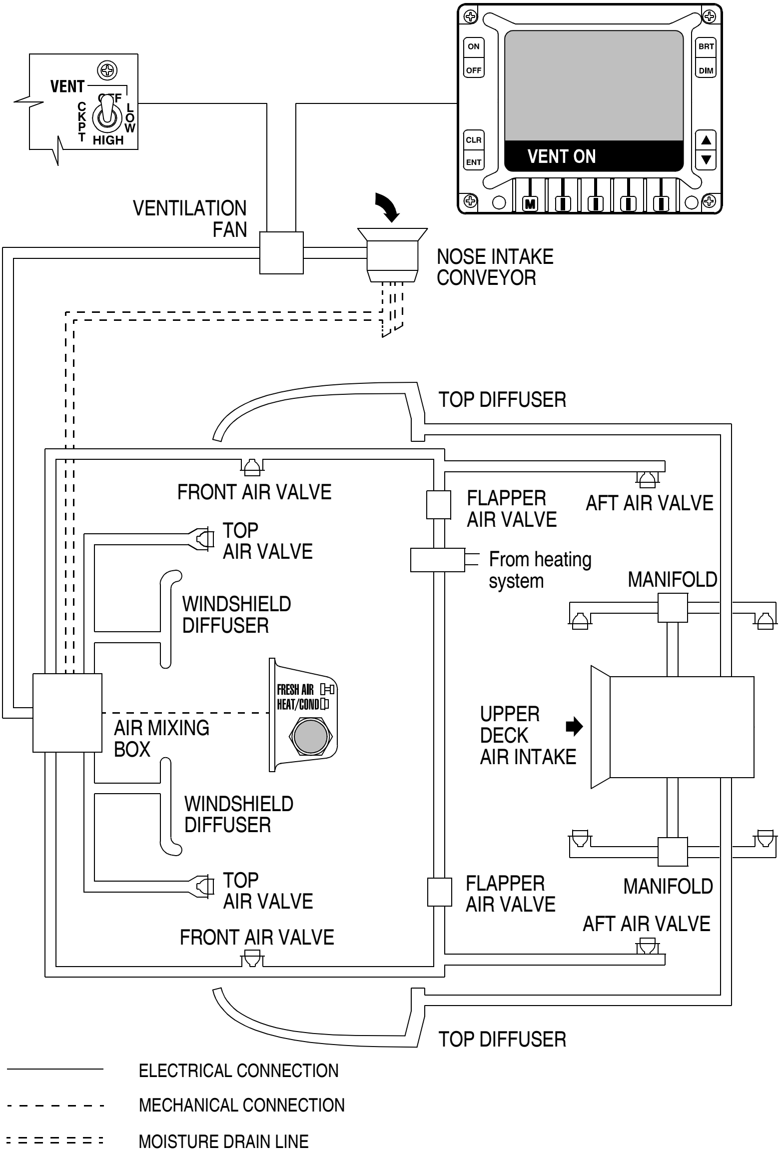

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Figure 7-15 Ventilation System - Schematic Diagram

INTEGRATED ECS

The function of the integrated ECS is to supply a temperature-controlled airflow, under a light pressure, to the cockpit and the cabin in order to maintain a comfortable temperature in the two compartments.
The main components of the ECS system are shown in Figure 7-16.
Refer to Figure 7-17 for the schematic diagram of the system.
For its operation the system uses hot and compressed air bled from the engines.
The bleed flow is routed via a shut-off valve from each engine to the air conditioning unit.
The air conditioning unit consists mainly of a heat exchanger, a condenser, a turbine-blower unit and a temperature control circuit.
The compressed air from the air shut-off valves and the pressure reducer control valve is initially cooled in the heat exchanger by a ram air flow supplied by the blower. The ram air enters the air intake and flows overboard through a duct.
Additional cooling to below ram air temperature then occurs in the condenser where the bleed air is continually cooled by recirculated cold air from the turbine.
Condensed water from the air is collected in a water separator and sprayed by the water injector into the ram air inlet where it re-evaporates and increases the cooling effectiveness of the heat exchanger.
The cool high-pressure bleed air from the condenser passes through the turbine inlet control valve which controls the air supply to the cooling turbine during heating demands on the system, and is then expanded through the cooling turbine where its temperature is further reduced by the extraction of energy from air.
Shaft energy produced in the turbine drives the fan which induces the ram air cooling across the heat exchanger. The air leaving the turbine enters the primary nozzle of the jet pump where it induces recirculation airflow across the cold side of the condenser. This recirculation air is drawn from the conditioned air supply duct, and passes through the condenser where it cools the bleed air, then mixes with the cold turbine discharge air in the jet pump, and returns to the conditioned air supply ducts.
The temperature control circuit has two modes of operation: normal and economy.
The two modes are selected by means of the ECS - ECON/NORM switch on the ECS control panel on the overhead console. With the

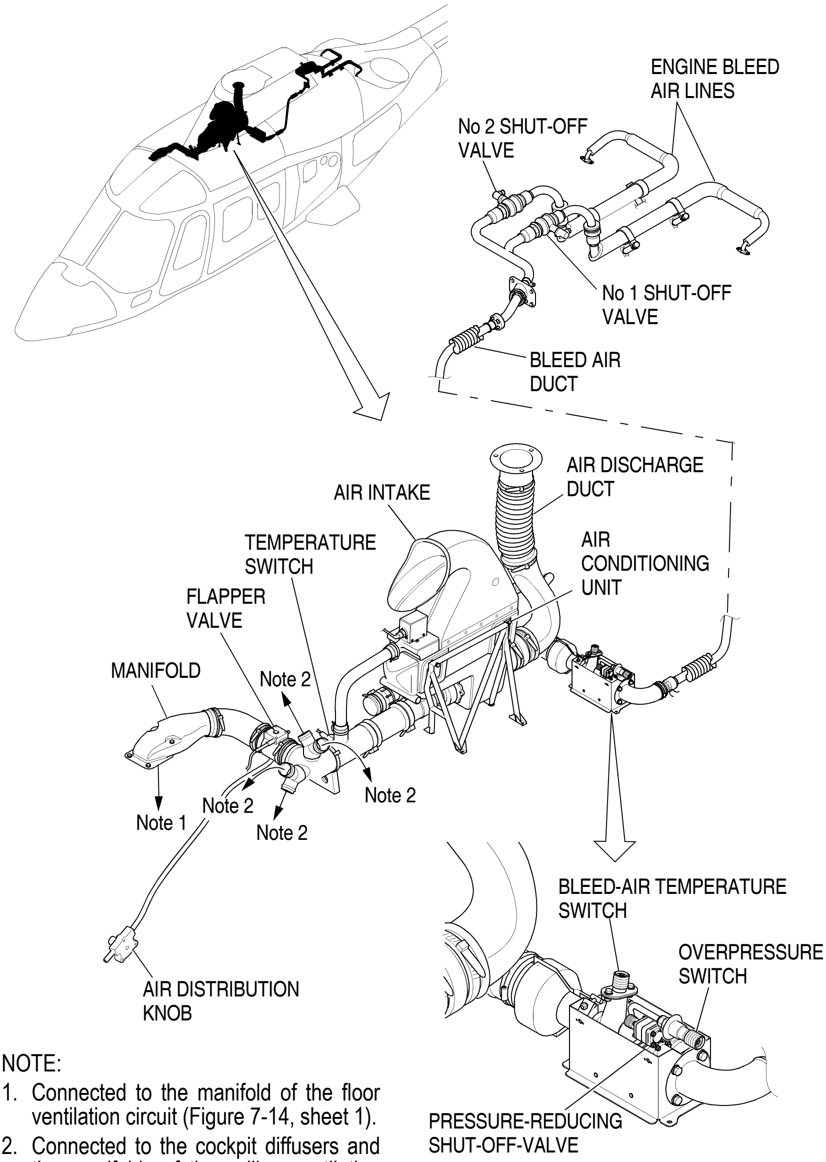
switch set to NORM, the economy mode valve is energized in the closed position. The conditioned air temperature is controlled by means of potentiometer actuated by the CONT knob in the ECS control panel.
The potentiometer activates the temperature sensor, to energize the temperature control valve, positioning the valve to achieve the selected temperature. The temperature control valve controls the flow of the hot (bleed) air mixed with the turbine discharge air and supplies actuating pressure to close the turbine inlet control valve in the same proportion as the valve moves open.
The economy operating mode is selected by setting the ECS - ECON/ NORM switch to ECON. In this condition the economy mode valve is de-energized and opens, thus increasing the actuating pressure supplied to the turbine inlet control valve, which in turn reduces the quantity of conditioned air supplied in the cabin.

Between the collector and the manifold is installed the flapper valve assembly. The flapper valve assembly is controlled by the AIR DISTRIBUTION knob on the right side of the overhead panel.
The flapper valve assembly permits at the conditioned air to flow through the two cockpit diffusers only.

In event of engine fire, the operation of the ENG1 (2) - S/OFF FIRE lighted push-button switch causes the closure of the respective bleed-air shut-off valve. When fire is extinguished, to open the bleed-air shut-off valves again it is necessary to push the ENG1 (2) - S/OFF FIRE lighted push-button switch.

INTEGRATED ECS CONTROLS AND DISPLAYS

Refer to Figure 7-18.



NOTE:

1. Connected to the manifold of the floor ventilation circuit (Figure 7-14, sheet 1).
2. Connected to the cockpit diffusers and the manifolds of the ceiling ventilation circuit (Figure 7-14, sheet 2)

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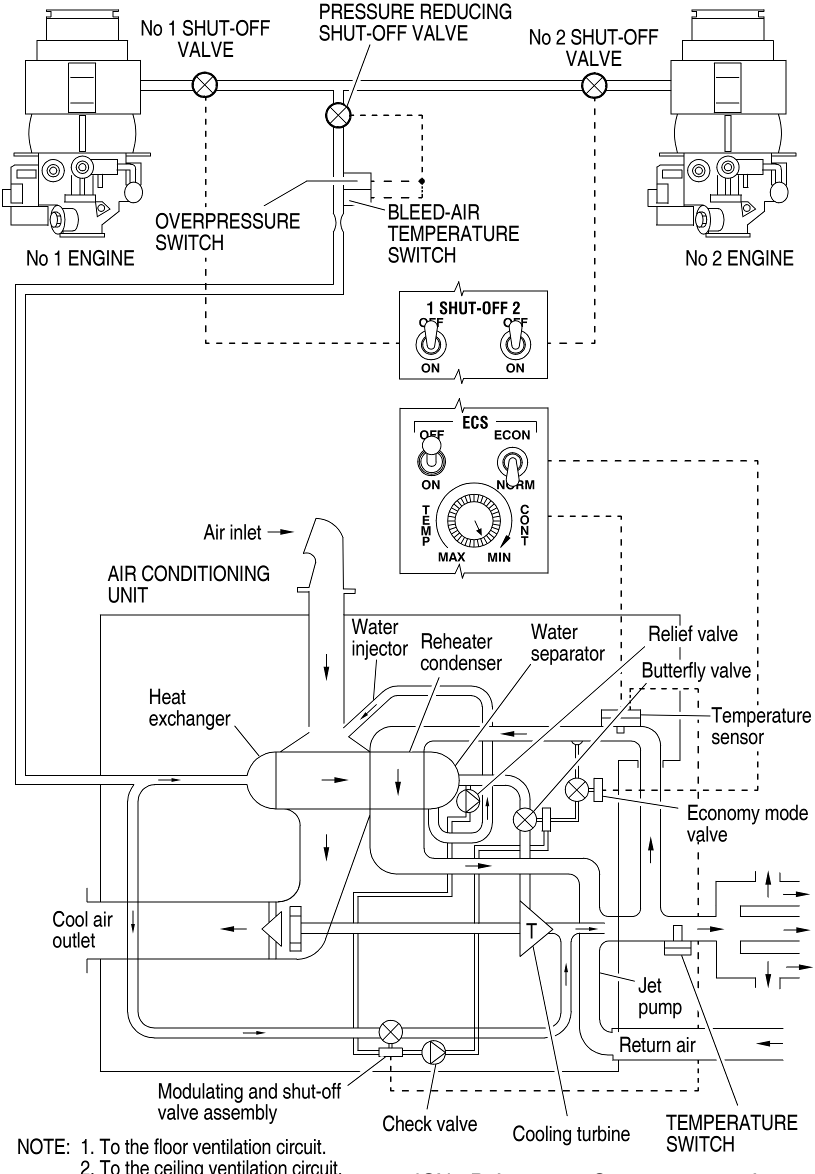
Figure 7-16 Integrated ECS - Component Location



NOTE: 1. To the floor ventilation circuit.
2. To the ceiling ventilation circuit.

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Figure 7-17 Integrated ECS - Schematic Diagram

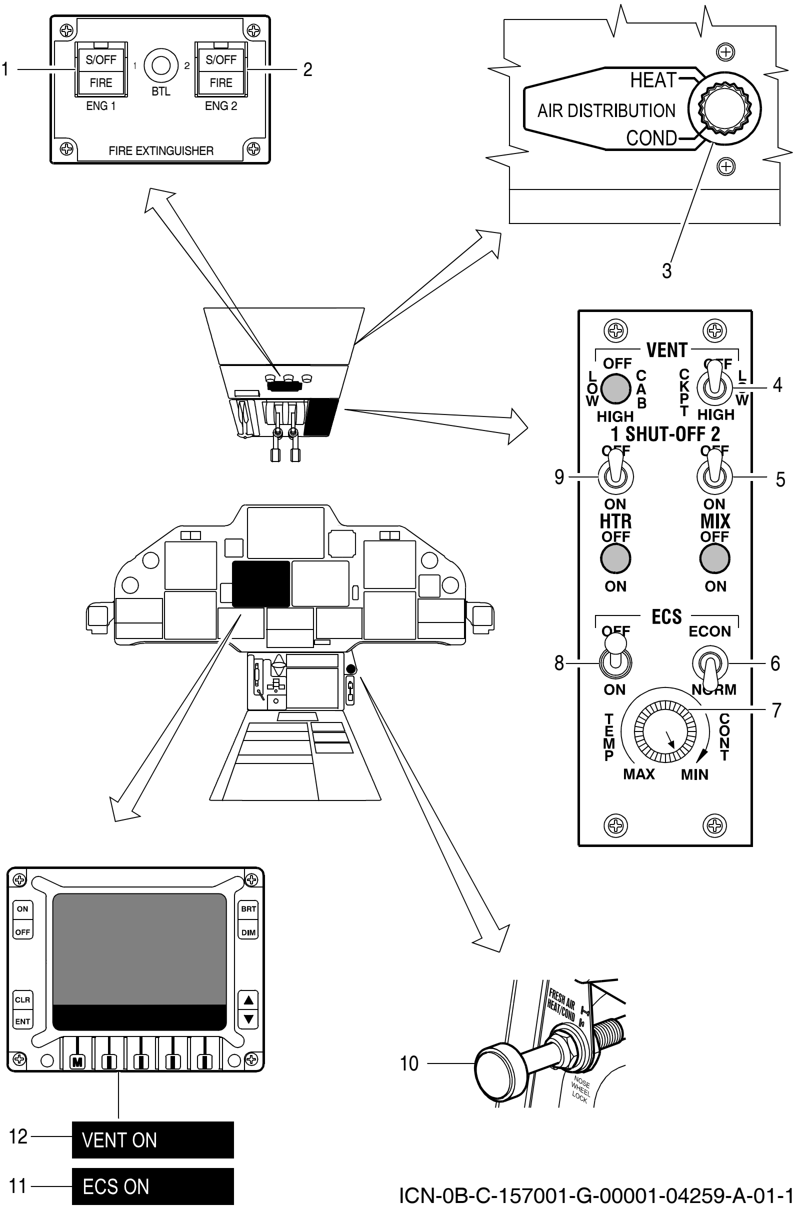


Figure 7-18 Integrated ECS - Controls and Displays

Key to Figure 7-18

Ref.	Control/Display	Function
1	ENG1 - S/OFF FIRE lighted push-button switch	Refer to Chap 26.
2	ENG2 - S/OFF FIRE lighted push-button switch	Refer to Chap 26.
3	AIR DISTRIBUTION knob	HEAT - The gate valve of the flapper valve is open. The conditioned air flows to the cockpit and the cabin. COND- The gate valve of the flapper valve is closed. The conditioned air flows only to the two cockpit diffusers and the cabin air valves.
4	VENT - CKPT switch	OFF - the ventilation fan is de-energized. LOW - the ventilation fan is operating at low speed. HIGH - the ventilation fan is operating at maximum speed.
5	SHUT-OFF 2 switch	OFF - the No 2 shut-off valve is de-energized (valve closed). ON - the No 2 shut-off valve is energized (valve open).
6	ECS - ECON/NORM switch	NORM - the economy mode valve is energized (valve closed). The temperature control circuit operates in the normal mode. ECON - the economy mode valve is de-energized (valve open). The temperature control circuit operates in the economy mode.
7	TEMP CONT knob	Adjusts the temperature of the conditioned air from MIN to MAX.
8	ECS - OFF/ON switch	OFF - the integrated ECS electrical-circuit is not powered. ON - the integrated ECS electrical-circuit is powered.

Key to Figure 7-18 (Continued)

Ref.	Control/Display	Function
9	SHUT-OFF 1 switch	OFF - the No 1 shut-off valve is de-energized (valve closed). ON - the No 1 shut-off valve is energized (valve open).
10	Ventilation knob	HEAT/COND No air from the outside enters the cockpit and the cabin. FRESH AIR - the gate valve of the mixing box is open. Ram air from the outside enter the cockpit and the cabin. When the knob is in an intermediate position the air flow is proportional to the knob position.
11	ECS ON advisory message	Shown when the ECS switch is set to ON.
12	VENT ON advisory message	Shown when the VENT - CKPT switch is set to LOW or HIGH.

HEATING SYSTEM

The heating system supplies “temperature - controlled” hot air to the aircrew and to the passengers.

The main components of the heating system are shown in Figure 7-19. Refer to Figure 7-20 for the heating system schematic diagram.

The hot air is taken from the compressor sections of the two engines. The flow of the hot air is controlled by the two shut-off valves and is ducted to the mixing valve. In the mixing valve, the hot air is mixed with the ram air from an air scoop. From the mixing box the air is ducted to the plenum chamber, installed on the manifold of the floor ventilation circuit.

The plenum chamber accomodates the temperature switch and the temperature sensor.

The temperature switch removes the electrical power to the system when the temperature in the plenum chamber exceed 127 °C.

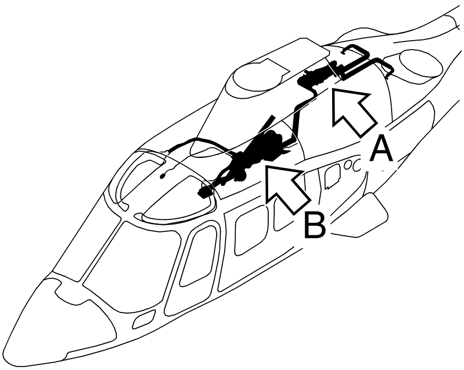
The temperature sensor of the thermal pole expansion type, discharges the hot air from the inlet port of mixing valve, altering the valve internal pressure and adjusting the mixed air flow through the mixing valve.

The temperature sensor is controlled by the TEMP CONT knob on the heating control panel on the overhead console. The operating range is 18 thru 113 °C (±5 °C).

In event of engine fire, the operation of the ENG1 (2) - S/OFF FIRE lighted push-button switch causes the closure of the respective bleed-air shut-off valve. When fire is extinguished, to open the bleed-air shut-off valves again it is necessary to push the ENG1 (2) - S/OFF FIRE lighted push-button switch.

HEATING SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-21.



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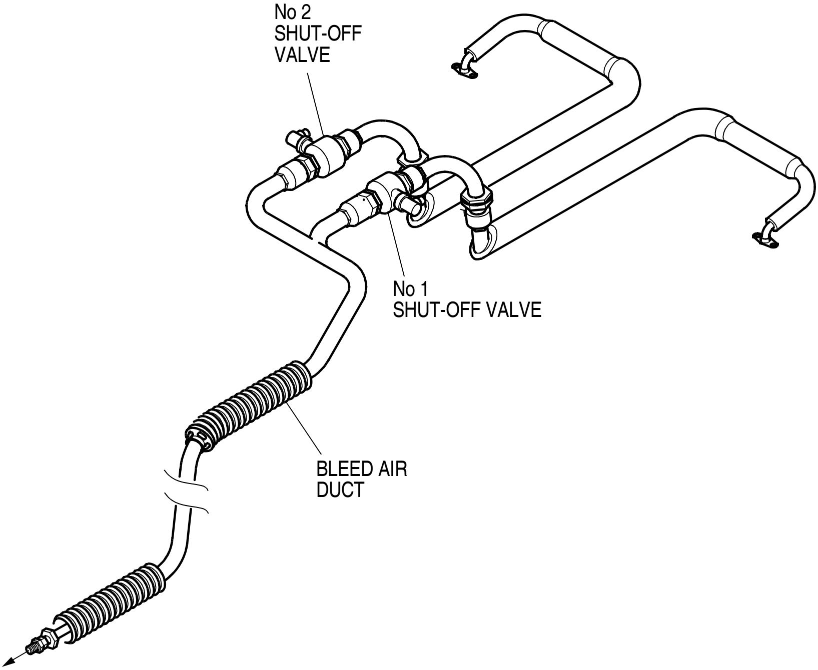

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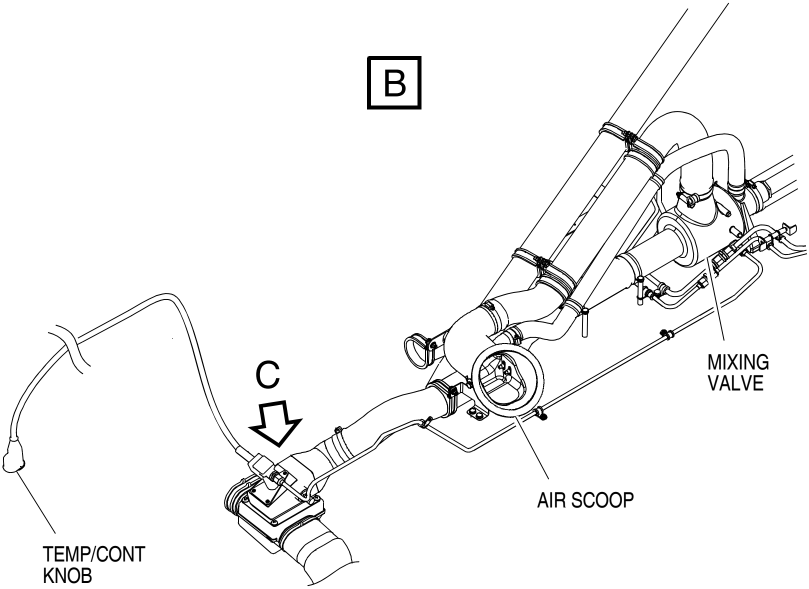
Figure 7-19 Heating System - Component Location (Sheet 1 of 2)



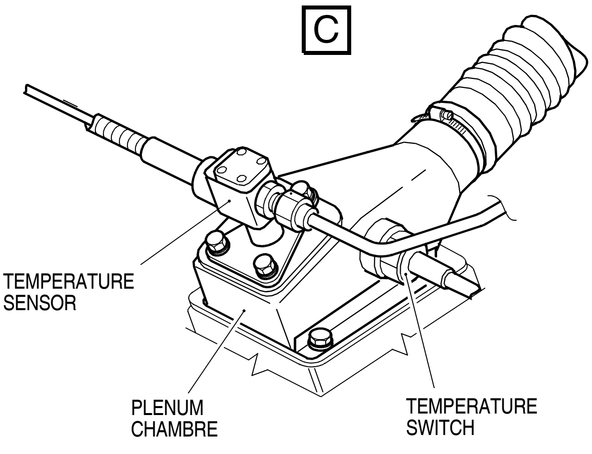


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Figure 7-19 Heating System - Component Location (Sheet 2 of 2)

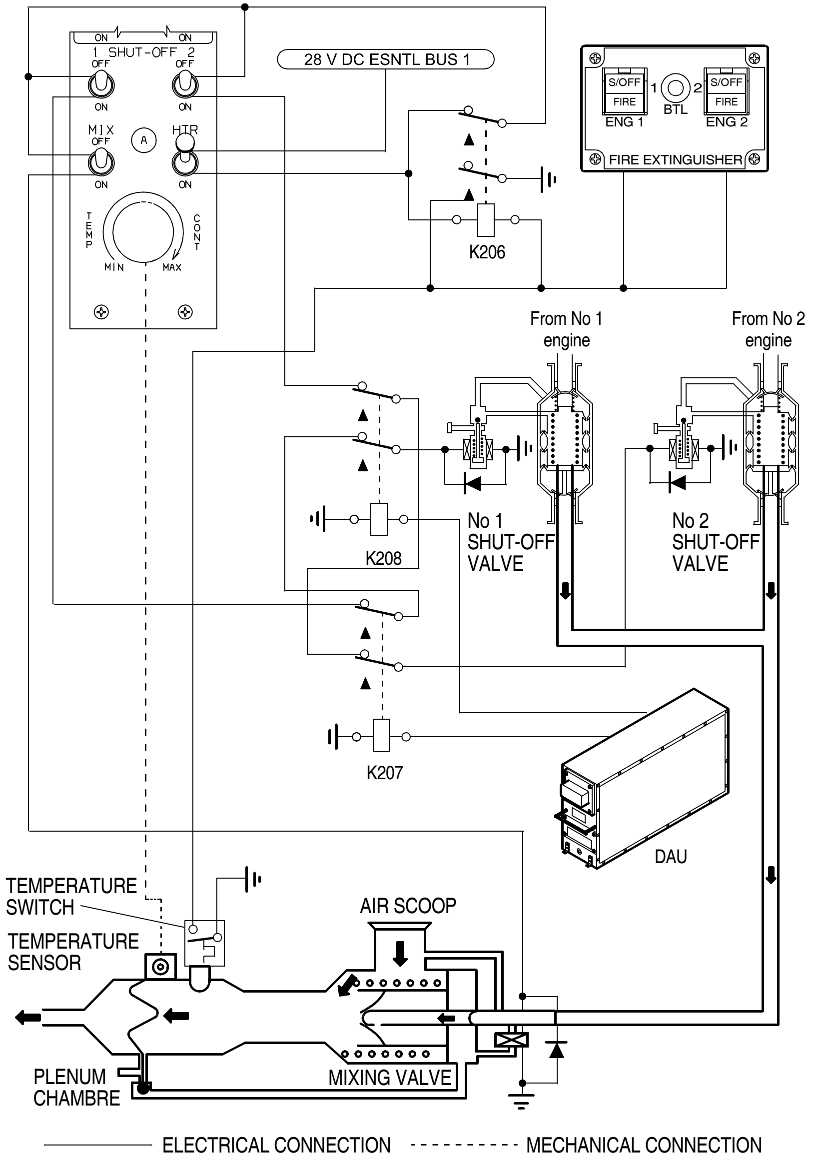

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Figure 7-20 Heating System - Schematic Diagram

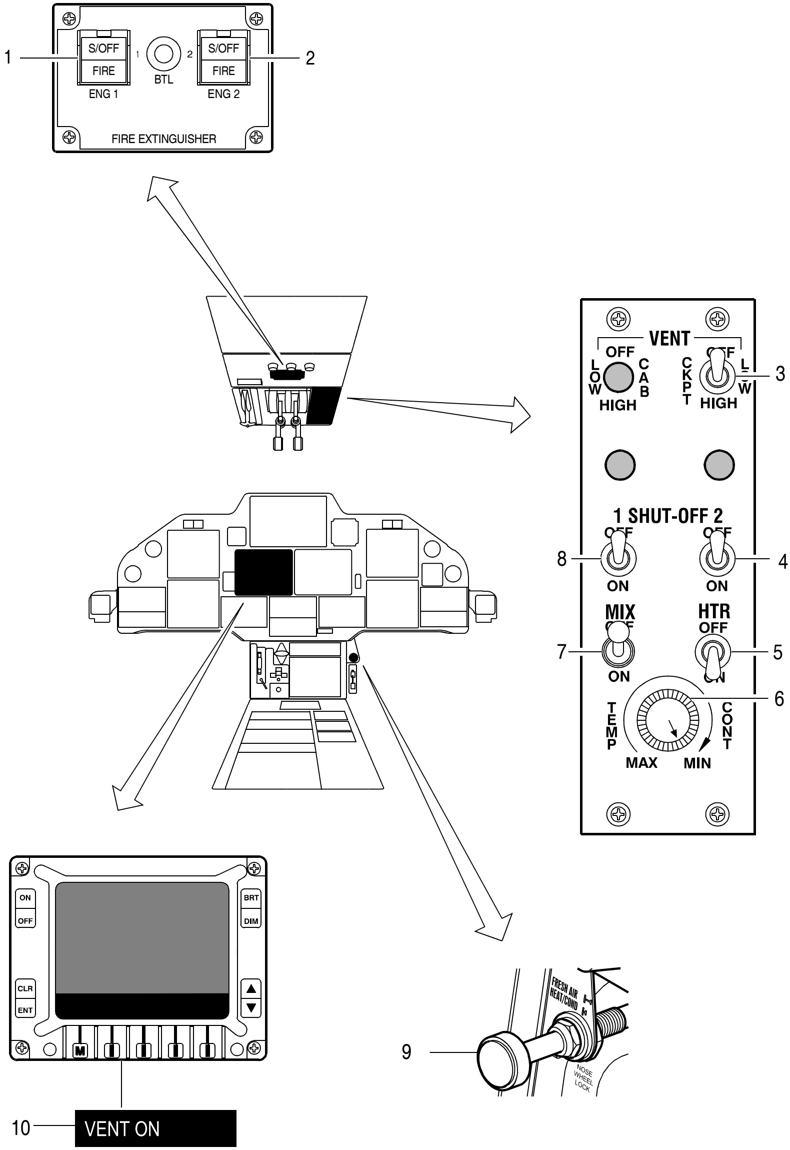

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Figure 7-21 Heating System - Controls and Displays

Key to Figure 7-21

Ref.	Control/Display	Function
1	ENG1 - S/OFF FIRE lighted push-button switch	Refer to Chap 26.
2	ENG2 - S/OFF FIRE lighted push-button switch	Refer to Chap 26.
3	VENT - CKPT switch	OFF - the ventilation fan is de-energized. LOW - the ventilation fan is operating at low speed. HIGH - the ventilation fan is operating at maximum speed.
4	SHUT-OFF 2 switch	OFF - the No 2 shut-off valve is de-energized (valve closed). ON - the No 2 shut-off valve is energized (valve open).
5	HTR switch	OFF - the heating system does not operate. ON - the heating system operates normally. The temperature control circuit operates in the normal mode.
6	TEMP CONT knob	Adjusts the temperature of the mixed air from MIN to MAX.
7	MIX switch	OFF .- the mixing valve is de-energized (valve closed). ON - the mixing valve is energized (valve open).
8	SHUT-OFF 1 switch	OFF - the No 1 shut-off valve is de-energized (valve closed). ON - the No 1 shut-off valve is energized (valve open).

Key to Figure 7-21 (Continued)

Ref.	Control/Display	Function
9	Ventilation knob	HEAT/COND No air from the outside enters the cockpit and the cabin. FRESH AIR - the gate valve of the mixing box is open. Ram air from the outside enter the cockpit and the cabin. When the knob is in an intermediate position the air flow is proportional to the knob position.
10	VENT ON advisory message	Shown when the VENT - CKPT switch is set to LOW or HIGH.

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AUTO FLIGHT

(Chapter 22)

GENERAL

The auto flight system gives the basic stabilization and auto pilot functions to the helicopter. The auto flight system includes:

– The Automatic Flight Control System (AFCS).

The schematic diagram of the AFCS is shown in Figure 7-22.

AFCS CONTROLS AND DISPLAYS

Refer to Figure 7-23.

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

The Automatic Flight Control System (AFCS) mainly consists of a four-axis dual-duplex HAC3000-109 autopilot system.

The system comprises:

– One Flight Control Computer (FCC)
– One AutoPilot Mode Selector (APMS) control panel
– Three pairs of linear actuators
– Four trim actuators (or magnetic brakes)
– Four angular position transducers.

The FCC, installed in the electro-avionic compartment, has two independent channels (Channel 1 and Channel 2). Each channel, powered by an independent power supply unit, has two sections (Section A and Section B). Each section includes a CPU-board and an I/O-board. The two channels communicate with each other via dedicated internal and external data links.

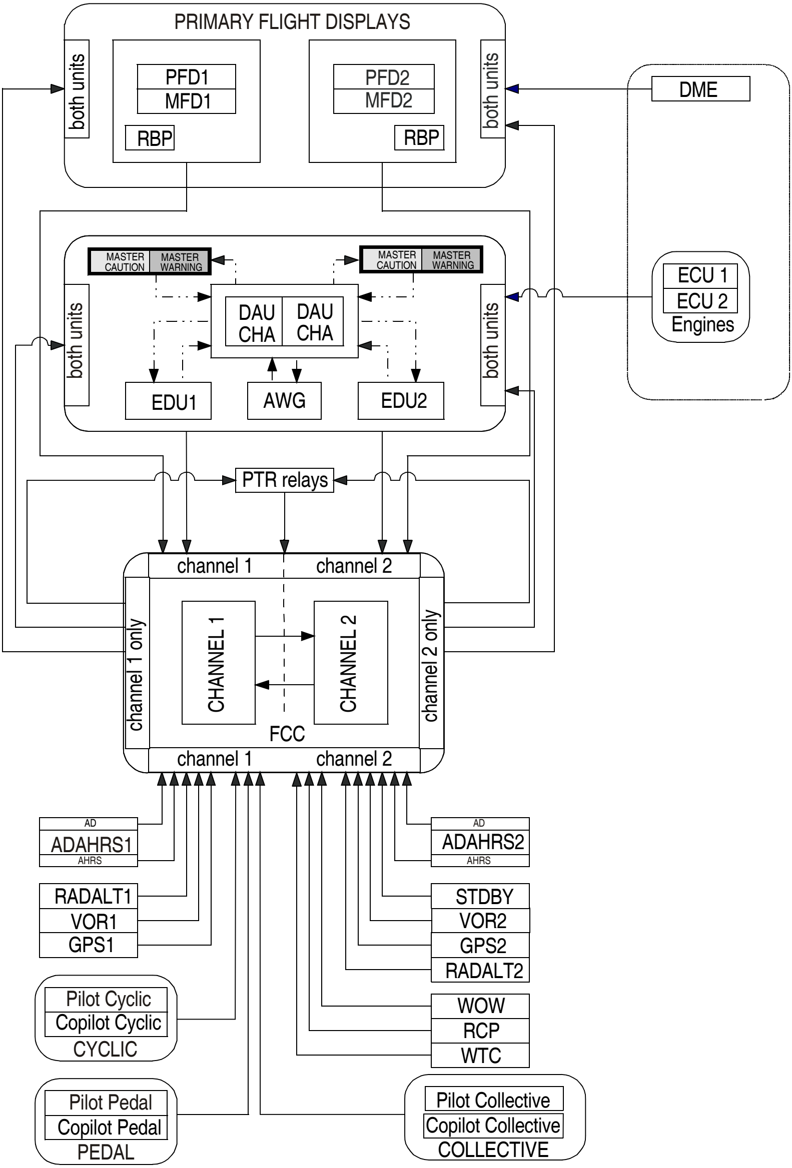

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Figure 7-22 AFCS - Schematic Diagram

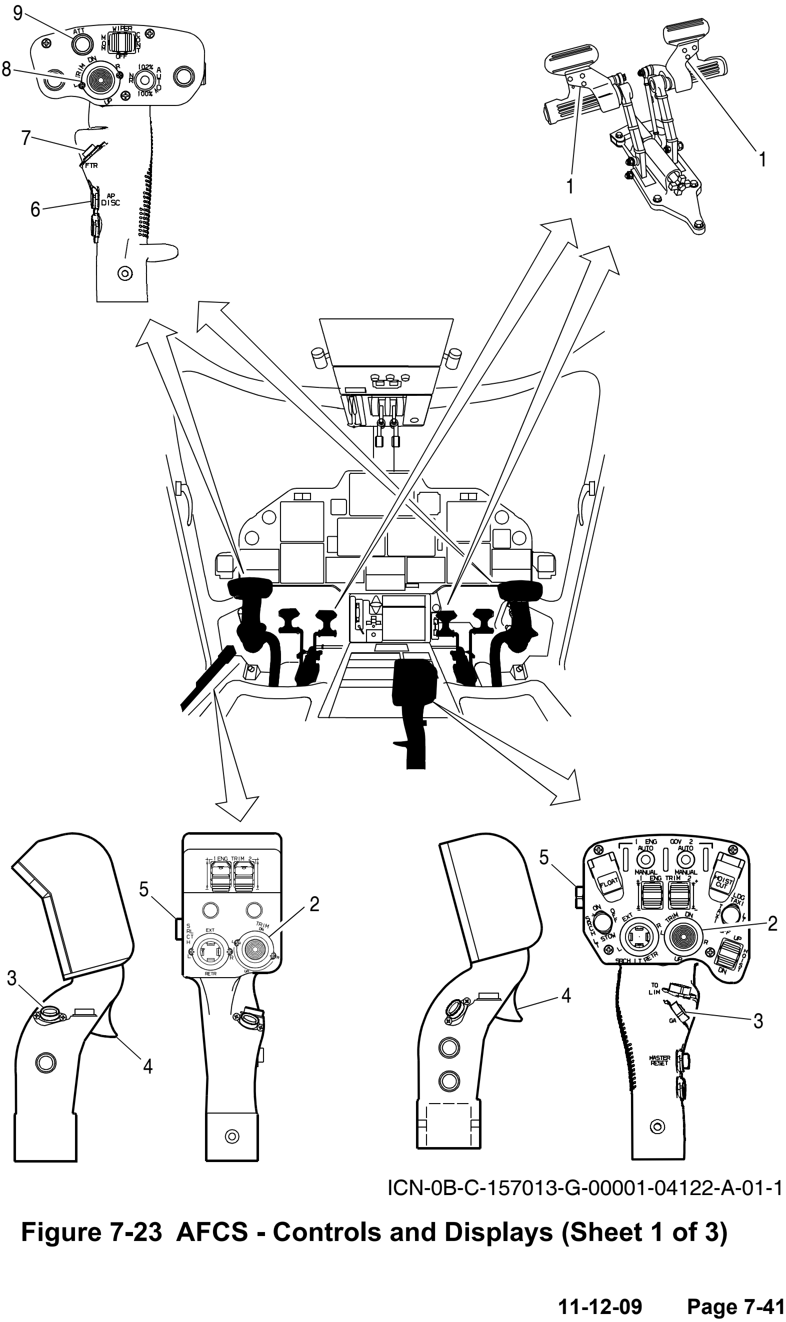

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Figure 7-23 AFCS - Controls and Displays (Sheet 1 of 3)

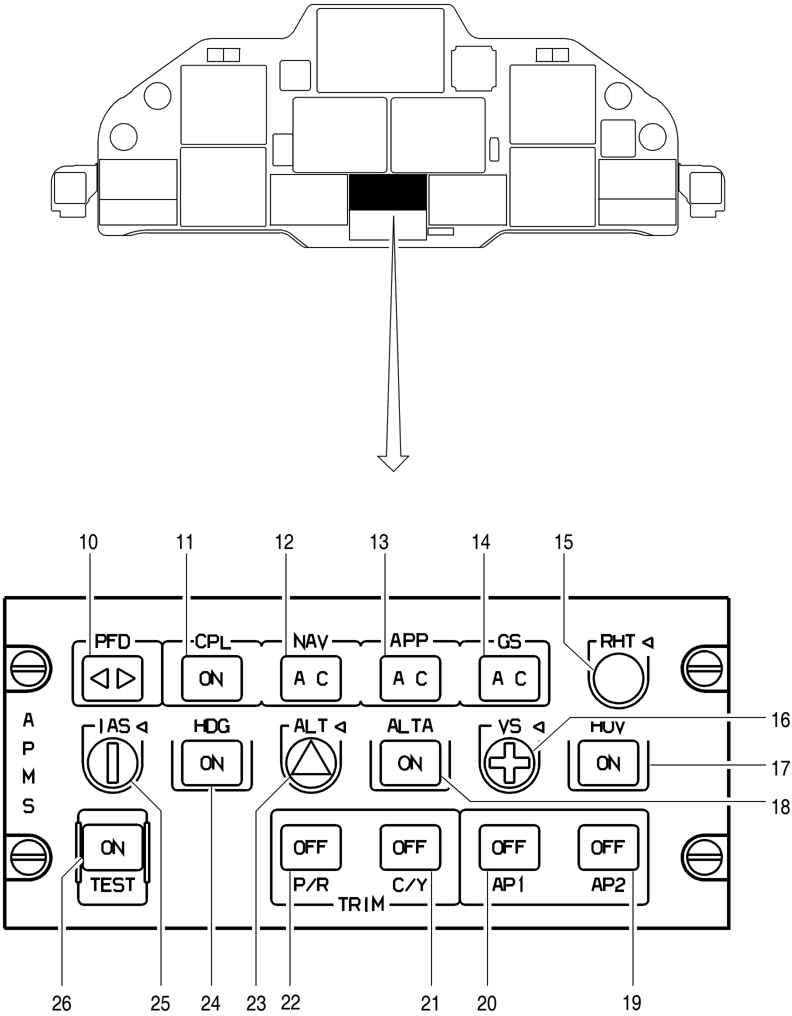

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Figure 7-23 AFCS - Controls and Displays (Sheet 2 of 3)

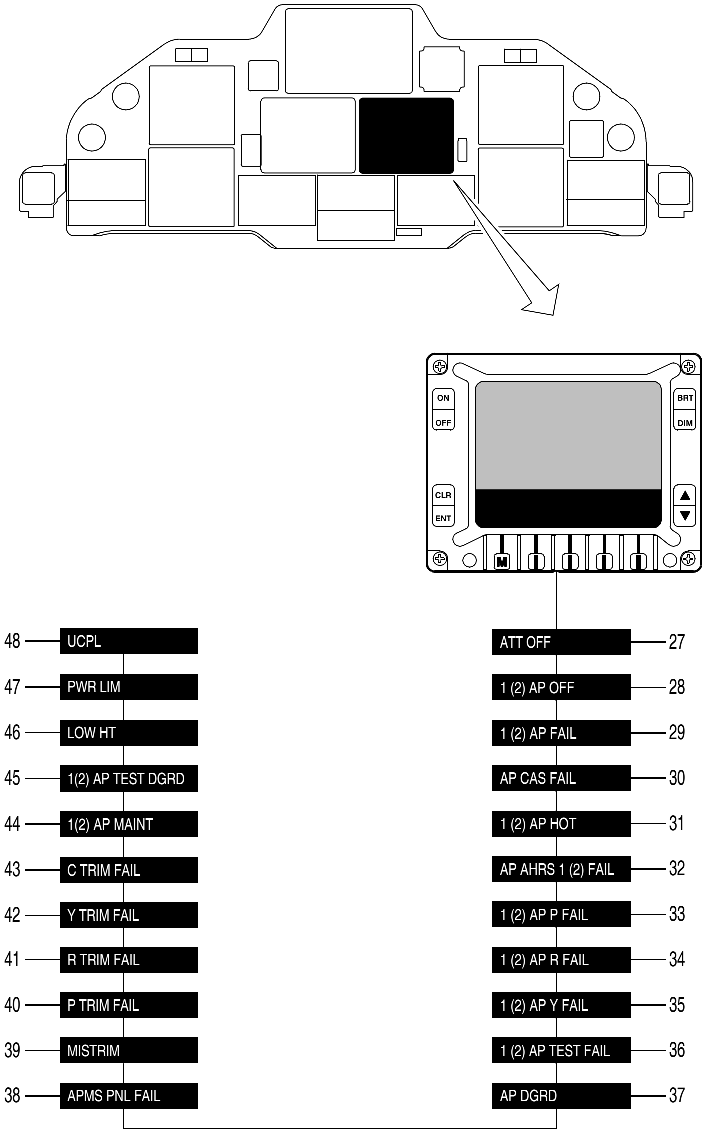

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Figure 7-23 AFCS - Controls and Displays (Sheet 3 of 3)

Key to Figure 7-23

Ref.	Control/Display	Function
1	Force Trim Release (FTR) push-button on pedals	Pushed - Releases the yaw trim actuator and synchronizes the position reference of the tail rotor control in the operation modes ATT, FC and HDG (when engaged) with the current reference.
2	TRIM selector switch	Moved forward, aft, left or right. - When the upper operation modes are disengaged, it provides small attitude changes. When the upper operation modes are engaged, it provides small changes of heading, lateral acceleration, altitude or vertical speed. Pushed - Disengages the upper operation modes previously engaged.
3	GA push-button	Pushed - Engage or disengage the GA operation mode.
4	Collective FTR trigger	Pushed - Releases the collective trim actuator and synchronizes the position reference of the collective control in the operation modes RHT, VS, ALTA, or ALT (when engaged) with the current reference
5	WING LEVEL push-button	Pushed - Engages the WLVL operation mode.
6	AP DISC push-button	Pushed - Disconnects the two FCC channels.
7	Cyclic FTR push-button	Pushed - Releases the pitch and roll trim actuators and synchronizes the position reference of the longitudinal and lateral cyclic control in the operation modes HOV, ALT, HDG, IAS, ATTP or ATTR (when engaged) with the current reference.

Key to Figure 7-23 (Continued)

Ref.	Control/Display	Function
8	TRIM selector switch	Moved forward, aft, left or right - When the upper operation modes are disengaged, it provides small attitude changes. When the upper operation modes are engaged, it provides small changes of heading, lateral acceleration, altitude or vertical speed. Pushed - Updates the references for the HOV operation mode.
9	ATT push-button	Pushed - When an upper operation mode is engaged, it engages the ATT operation mode.
10	PFD push-button	Pushed - Changes over the reception of the navigation data of the AFCS from the pilot EADI and EHSI to the copilot EADI and EHSI and vice-versa.
11	CPL push-button	Pushed - Couples or decouples the AFCS to the operation modes as flight director providing the automatic or manual management of the flight of the helicopter. When the ON legend is on, the AFCS is coupled.
12	NAV push-button	Pushed - Arms/disarms or engages/disengages, according to the condition, the NAV operation mode (when the FMS is set as navigation system) or the VOR operation mode (when a VOR is set). When A letter is on, the operation mode is armed. When C letter is on, the operation mode is engaged.

Key to Figure 7-23 (Continued)

Ref.	Control/Display	Function
13	APP push-button	Pushed - Arms/disarms or engages/disengages, according to the condition, the NLOC operation mode (when the FMS is set as navigation system) or the VAPP operation mode (when a VOR is set) or the LOC operation mode (when an ILS is set). When A letter is on, the operation mode is armed. When C letter is on, the operation mode is engaged.
14	GS push-button	Pushed - Arms/disarms or engages/disengages, according to the condition, the NGS operation mode (when the FMS is set as navigation system) or the GS operation mode (when a VOR is set). As consequence, it arms the NLOC or LOC operation mode respectively. When A letter is on, the operation mode is armed. When C letter is on the operation mode is engaged.
15	RHT knob	Turned - Selects the radio-altimeter altitude that is held in the RHT operation mode. Pushed - Engages or disengages the RHT operation mode. When the triangular symbol is on, the RHT operation mode is engaged. When it is off, the operation mode is disengaged.
16	VS knob	Turned - Selects the vertical speed that is held in the VS operation mode. Pushed - Engages or disengages the VS operation mode. When the cross symbol is on, the VS operation mode is engaged. When it is off, the operation mode is disengaged.

Key to Figure 7-23 (Continued)

Ref.	Control/Display	Function
17	HOV push-button	Pushed - Engages or disengages the HOV operation mode. When the ON legend is on, the operation mode is engaged.
18	ALTA push-button	Pushed - Engages or disengages the ALTA operation mode. When the ON legend is on, the operation mode is engaged.
19	AP2 push-button	Pushed - Engages or disengages the Channel 2 of the FCC. When the OFF legend is on, the Channel 2 is disengaged.
20	AP1 push-button	Pushed - Engages or disengages the Channel 1 of the FCC. When the OFF legend is on, the Channel 1 is disengaged.
21	C/Y TRIM push-button	Pushed - Engages or disengages the automatic function of "Force Trim" of the yaw and collective axes. When the OFF legend is on, the automatic function of "Force Trim" is disengaged.
22	P/R TRIM push-button	Pushed - Engages or disengages the automatic function of "Force Trim" of the pitch and roll axes. When the OFF legend is on, the automatic function of "Force Trim" is disengaged.
23	ALT knob	Turned - Selects the barometric altitude that is held in the ALT operation mode. Pushed - Engages or disengages the ALT operation mode. When the triangular symbol is on, the ALT operation mode is engaged. When it is off, the operation mode is disengaged.

Key to Figure 7-23 (Continued)

Ref.	Control/Display	Function
24	HDG push-button	Pushed - Engages or disengages the HDG operation mode. When the ON legend is on, the operation mode is engaged.
25	IAS knob	Turned - Selects the IAS that is held in the ALT operation mode. Pushed - Engages or disengages the IAS operation mode. When the triangular symbol is on, the IAS operation mode is engaged. When it is off, the operation mode is disengaged.
26	TEST push-button	Pushed - Starts the pre-flight check of the FCC active channels. The push-button operates only when the WOW signal is present. When the ON legend is on, the check of active channels is in progress.
27	ATT OFF caution message	Shown when the ATT operation mode is disengaged, while one or both the FCC channels are engaged.
28	1 (2) AP OFF caution message	Shown when the Channel 1 or Channel 2 of the FCC are disengaged. The OFF legend on the AP1 or AP2 push-buttons of the APMS control panel is on.
29	1 (2) AP FAIL caution message	Shown when the Channel 1 or Channel 2 is failed.
30	AP CAS FAIL caution message	Shown when the AFCS caution messages are not available.
31	1 (2) AP HOT caution message	Shown when the temperature of the Channel 1 or Channel 2 of the AFCS is above the safety limits.
32	AP AHRS 1 (2) FAIL caution message	Shown when one or both the AFCS channels do not receive the data from the associated AHRS.
33	1 (2) AP P FAIL caution message	Shown when the Channel 1 or Channel 2 of the FCC has lost the linear actuator capability in the pitch axis.

Key to Figure 7-23 (Continued)

Ref.	Control/Display	Function
34	1 (2) AP R FAIL caution message	Shown when the Channel 1 or Channel 2 of the FCC has lost the linear actuator capability in the roll axis.
35	1 (2) AP Y FAIL caution message	Shown when the Channel 1 or Channel 2 of the FCC has lost the linear actuator capability in the yaw axis.
36	1 (2) AP TEST FAIL caution message	Shown when the Channel 1 or Channel 2 of the FCC has failed the pre-flight check.
37	AP DGRD caution message	Shown when an AFCS failure has degraded some functions of the AFCS.
38	APMS PNL FAIL caution message	Shown when a failure on the APMS control panel has occurred.
39	MISTRIM caution message	Shown when a linear actuator in the pitch, roll or yaw axes is not centered
40	P TRIM FAIL caution message	Shown when the AFCS has lost the trim actuator capability in the pitch axis
41	R TRIM FAIL caution message	Shown when the AFCS has lost the trim actuator capability in the roll axis.
42	Y TRIM FAIL caution message	Shown when the AFCS has lost the trim actuator capability in the yaw axis.
43	C TRIM FAIL caution message	Shown when the AFCS has lost the trim actuator capability in the collective axis.
44	1 (2) AP MAINT caution message	Shown when a failure is detected on the Channel 1 or Channel 2 of the FCC. This caution message is shown only on ground to inform that maintenance is required.
45	1 (2) AP TEST DGRD caution message	Shown when the FCC completed the pre-flight check with non-critical failures or the test has not been completed.
46	LOW HT caution message	Shown when the height is below 15 feet or in case of excessive height rate approaching ground.
47	PWR LIM caution message	Shown when the collective control override is active due to exceeded Transmission Limit or Autorotation Limit.

Key to Figure 7-23 (Continued)

Ref.	Control/Display	Function
48	UCPL caution message	Shown when the AP is uncoupled.
-	One advisory tone	Advises that an AFCS operation mode is automatically engaged or disengaged.
-	Aural warning message "AIRSPEED"	Vne exceeded.
-	Aural warning message "AUTOPILOT, AUTOPILOT"	Warns that an FCC channel is lost.
-	Aural warning message "ALTITUDE, ALTITUDE"	Warns that the pre-selected radio-altimeter altitude cannot be held in the limits.
-	Aural warning message "ALTITUDE"	The reference Rad-Alt, Bar-Alt or Baro setting has been modified.

The AMPS control panel, installed at the center of the instrument panel, gives the controls for the AFCS functions. The AMPS control panel has 13 push-buttons and four knobs that have different shapes. This help the pilot to identify the knobs.

The three pairs of linear actuators are installed respectively in the longitudinal cyclic control linkage, in the lateral cyclic control linkage and in the tail rotor control linkage. Each pair is mounted in series in the linkage. The Channel 1 controls one actuator of each pair, the Channel 2 controls the other. An LVDT on each actuator measures the movement of the linear actuator shaft.

The four trim actuators are connected in parallel to each control axis (longitudinal cyclic, lateral cyclic, tail rotor pitch) and to the collective control linkage. The four trim actuators create an artificial force feel around an anchor point. The position of the anchor point can be changed with the TRIM selector switches on the collective and cyclic stick grips.

When a linear actuator is above the 30% of its travel for a given time, the related trim actuator automatically changes its anchor point. While the linear actuator returns to its central position the helicopter attitude does not change. The automatic change of the trim actuator anchor point can be disengaged with the C/Y TRIM and P/R TRIM push-buttons on the APMS control panel.

The four angular position transducers measure the rotation of each control axis. Each transducer is connected to a different channel of the FCC.

The AFCS is interfaced to the following systems and equipment of the helicopter:

– The electrical power generation and distribution system provides power supply for the AFCS equipment.
– The Air Data and Attitude Heading Reference System (ADAHRS) provides air data, attitude and heading data.
– The Navigation System (NAVS) provides radio navigation data.
– The Radar altimeter provides the radar height.
– The Primary Flight Display System shows the indications related to the AFCS.

– The Weight on Wheel (WOW) switches to determine the ground or flight conditions of the helicopter.

The AFCS is supplied with electrical power through the circuit breakers that follow:

– FCC CH1 PRIM (28 V DC ESS BUS #2)
– FCC CH1 SEC (28 V DC BATTERY BUS)
– FCC CH2 PRIM (28 V DC ESS BUS #2)
– FCC CH2 SEC (28 V MAIN BUS #2)
– APMS CH1 (28 V DC EMERG BUS #1)
– APMS CH2 (28 V DC ESS BUS #2)
– FORCE TRIM PRIM (28 V DC BATTERY BUS)
– FORCE TRIM SEC (28 V DC MAIN BUS #2)
– FCC CH1 (26 V AC BUS #1)
– FCC CH2 (26 V AC BUS #2).

The main components of the system are shown in Figure 7-24.

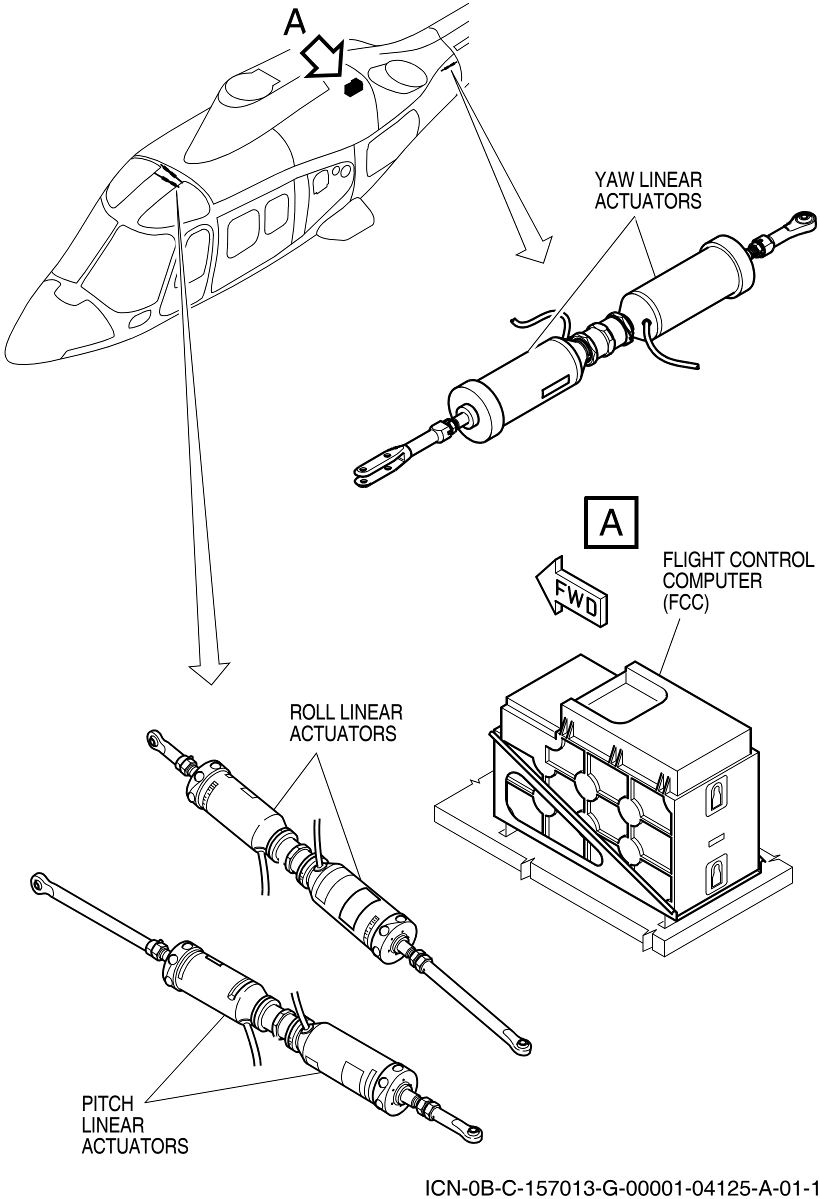

Figure 7-24 AFCS - Components Location (Sheet 1 of 2)

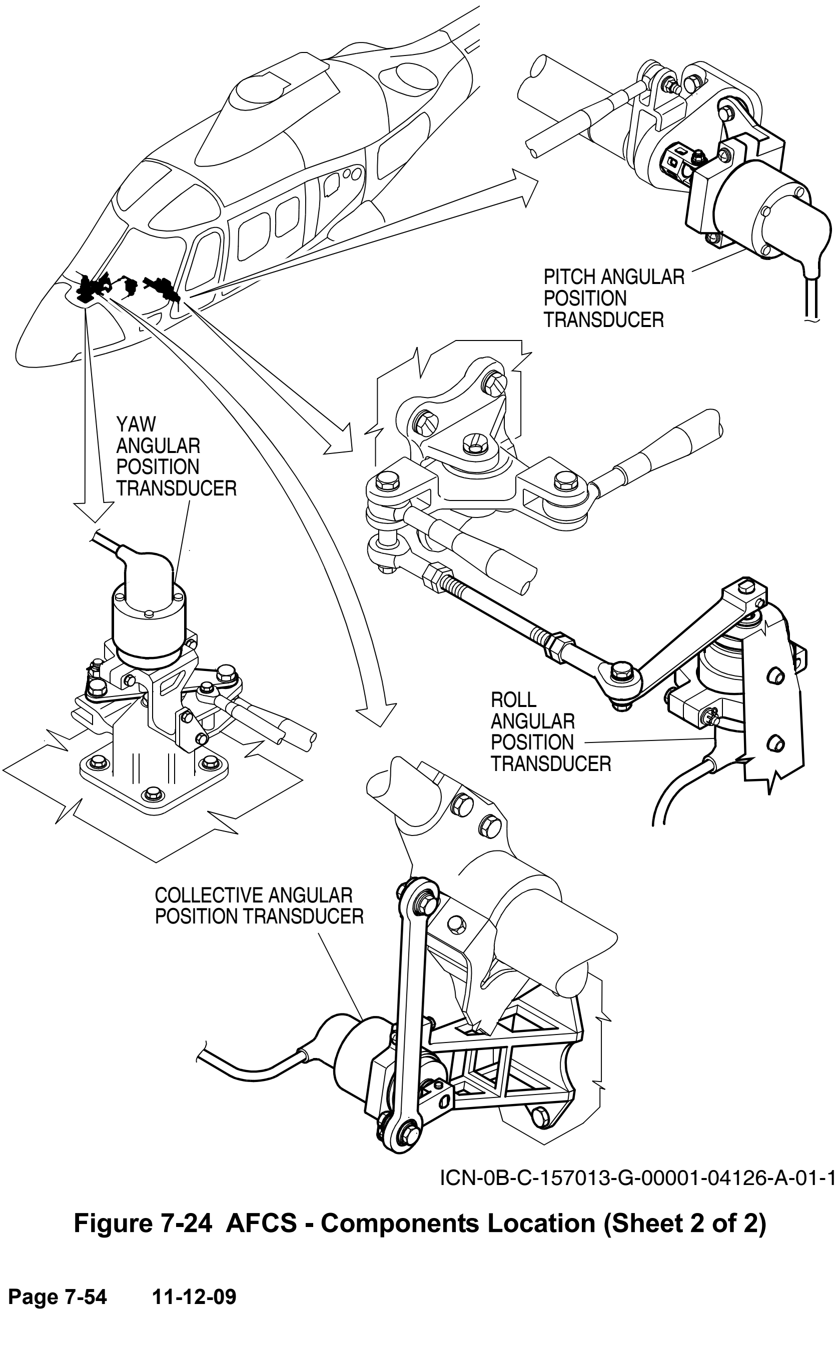

Figure 7-24 AFCS - Components Location (Sheet 2 of 2)

COMMUNICATIONS

(Chapter 23)

GENERAL

The communication system permits the helicopter to communicate with other aircraft and ground stations.

The communication system includes the following sub-systems:

– The Radio Management System (RMS)
– The VHF/AM1 system
– The VHF/AM2 system
– The audio integrating system
– The Satcom Aircell airborne telephone system (with or without handset).

RADIO MANAGEMENT SYSTEM (RMS)

The Radio Management System (RMS) permits the integrated management of the radio-communication and navigation systems of the helicopter.
The RMS includes two Radio Tuning Units (RTU1 and RTU2), that are installed at the center of the instrument panel.
The front panel of each RTU has a Liquid Crystal Display (LCD), on which the control pages of the connected systems and the related data are shown.
Multi-function push-buttons used for the navigation through the Menu pages and for the selection of operation modes of the systems are installed on both sides of the display.
The RTUs can be used to tune each transceiver to the active frequency or preset channel.
The radio signal volume cannot be adjusted using the RTU, this must be done through the audio panels.
When the RTUs are energized, internal checks are performed. If a failure is detected, the display shows the RTU FAILURE warning message in red at full screen.
The RMS is able to manage the systems that follows:

– VHF/AM1 (COM1)
– VHF/AM2 (COM2)
– VOR1/ADF (NAV1/ADF)
– VOR2/ILS (NAV2)
– DME
– ATC transponder (ATC).

The RMS is connected to the various systems through ARINC 429 lines (Figure 7-25).

The RMS directly manages the feedback signals from these systems and shows the malfunction messages on the RTU display in case of signal interruption and the RADIO warning message in red on the bottom right corner of the display in case of signal loss of any system.

In this case the EDU1 shows the RMS MESSAGES caution message to indicate the presence of a message on an RTU display.

In case of an RTU failure, the related failure message is shown on the display of the other RTU.

The RMS is supplied with electrical power from the 28 V dc Number 1 main bus through the RTU1 PRIM and RTU2 SEC circuit breakers, from the 28 V dc Number 2 main bus through the RTU1 SEC circuit breaker and from the 28 V dc ground bus through the RTU2 PRIM circuit breaker.

OPERATION OF RMS

The radio-communication and navigation systems connected to the RMS are controlled through a Menu composed by two pages named "Top Level Page" (TLP) (Figure 7-26).
The second TLP can be selected by pushing the NEXT PAGE pushbutton in the bottom left corner of the RTU.
Each TLP is divided into more parts, each included in a rectangular box and dedicated to a single system.
When a TLP is selected, the system included in the top box is automatically set as active system.

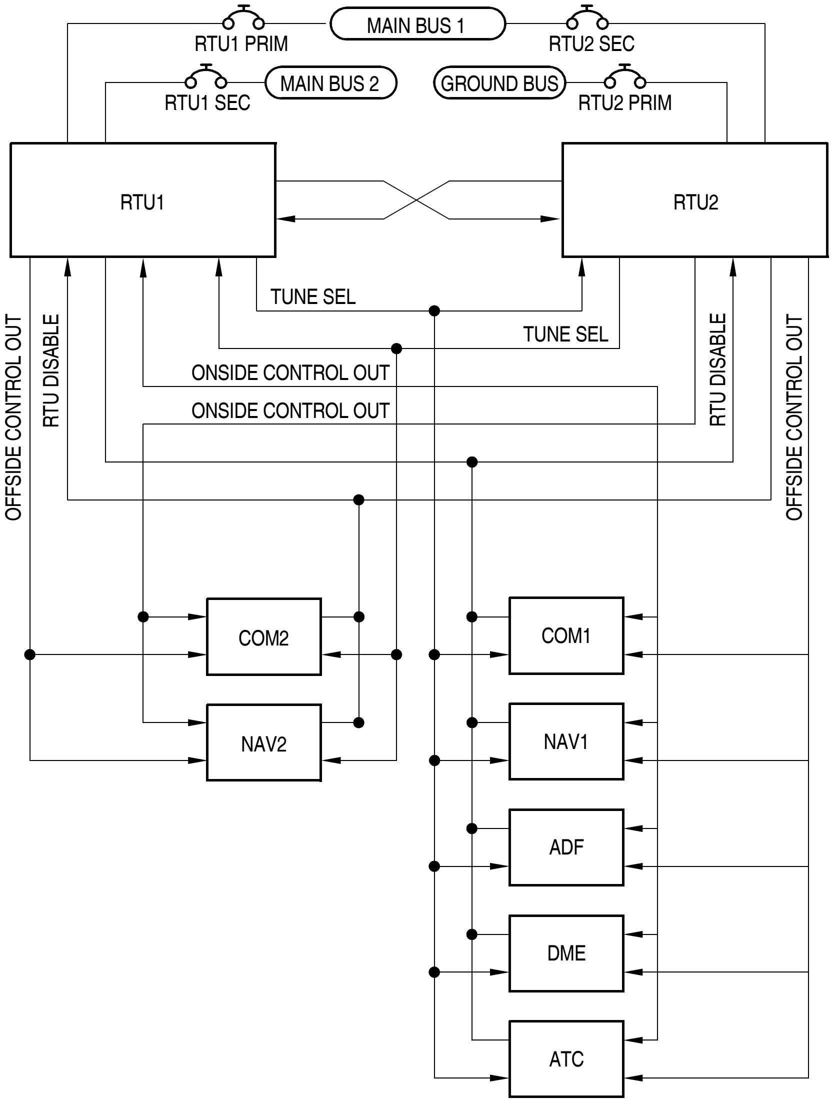

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Figure 7-25 Radio Management System - Block Diagram

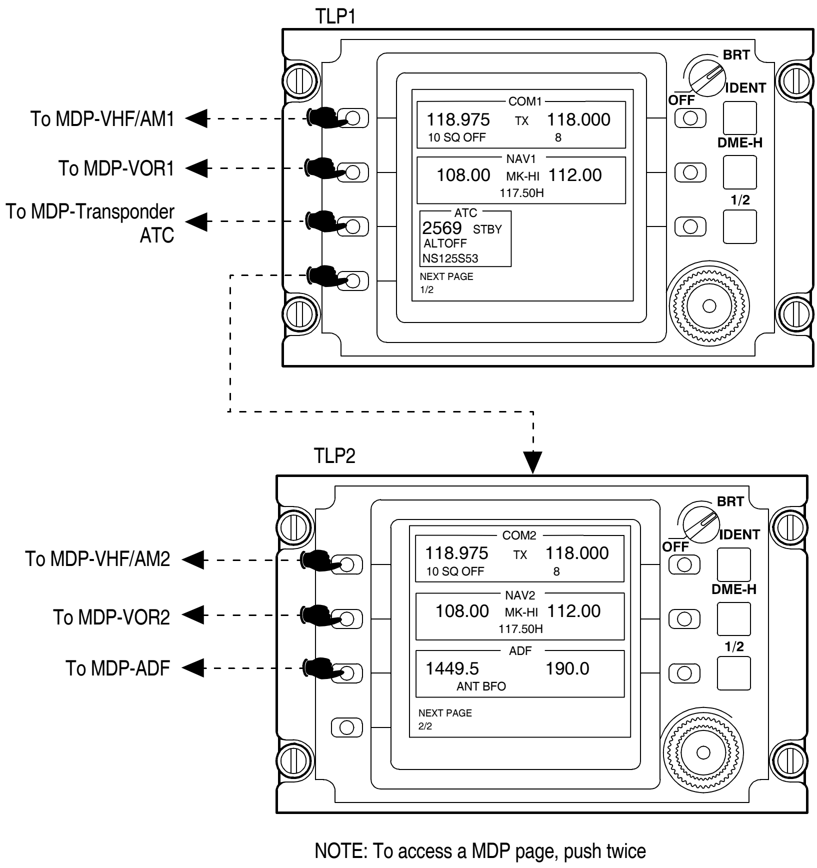

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Figure 7-26 Display Sequence of TLP Pages on RTU Panels

The TLPs control the radio-communication and navigation systems according to the subdivision that follows:

– TLP1: COM1, NAV1, ATC
– TLP2: COM2, NAV2, ADF.

By pushing twice the multi-function push-button on the left of each rectangular box, it is possible to access one or more pages named "Main Display Page" (MDP) that controls the related system. In case of more pages, these can be selected with the NEXT PAGE push-button.

The MDP of each system permits to:

– show the active frequency and/or channel
– show the preset frequency and/or channel
– change the active or preset frequency or channel
– perform the system test
– access the preset page
– access the parameter page (when available)
– select the operation mode of the system, when more options are possible
– turn on and off (or to standby) the transceiver (when available).

By pushing the "1/2" push-button, it is possible to exit from the MDP and come back to the TLP.
For detailed information about the MDP of each system, refer to the related paragraph of the system.
By pushing the multi-function push-buttons installed in the top left and right corners and in the bottom right corner of the RTU at the same time for about 2 s, it is possible to access to Menu page dedicated to the RTU configuration and diagnostic management.

This Menu page permits to access the pages that follow, by pushing the related multi-function push-button:

– RTU Diagnostic Page: shows the information about the RTU software, the power supply voltage and the operating time.
– System Diagnostic Page: shows the input status of the serial busses and the RTU discrete inputs divided into different subpages.

- Radio Diagnostic: permits to select the diagnostic pages related to each transceiver.
- Fault History: shows all the failures recorded during the RTU operation.
- Fault History Reset: permits to cancel the failures recorded in the RTU memory.
- System Configuration Page: permits to show and change the RTU configuration.

If more pages of the same type are present, they can be selected by using the tuning knobs.
By pushing the RETURN or "1/2" push-buttons, it is possible to return to the Menu page.

On the right side of the instrument panel there is the EMERG FREQ push-button switch. A transparent guard prevents its inadvertent operation. In case of failure of both RTU1 and RTU2, it is possible to tune the VHF/AM 2 to the emergency frequency of 121,5 MHz by pushing the EMERG FREQ push-button switch. The related audio communication can be managed by the pilot's audio control panel.

RMS CONTROLS AND DISPLAY

Refer to Figure 7-27.

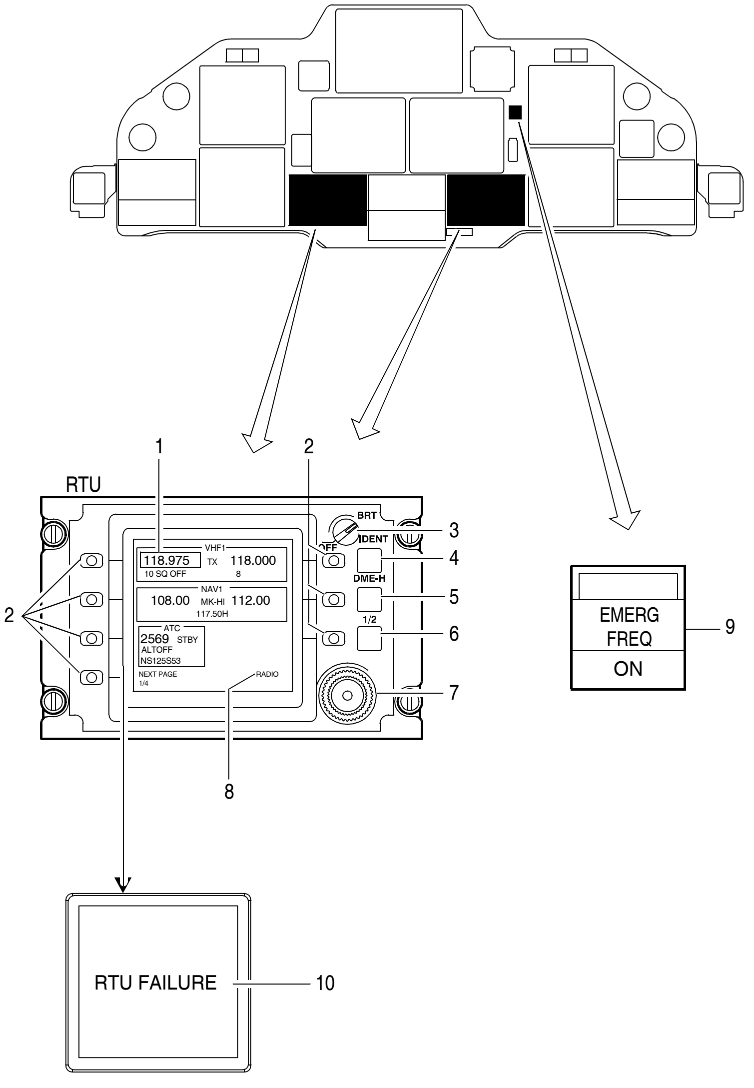

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Figure 7-27 Radio Management System - Controls and Displays

Key to Figure 7-27

Ref.	Control/Display	Function
1	Tune window	It is a white frame that is shown around a value selected with the adjacent multi-function push-button. The value highlighted by the tune window (frequency, channel or code) can be changed with the two tune knobs.
2	Multi-function push-button	The function associated to each push-button depends on the selected page and on the value annunciated on the display next to that push-button. When the value next to the push-button is adjustable (f.e. a frequency), if the push-button is pushed, the value is highlighted by the tune window that permits the value adjustment.
3	Power-up and brightness knob	OFF - The RMS is de-energized. Turned ahead the OFF detent position - The RMS is energized and the brightness is adjusted according to the knob position. When the knob is turned clockwise, the brightness increases; when the knob is turned counterclockwise, the brightness decreases.
4	IDENT push-button	Pushed - Starts the identification process of the helicopter through the ATC transponder.
5	DME-H push-button	Pushed - Holds the frequency, selected on one of the two VOR transceivers for the DME function, always active. The frequency, shown in the related field on the display together with the letter H, is always active even if the VOR frequency is changed, until the push-button is pushed again.
6	"1/2" push-button	Pushed - The display shows the cross-side top-level page.

Key to Figure 7-27 (Continued)

Ref.	Control/Display	Function
7	Tuning knobs	Turned - Adjust the value highlighted by the tune window. When turned clockwise, the value in the tune window increases and vice-versa. For the frequencies the large knob adjusts the MHz part of the frequency, the small knob adjusts the kHz part of the frequency. The increase/decrease is proportional both the angle and the speed of rotation of the knob.
8	RADIO warning message	Shown in case of failure of a transceiver connected to the RMS.
9	EMERG FREQ push-button switch	Pushed - Tune the VHF/AM 2 to the frequency of 121,5 MHz in case of failure of both RTU1 and RTU2.
10	RTU FAILURE warning message	Shown when the internal checks after the RTU power-up detect an RTU failure.

VHF/AM SYSTEMS

The helicopter has two VHF/AM radio communication systems installed (VHF/AM1 system and VHF/AM2 system).
Each VHF/AM system consists of a Rockwell Collins type VHF-4000E transceiver, installed in the forward avionic compartment, and an antenna, installed in the bottom part of the forward fuselage at the center line.
Each system allows two-way AM voice communications in the frequency range from 118,000 through 151,975 MHz (it is only possible to receive in the frequency ranges from 137,000 to 137,975 MHz, from 144,025 to 147,975 MHz, from 149,925 to 150,475 MHz and from 150,825 to 151,975 MHz), in both 8,33 kHz and 25 kHz increments in the frequency range from 118,000 to 136,690 MHz and in only 25 kHz increments in the frequency range from 137,000 to 151,975 MHz.
The selection of operating frequency and the programming of the preset channels are possible through the RTU panels of the RMS.
Each transceiver is interfaced with the RMS through ARINC 429 lines and discrete signals and with the audio integrating system through two channels, one for the reception and the other for the transmission.
The VHF/AM1 system is supplied with electrical power from the 28 V dc Number 1 main bus through the VHF1 PWR circuit breaker. The VHF/AM2 system is supplied with electrical power from the 28 V dc ground bus through the VHF2 PWR circuit breaker.

OPERATION OF VHF/AM SYSTEMS

The systems are controlled through the RTU panels of the RMS, on which the TLP1 page must be selected to access the COM1 (VHF/AM1 system) control display or the TLP2 page to access the COM2 (VHF/AM2 system) control display.
By pushing twice the multi-function push-button on the left side of the panel and adjacent to the COM1 or COM2 control display, it is possible to access the related MDP page and then by pushing the PRESET PAGE push-button the preset page.

The display sequence of the control pages is shown in Figure 7-28.

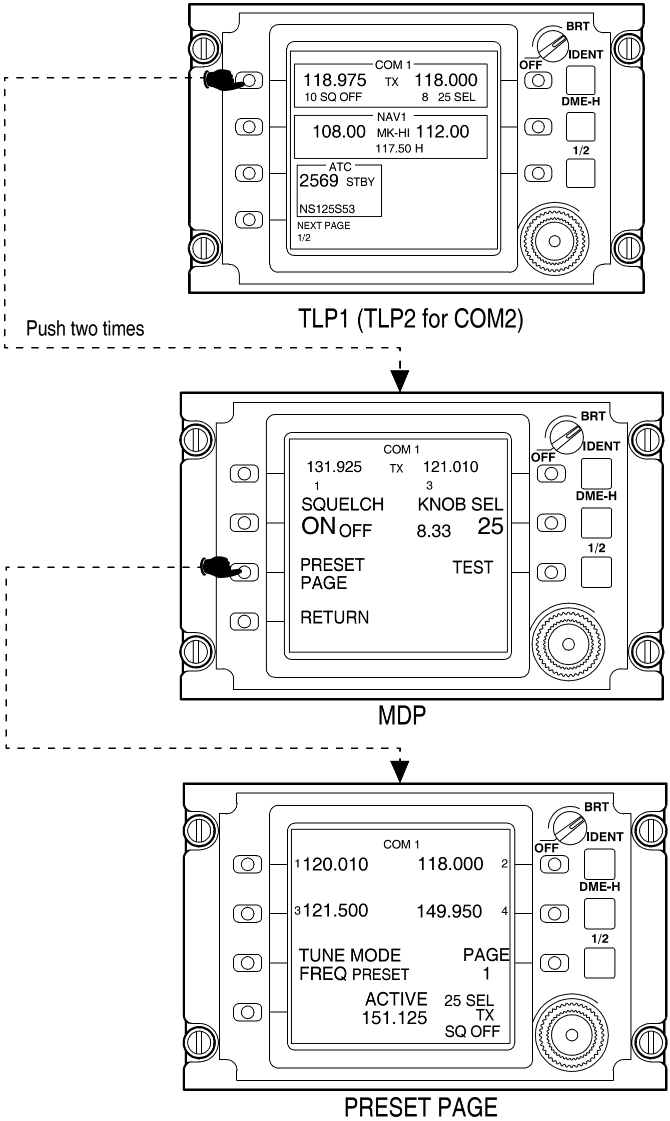

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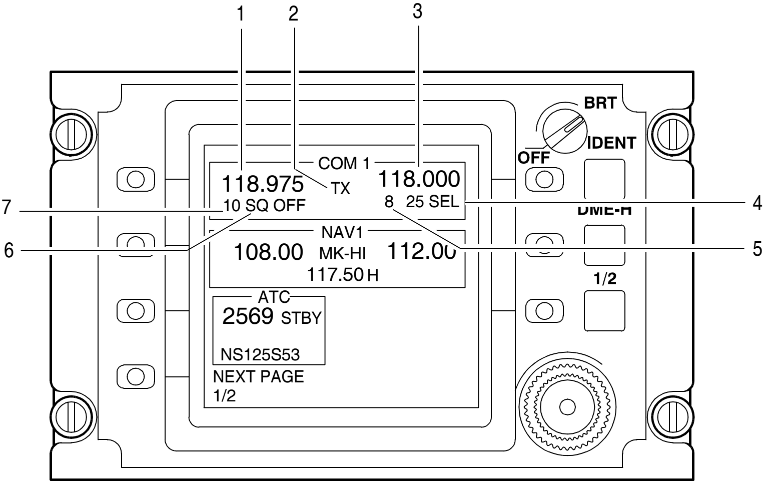
Figure 7-28 Display Sequence of VHF/AM Systems Control Pages

Each VHF/AM system allows communications in transmission or in reception to be performed using the active frequency shown on the left side of the VHF control display, or on one of the 20 preset channels.
A standby frequency, programmed with the adjacent multi-function push-button and the tuning knobs, is shown on the right side of the display.
The active frequency and the standby frequency can be swapped. Both the active frequency and the standby frequency can be directly changed with the tuning knobs of the RTU panel.
All the channels, except channel 20 which is programmed to the permanent emergency frequency of 121,5 MHz, can be programmed by selecting the preset page.

VHF/AM SYSTEMS CONTROLS AND DISPLAYS

Refer to Figure 7-29.

TLP1 PAGE



MAIN DISPLAY PAGE

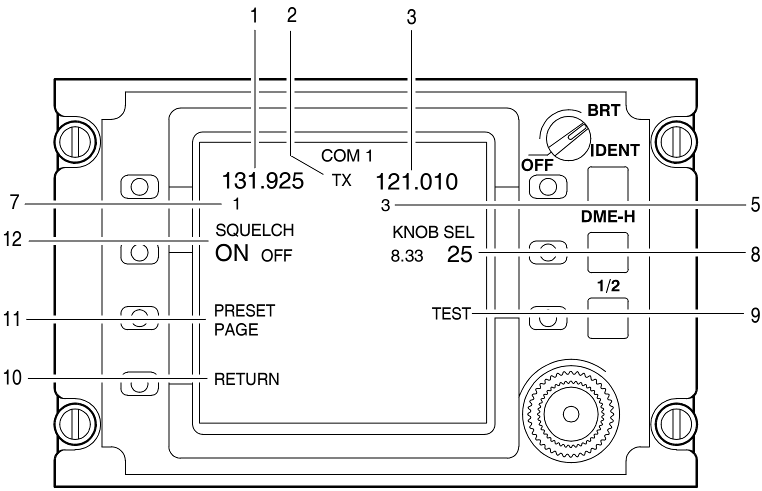


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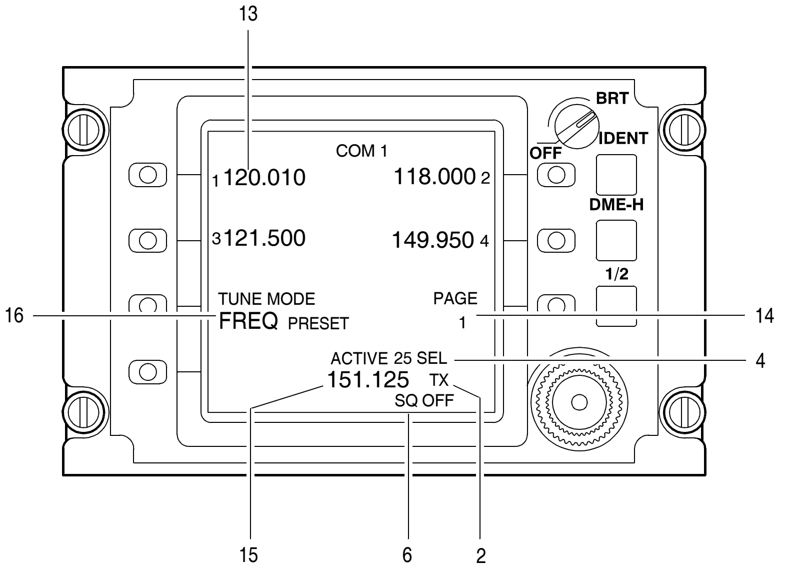
Figure 7-29 VHF/AM Systems - Controls and Displays (Sheet 1 of 2)

PRESET PAGE



NOTE: The controls and displays for VHF/AM1 and VHF/AM2 systems are the same. For controls and display of VHF/AM2 system TLP2 page must be selected.

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Figure 7-29 VHF/AM Systems - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-29

Ref.	Control/Display	Function
1	Active frequency indicator	Shows the active transmission and reception frequency. By pushing the adjacent multi-function push-button, the frequency is highlighted by a white box (tune window), then the frequency can be changed with the tuning knobs in the range from 118,00 and 150,80 MHz.
2	TX annunciator	Shown when the transceiver is transmitting on the active frequency or the active channel.
3	Standby frequency indicator	Shows the standby transmission and reception frequency. By pushing the adjacent multi-function push-button, the frequency is highlighted by a white box (tune window), then the frequency can be changed with the tuning knobs in the range from 118,00 and 150,80 MHz. By pushing again the adjacent multi-function push-button, the frequency is swapped as active frequency.
4	Channel spacing indicator	The "25 SEL" legend is shown when the 25 kHz channel spacing is selected (see ref. 8).
5	Standby channel indicator	Shows the standby channel number selected when the transceiver tuning is set to channel (PRESET) mode (see ref. 16). The standby channel can be swapped as active channel with the adjacent multi-function push-button. 19 preset channels are available (see ref. 13). **NOTE** When the frequency associated with a channel is swapped as active frequency, the standby channel indicator shows the "RCL" legend. The previous active frequency remains associated with the standby channel. This frequency can become again the active frequency by selecting the "RCL" legend in the standby channel indicator.

Key to Figure 7-29 (Continued)

Ref.	Control/Display	Function
6	SQ OFF annunciator	Shown when the squelch function has been removed (SQUELCH OFF) (see ref. 12).
7	Active channel indicator	Shows the active channel number selected when the transceiver tuning is set to channel (PRESET) mode (see ref. 16).
8	KNOB SEL indicator	Shows which channel spacing is selected. The selected spacing is shown in large letters. By pushing the adjacent multi-function push-button, the spacing changes from 8,33 kHz to 25 kHz and vice-versa.
9	TEST identifier	By pushing the adjacent multi-function push-button on the right side, the IBIT of the transceiver is activated. When active, the TEST identifier is shown in large letters. At the end of IBIT the CODE annunciator and a two digit hexadecimal code are shown below the TEST identifier. The code is 00 if no failures are found.
10	"RETURN" legend	By pushing the adjacent multi-function push-button on the left side, the Menu on the display comes back to TLP1 (or TLP2) page.
11	"PRESET PAGE" legend	By pushing the adjacent multi-function push-button on the left side, it is possible to access the preset page for the programming of the preset channels.
12	SQUELCH indicator	Shows if the squelch function is active or not. The active condition is shown in cyan and in large letters. By pushing the adjacent multi-function push-button, the squelch function changes from the ON condition to the OFF condition and vice-versa.

Key to Figure 7-29 (Continued)

Ref.	Control/Display	Function
13	Preset channel programming indicator	Shows the channel number and the associated frequency value. By highlighting the channel number with the adjacent multi-function push-button, it is possible to associate in the memory a frequency value selected with the tuning knobs. 19 channels can be programmed, while channel 20 is permanently programmed as emergency channel on 121,5 MHz. The preset channels are distributed on five pages. The other fields are the same on all the five pages.
14	Page number indicator	Shows which page of the five preset pages is shown. When the indicator is selected by pushing the adjacent multi-function push-button, it is possible to scroll the page number by turning one of the two tuning knobs.
15	Active frequency indicator	Shows the active frequency used on the transceiver.
16	Tuning mode indicator	Shows the active tuning mode. By pushing the adjacent multi-function push-button on the left side, it is possible to change the tuning mode. The active tuning mode is shown in large letters. The selectable tuning modes are as follows: – FREQ: The transceiver is tuned on the selected VHF/AM frequency. – PRESET: The transceiver is tuned on the frequency associated with one of the preset channels.

AUDIO INTEGRATING SYSTEM

The audio integrating system (or InterCommunication System - ICS) (Figure 7-30) permits the communications between the flight crew and the passengers, permits the selection of the radio system to be used to transmit or receive or the selection of the navigation system through which the related signals are to be received.
The system also permits the aural messages sent from the central warning and caution system to be heard.
The primary components of the audio integrating system are the pilot audio control panel and the copilot audio control panel, installed on the instrument panel, the Audio Management Unit (AMU), the remote memory unit, an USB port, four connectors for the connection of the pilot and copilot headsets and a connector for the ground crew headset.
The system also includes the ICS PTT trigger switch, installed on both cyclic stick grips, and two foot switches, installed on both sides of the cockpit.
The audio control panels are independent of each other so that any failures do not compromise the operation of the other communication station. The audio control panels have eight transceiver receive turning switches in the top of the front panel. Each receive turning switch can be selected with the transmit selector. When selected, the related transceiver can operate and the indicator light above the receive turning switch comes on. Eight receiver receive turning switches in the bottom of the front panel let the audio signals of the related navigation system be heard in the headset. The front panel also has the mode switch, the VOX knob that lets the threshold of the automatic control circuit of the microphone be adjusted, the ICS/TX switch, the ISO CALL push-button that enables or disable the Passengers Isolation mode, the RX and ICS knobs that let the audio level of radio and interphone communications be adjusted and the TX, ISO and CALL indicator lights.
The AMU, installed in the forward avionic compartment, is the interface between all the audio control panels, the communication and navigation systems and the central warning and caution system.
The AMU is also connected to the passenger intercom amplifier and the passenger audio control panel that control the ICS in the cabin.
The remote memory unit, installed in the forward avionic compartment above the AMU, is a device that stores the system configuration data.

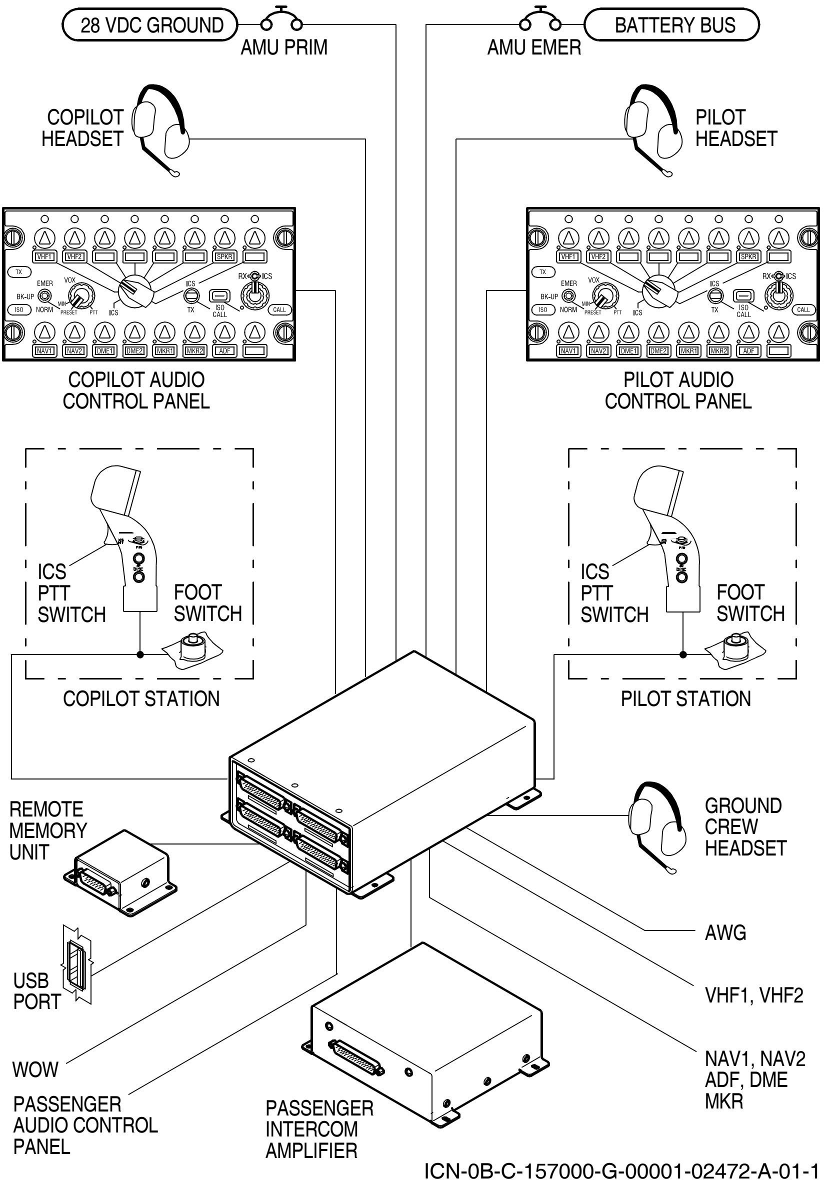

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Figure 7-30 Audio Integrating System - Schematic Diagram

This lets the AMU be replaced without the need of downloading a new configuration file. The configuration of the system can be done with a PC connected to the USB port and an applicable configuration management software.
The audio integrating system is supplied with electrical power from the 28 V dc battery bus through the AMU EMER circuit breaker and from the 28 V dc ground bus through the AMU PRIM circuit breaker.

OPERATION OF THE AUDIO INTEGRATING SYSTEM

The audio integrating system can operate in one of the following modes that can be selected with the mode switch:

– Normal mode: all system functions are available to the flight crew and the passengers.
– Backup mode: all system functions are available to the flight crew, while are disabled to the passengers.
– Emergency mode: this mode allows the flight crew to maintain a specific level of operation. The internal circuitry that processes the digital signals is bypassed and the audio levels are fixed.

When the cockpit and cabin interphone lines are not in communication and the passengers wish to speak to the pilots, the passengers can do their request by pushing a call push-button in the cabin.
When the call push-button is pushed, the CALL indicator light flashes to advise the pilot of the passenger request.
After the pilot has reset the normal operation mode, the CALL indicator light stop to blink.
When the system is energized, a Power-On BIT (PBIT) is performed. Also a continuous BIT is performed during the operation of the system. The results of PBIT and CBIT are stored in the non-volatile memory of AMU.

AUDIO INTEGRATING SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-31.

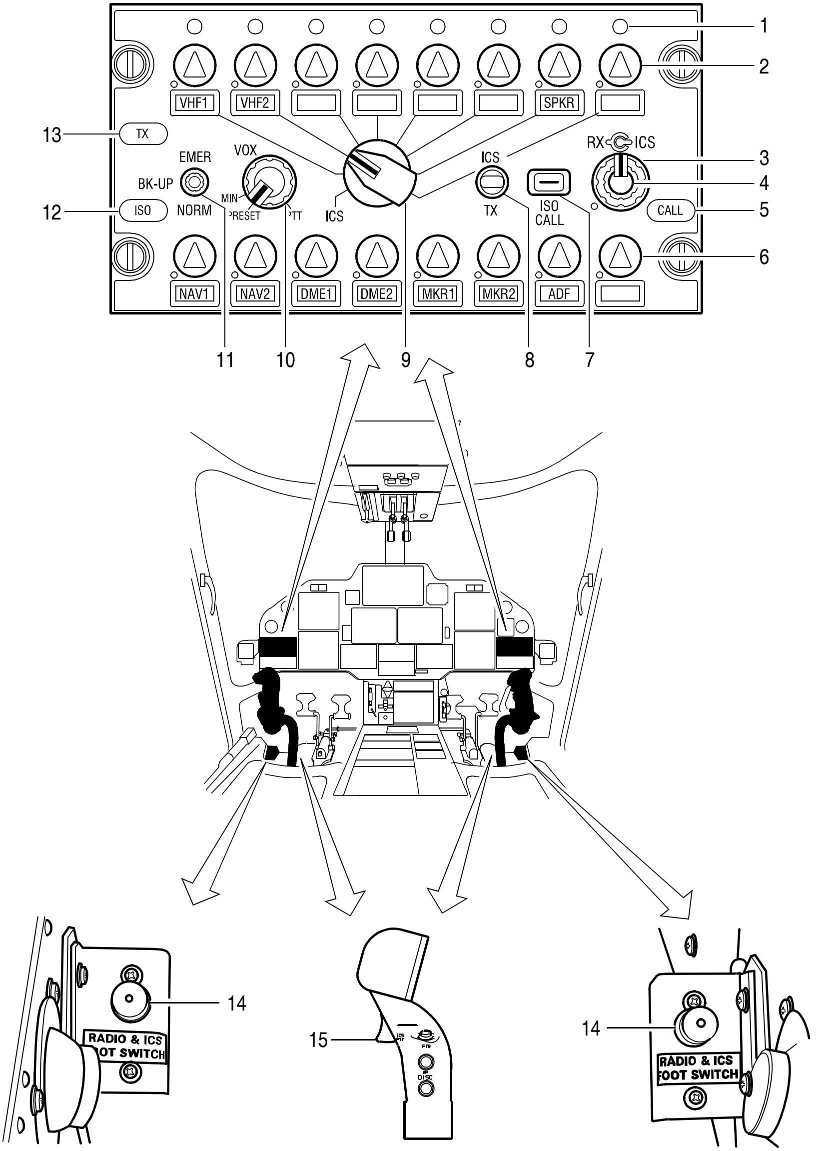

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Figure 7-31 Audio Integrating System - Controls and Displays

Key to Figure 7-31

Ref.	Control/Display	Function
1	Transmit select indicator light (eight)	On - Indicates that the transmit selector is set to the related transceiver.
2	Transceiver receive turning switch (eight)	Turned fully counterclockwise (off position) - The audio signal of the related transceiver is not received in the headset. Turned (on position) - The audio signal of the related transceiver is received in the headset at the minimum audio level. Pulled and turned - Adjusts the audio level of the signals received from the related transceiver. Turned clockwise the audio level increases, while turned counterclockwise the audio level decreases. When pushed again, the audio level is automatically set at the minimum.
3	RX knob (external knob)	Turned - Adjusts the audio level of all the selected transceivers and receivers,
4	ICS knob (internal knob)	Turned - Adjusts the audio level of the interphone communications received in the headset, separately from the audio level of the transceivers and receivers.
5	CALL indicator light	Flashes - Indicates that it is necessary for a passenger to speak with the flight crew. On - Indicates that the cockpit and cabin interphone lines are connected.
6	Receiver receive turning switch (eight)	Turned fully counterclockwise (off position) - The audio signal of the related receiver is not received in the headset. Turned (on position) - The audio signal of the related receiver is received in the headset at the minimum audio level. Pulled and turned - Adjusts the audio level of the signals received from the related receiver. Turned clockwise the audio level increases, while turned counterclockwise the audio level decreases. When pushed again, the audio level is automatically set at the minimum.

Key to Figure 7-31 (Continued)

Ref.	Control/Display	Function
7	ISO CALL push-button	Momentarily pushed - Enables or disables the Passengers Isolation mode. The ISO indicator light (see ref. 12) comes on or goes off.
8	ICS/TX switch	ICS (Momentary position) - Enables the interphone communication. TX (Momentary position) - Enables the transmission of the selected transceiver.
9	Transmit selector	ICS - No system can be used to make a transmission. VHF1 - Selects the VHF/AM1 system to be used for a transmission. VHF2 - Selects the VHF/AM2 system to be used for a transmission. SPKR - Selects the function of the passenger communication through the internal loudspeakers.
10	VOX knob	Adjusts the threshold of the automatic control circuit of the microphone, when the interphone communication is in the VOX mode (VOX knob in a position between MIN a PTT). PRESET - Is the detent position of the VOX knob when it is turned fully counterclockwise. MIN - The threshold level is at the minimum. PTT - The threshold level is at the maximum. The system is in the PTT mode.
11	Mode switch	NORM - Selects the normal operation of the system. In this mode all system functions are available. BK-UP - Selects the backup operation mode. All system functions are available to the flight crew, while are disabled for the passengers. EMER - Selects the Emergency operation mode that allows the flight crew to maintain a specific level of operation. The pilot maintains the operation of VHF1, NAV1 and audio alarm. The copilot maintains the operation of VHF2, NAV2 and audio alarm.

Key to Figure 7-31 (Continued)

Ref.	Control/Display	Function
12	ISO indicator light	On - Indicates that the Passengers Isolation mode is active.
13	TX indicator light	On - Indicates that a transmitter operates.
14	RADIO & ICS foot switch	Pushed - When the transmit selector is in the ICS position, it permits the radio and interphone communications.
15	ICS PTT trigger switch	Set at the first detent - Permits the interphone communications. Set at the second detent - Permits the radio communications through the system set with the transmit selector.

SATCOM AIRCELL AIRBORNE TELEPHONE SYSTEM

The Satcom Aircell system provides access to a range of two-ways communication options from the helicopter, including voice and data applications. These services are available when the aircraft is in the air or on the ground. The Satcom Aircell system accesses the Iridium Satellite Network of 66 Low-Earth Orbit (LEO) satellites and operates in the frequency range of 1616 MHz to 1625.5 MHz.

The primary components of the Satcom Aircell system are:

- The antenna
- The transceiver
- The satcom keyboard
- The SATCOM CALL light.

The main components of the Satcom Aircell system are shown in Figure 7-31A.

The antenna is installed at the base of the tail fin. A radome gives protection against rain, ice and lightning strikes. The antenna is connected to the transceiver with a coaxial cable.

The transceiver is installed in the baggage compartment. Six screws attach the transceiver to the two mounting rails. All the electrical connectors are on the front face of the unit.

The satcom keyboard is installed on the interseat console. It lets the pilots to make telephone calls. Function buttons, an alphanumeric keyboard, a display and two LEDs provide controls and indications for all the system functions.

The SATCOM CALL light is installed on the instrument panel. The SATCOM CALL indication flashes when a telephone call is coming.

The Satcom Aircell system is supplied with electrical power through the circuit breaker that follows:

- PHONE PWR
- PHONE CTRL.

SATCOM AIRCELL SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-31B.

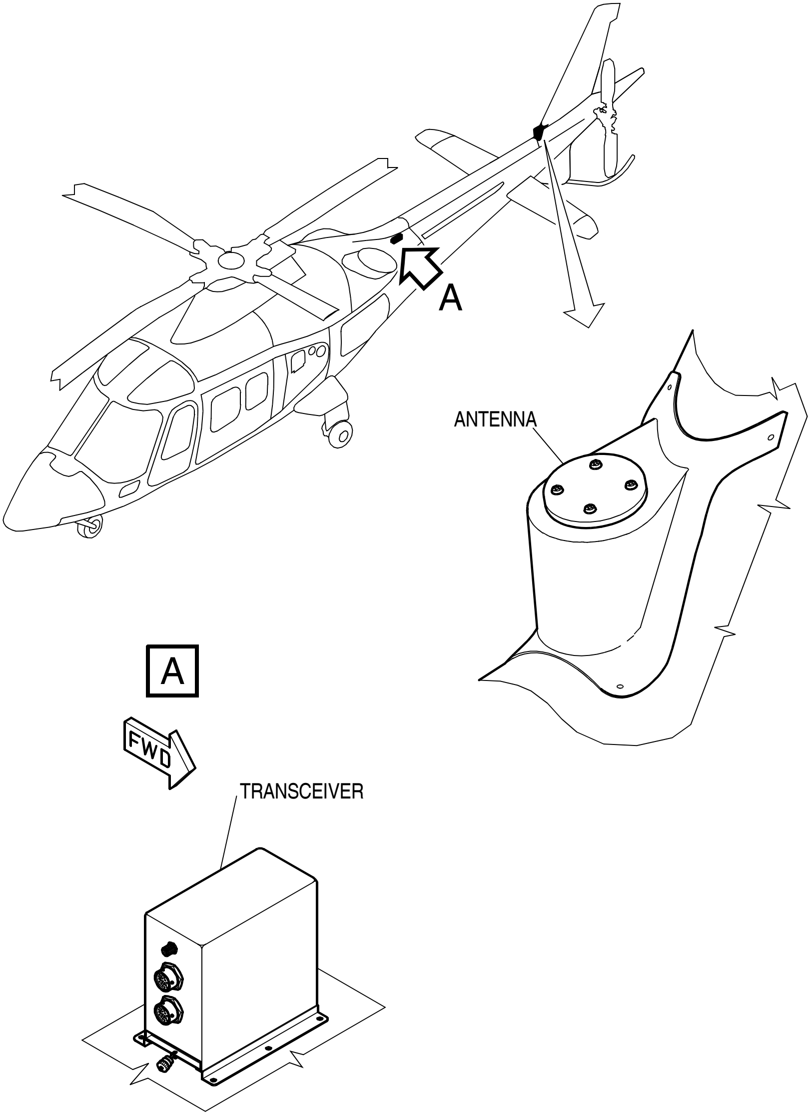

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Figure 7-31A Satcom Aircell System - Component location (Sheet 1 of 2)

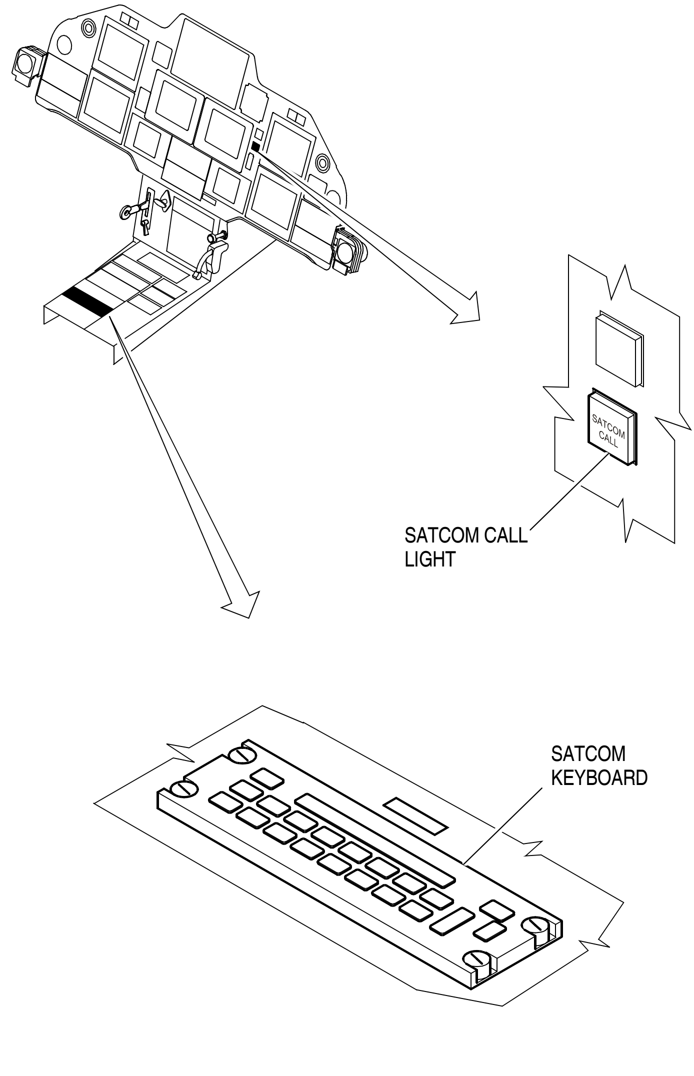

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Figure 7-31A Satcom Aircell System - Component location (Sheet 2 of 2)

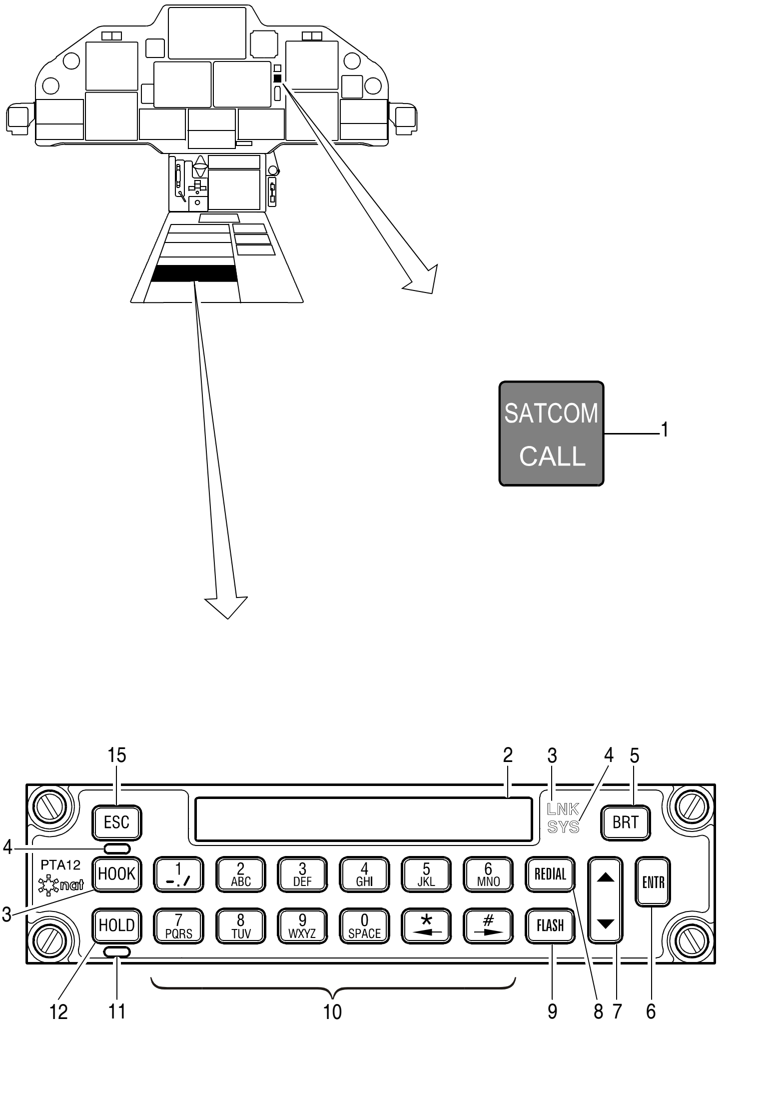

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Figure 7-31B Satcom Aircell System - Controls and Displays

Key to Figure 7-31A

Ref.	Control/Display	Function
1	SATCOM CALL light	Flashes when a telephone call is coming.
2	Display	Shows the messages related to the selections done with the buttons.
3	LNK indication	The LNK indication is visible (green light) when the Satcom link is set up. The LNK indication is not visible if the Satcom link is down.
4	SYS indication	The SYS indication is visible (green light) when the Satcom system is ready. The SYS indication is not visible when the Satcom system is busy.
5	BRT button	Controls the display brightness. Each time the BRT button is pushed, the display brightness increases. When the system is in the Default Power-Up mode, the display shows the message BRIGHTNESS and the brightness value (1 thru 8). These data are shown all the time while you operate the button. If you push again the button when the maximum brightness (8) is set, the brightness goes back to 1.
6	ENTR button	(Default Power-Up mode) If pushed when the system is in this operation mode, it puts the system in the Edit mode. If pushed after a phone number is set, it puts the system in the Call mode and does the selection of the phone number. (Recall mode) If pushed after a phone number is recalled, it puts the system in the Call mode and does the selection of the phone number. (Edit mode) If pushed when the system is in this operation mode, it records the data written in the display.

Key to Figure 7-31A

Ref.	Control/Display	Function
7	VOL button	The VOL button is a momentary switch used to increase an decrease the phones output volume. The volume increases or decreases by pressing the up or down arrow. There are a total of 32 steps from the minimum to the maximum volume. By holding either the up or down arrow for more than one second, the volume increases or decreases at a faster rate. The last selected value is stored in the internal memory and is retained upon power down.
8	REDIAL button	Recalls and dials the last number dialed. The number is stored in the internal memory and is retained upon power down.
9	FLASH button	Interrupts a call and allows the user to deal with a second call.
10	Alphanumeric keyboard	The number and symbol buttons on the keyboard are used to dial the required telephone number.
11	HOLD led	The led is on to indicate that a call is on hold. The led is off when the call is returned to active status.
12	HOLD button	Allows the operator to put a call on hold by muting the microphone and phones audio. This is possible only when the unit is active.
13	HOOK button	Starts or receives a call. This corresponds to lifting or hang up the handset of a telephone. Each time the HOOK button is pushed, the hook switch toggles on (active) or off (inactive).
14	HOOK led	The led is off when the unit is inactive and is on when the unit is active. When the unit is inactive the led flashes to indicate an incoming call.

Key to Figure 7-31B

Ref.	Control/Display	Function
15	ESC button	(Default Power-Up mode) When pushed, it cancels the last operation done. (Recall and Edit modes) When pushed, it cancels the last operation done and the system goes back to the Default Power-Up mode.

SATCOM AIRCELL AIRBORNE TELEPHONE SYSTEM WITH HANDSET

The Satcom Aircell system provides access to a range of two-ways communication options from the helicopter, including voice and data applications. These services are available when the aircraft is in the air or on the ground. The Satcom Aircell system accesses the Iridium Satellite Network of 66 Low-Earth Orbit (LEO) satellites and operates in the frequency range of 1616 MHz to 1625.5 MHz.

The primary components of the Satcom Aircell system are:

– The antenna
– The transceiver
– The telephone adapter
– The cordless handset
– The cradle base
– The satcom keyboard
– The PES/SATCOM control panel.

The main components of the Satcom Aircell system with handset are shown in Figure 7-31C.

The antenna is installed at the base of the tail fin. A radome gives protection against rain, ice and lightning strikes. The antenna is connected to the transceiver with a coaxial cable.

The transceiver is installed in the baggage compartment. Six screws attach the transceiver to the two mounting rails. All the electrical connectors are on the front face of the unit.

The satcom keyboard is installed on the interseat console. It lets the pilots to make telephone calls. Function buttons, an alphanumeric keyboard and two LEDs provide controls and indications for all the system functions.

The PES/SATCOM control panel, installed on the instrument panel, lets the pilot to control the Satcom Aircell system.

The cordless handset is installed in the passenger cabin and can be used only by passengers. The cordless handset provides voice communications, control and programming of the Satcom Aircell system.
The cradle base holds the handset in position when it is not in use.

The CALL light is installed in the passenger cabin. The CALL indication flashes when a telephone call is coming.

The Satcom Aircell system is supplied with electrical power through the circuit breaker that follows:

– SATCOM (28 V DC BATTERY BUS).

SATCOM AIRCELL SYSTEM (WITH HANDSET) CONTROLS AND DISPLAYS

Refer to Figure 7-31D.

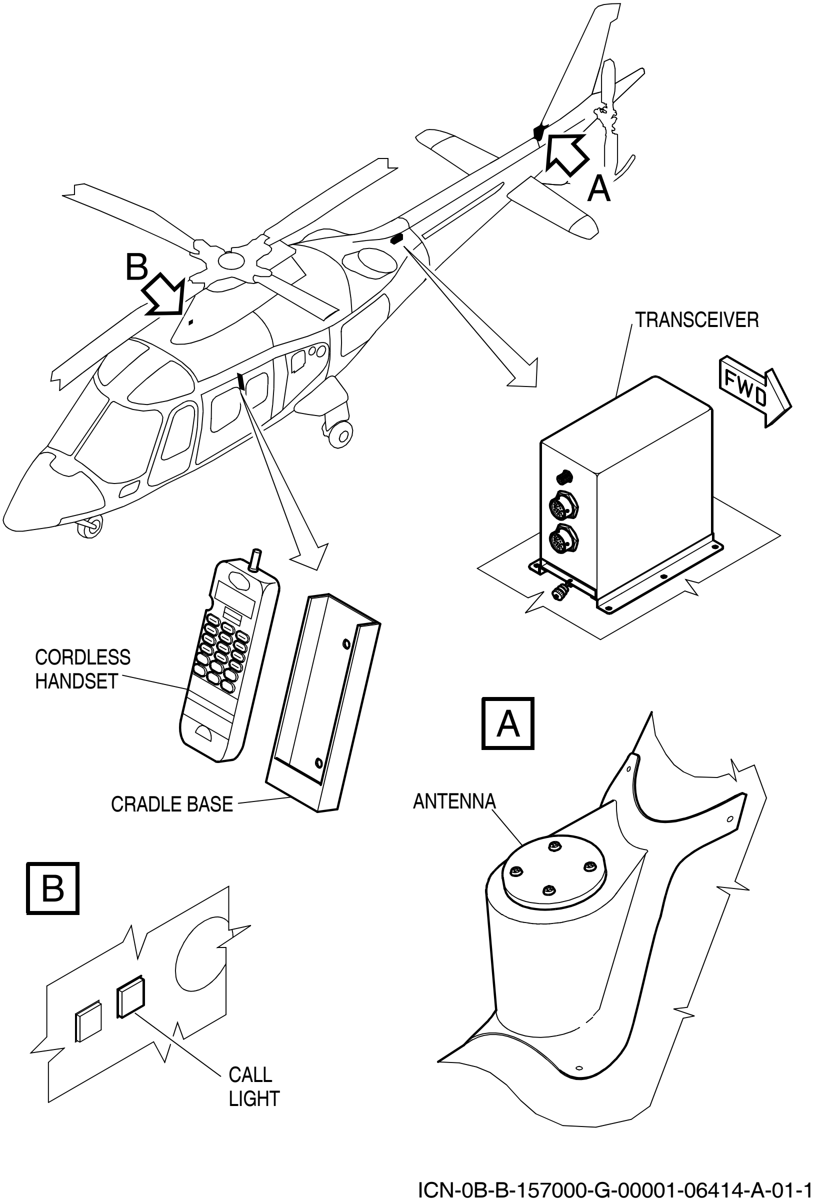

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Figure 7-31C Satcom Aircell with Handset - Component location (Sheet 1 of 2)

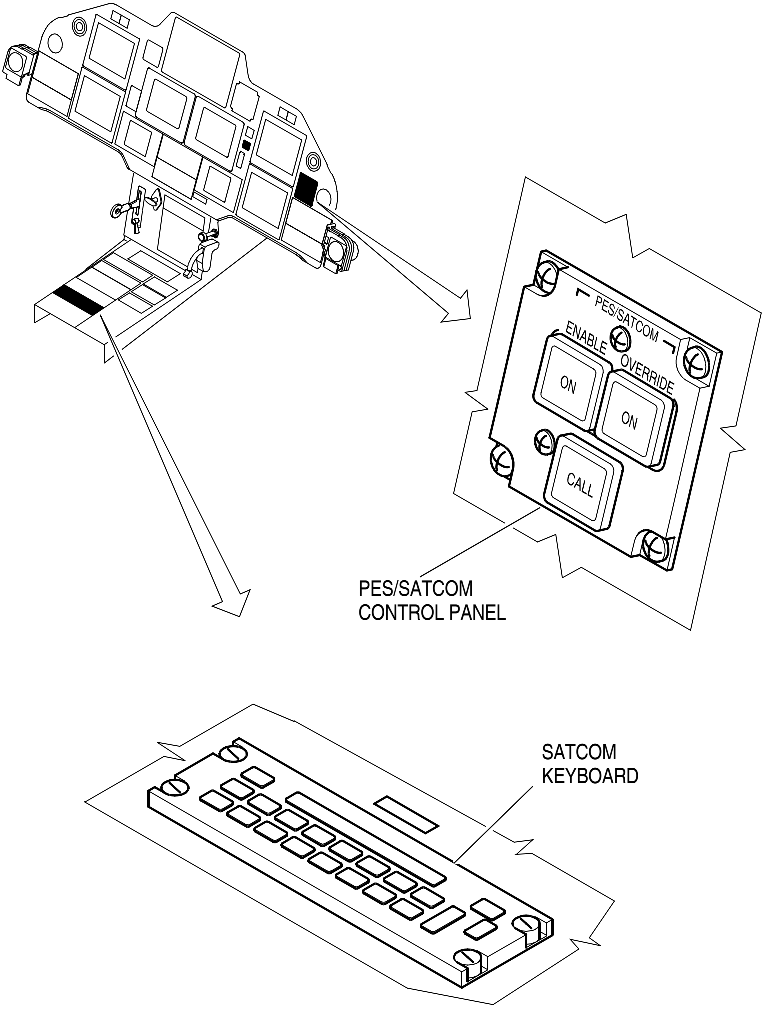

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Figure 7-31C Satcom Aircell with Handset - Component location (Sheet 2 of 2)

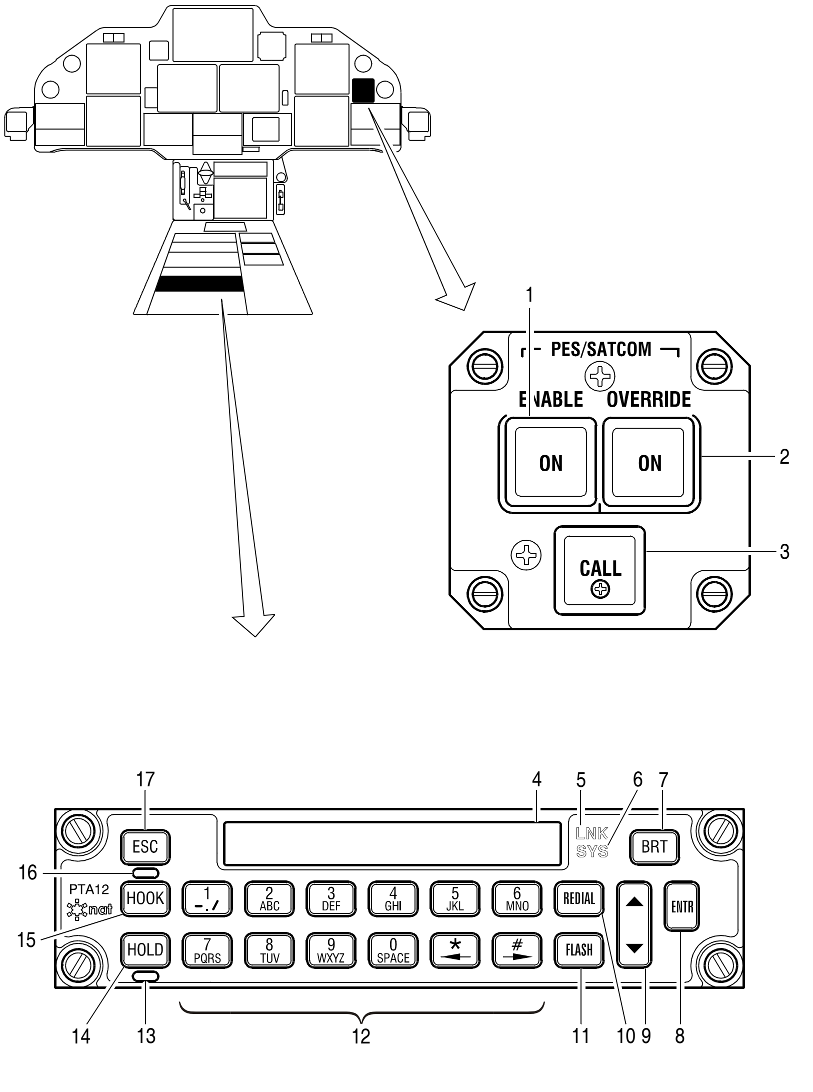

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Figure 7-31D Satcom Aircell with Handset - Controls and Displays (Sheet 1 of 2)

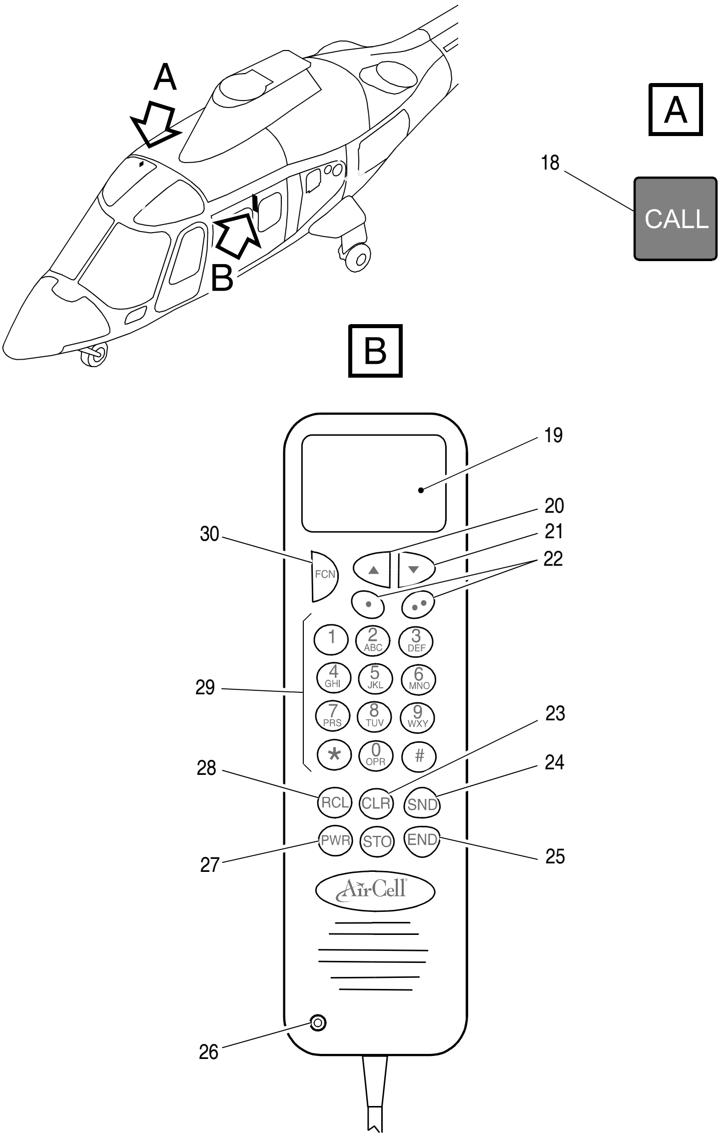

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Figure 7-31D Satcom Aircell with Handset - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-31D

Ref.	Control/Display	Function
1	ENABLE push-button	Allows the pilot to use the Satcom system with the ICS system.
2	OVERRIDE push-button	Allows the pilot to acquire the ICS priority. When pushed all the passengers are connected with ICS system only.
3	CALL light	Flashes when a telephone call is coming.
4	Display	Shows the messages related to the selections done with the buttons.
5	LNK indication	The LNK indication is visible (green light) when the Satcom link is set up. The LNK indication is not visible if the Satcom link is down.
6	SYS indication	The SYS indication is visible (green light) when the Satcom system is ready. The SYS indication is not visible when the Satcom system is busy.
7	BRT button	Controls the display brightness. Each time the BRT button is pushed, the display brightness increases. When the system is in the Default Power-Up mode, the display shows the message BRIGHTNESS and the brightness value (1 thru 8). These data are shown all the time while you operate the button. If you push again the button when the maximum brightness (8) is set, the brightness goes back to 1.
8	ENTR button	(Default Power-Up mode) If pushed when the system is in this operation mode, it puts the system in the Edit mode. If pushed after a phone number is set, it puts the system in the Call mode and does the selection of the phone number. (Recall mode) If pushed after a phone number is recalled, it puts the system in the Call mode and does the selection of the phone number. (Edit mode) If pushed when the system is in this operation mode, it records the data written in the display.

Key to Figure 7-31D

Ref.	Control/Display	Function
9	VOL button	The VOL button is a momentary switch used to increase an decrease the phones output volume. The volume increases or decreases by pressing the up or down arrow. There are a total of 32 steps from the minimum to the maximum volume. By holding either the up or down arrow for more than one second, the volume increases or decreases at a faster rate. The last selected value is stored in the internal memory and is retained upon power down.
10	REDIAL button	Recalls and dials the last number dialed. The number is stored in the internal memory and is retained upon power down.
11	FLASH button	Interrupts a call and allows the user to deal with a second call.
12	Alphanumeric keyboard	The number and symbol buttons on the keyboard are used to dial the required telephone number.
13	HOLD led	The led is on to indicate that a call is on hold. The led is off when the call is returned to active status.
14	HOLD button	Allows the operator to put a call on hold by muting the microphone and phones audio. This is possible only when the unit is active.
15	HOOK button	Starts or receives a call. This corresponds to lifting or hang up the handset of a telephone. Each time the HOOK button is pushed, the hook switch toggles on (active) or off (inactive).
16	HOOK led	The led is off when the unit is inactive and is on when the unit is active. When the unit is inactive the led flashes to indicate an incoming call.
17	ESC button	(Default Power-Up mode) When pushed, it cancels the last operation done. (Recall and Edit modes) When pushed, it cancels the last operation done and the system goes back to the Default Power-Up mode.

Key to Figure 7-31D

Ref.	Control/Display	Function
18	CALL light	Flashes when a telephone call is coming.
19	Display	Shows the number dialed, the system status and other options.
20	UP button	Scrolls the menu up. Turn the volume up.
21	DWN button	Scrolls the menu down. Turn the volume down.
22	Speed dialing keys	Selects stored telephone numbers.
23	CLR button	Deletes characters.
24	SND button	Connects a call.
25	END button	Disconnects a call.
26	Headphone connector	Lets the connection of the headphone.
27	PWR button	Not in use.
28	RCL button	Displays the phone book.
29	Alphanumeric keyboard	Enters alphabetic or numeric characters.
30	FCN button	Accesses phone’s additional menus and advanced functions.

ELECTRICAL POWER

(Chapter 24)

GENERAL

The main sources of electrical power are the engine-driven generators and the 24 V battery.

The electrical power system includes:

– The Alternating Current (AC) generation system
– The Direct Current (DC) generation system
– The External power system
– The AC electrical load distribution system
– The DC electrical load distribution system.

The block diagram of the electrical power system is shown in Figure 7-35.

ELECTRICAL POWER SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-36.

AC GENERATION SYSTEM

The main components of the AC generation system are as follows:

– Two control relays
– Two sensing relays.

The main components of the AC generation system are shown in Figure 7-37.

The inverters are of the single-phase static-type and require a 28 V dc power supply.
Each inverter provides a 115 V ac and a 26 V ac output.

The maximum output power available from a combination of 115 V ac and 26 V ac output power is 250 VA.

If one inverter has a failure, the other inverter, by its control relay (K311 or K312) gives the power to the loads of the bus supplied by the unserviceable inverter.

Two sensing relays (K316 or K317) send the signal of the unserviceable inverter to the DAU which causes the activation of the INV 1 or INV 2 caution message on the EDU 1.

The system is powered through the circuit breakers that follow:

– INV 1 (28 V DC EMER BUS #1)
– INV 2 (28 V DC MAIN BUS #2).

Refer to Figure 7-38 for the AC generation system schematic diagram.

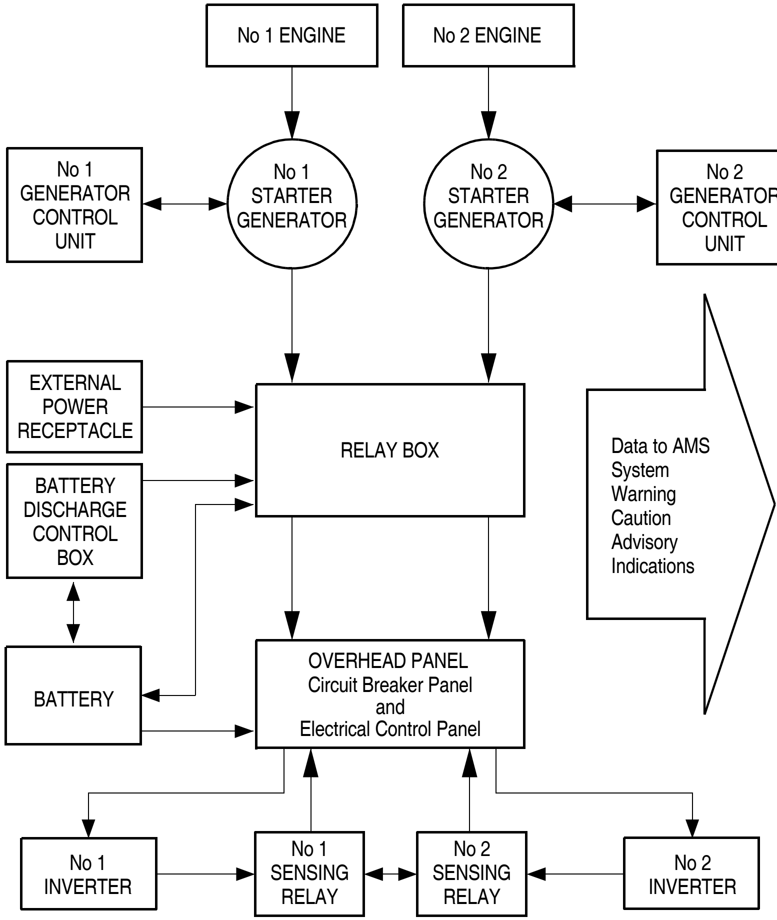

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Figure 7-35 Electrical Power System - Block Diagram

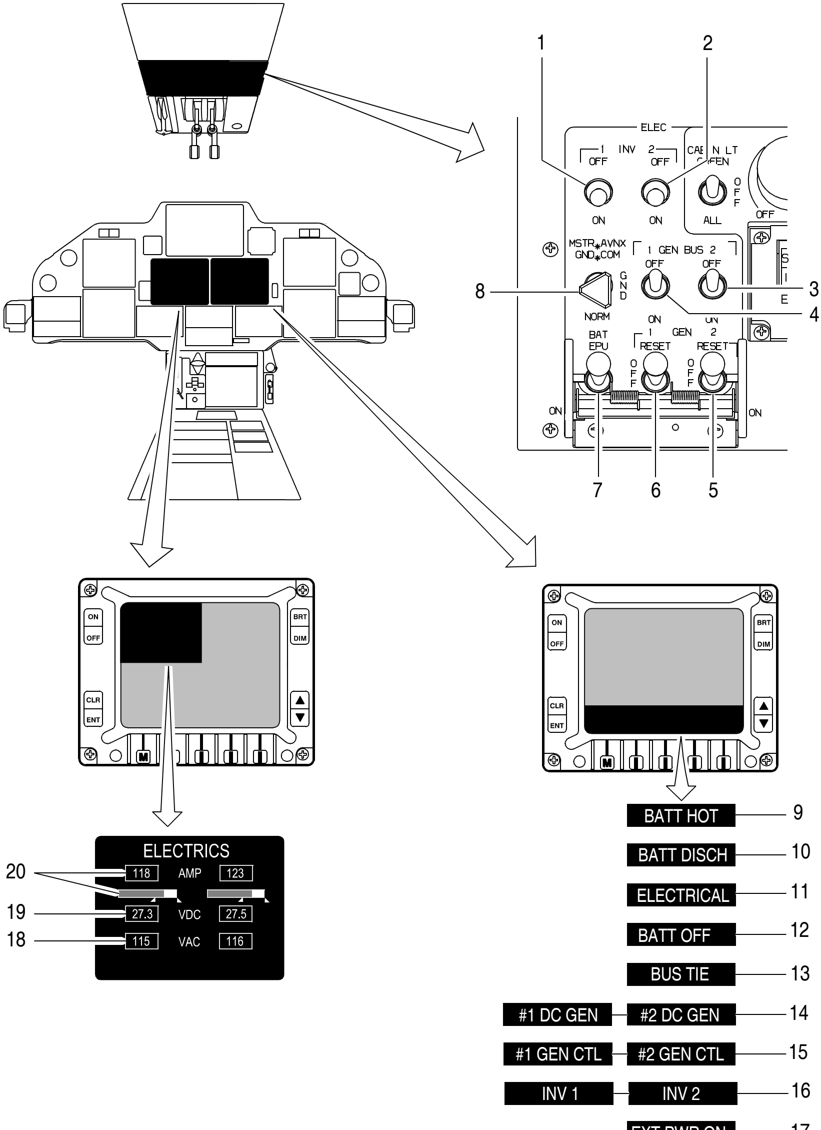

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Figure 7-36 Electrical Power System - Controls and Displays

Key to Figure 7-36

Ref.	Control/Display	Function
1	INV 1 switch	OFF - The No 1 inverter is de-energized. ON - The No 1 inverter is energized.
2	INV 2 switch	OFF - The No 2 inverter is de-energized. ON - The No 2 inverter is energized.
3	GEN 2 BUS switch	OFF - The No 2 generator bus is disconnected from the battery bus. ON - The No 2 generator is connected to the battery bus.
4	GEN 1 BUS switch	OFF - The No 1 generator bus is disconnected from the battery bus. ON - The No 1 generator bus is connected to the battery bus.
5	GEN 2 BUS RESET switch	OFF - The No 2 generator shunt field is de-energized. ON - The No 2 generator shunt field is energized.
6	GEN 1 BUS RESET switch	OFF - The No 1 generator shunt field is de-energized. ON - The No 1 generator shunt field is energized.
7	BAT switch	OFF - The battery is disconnected from the battery bus. ON - The battery feeds power to the battery bus.
8	MSTR AVNX switch	GND COM - The electrical power is supplied only to the GND BUS. GND - The electrical power is not supplied to AUX BUS 1 and AUX BUS 2. NORM - All the avionic equipment are supplied with electrical power. **NOTE** The MSTR AVNX switch is inhibited in flight through the Weight On Wheels (WOW) microswitch.

Key to Figure 7-36 (Continued)

Ref.	Control/Display	Function
9	BATT HOT warning message	Shown when the battery temperature reaches 71 ±3 °C.
10	BATT DISCH warning message	Shown when the battery is not charged. The output voltage of both generators is lower than 26.5 V.
11	ELECTRICAL warning message	Shown when both generators are off line or shut down by the GCU due to a malfunction.
12	BATT OFF caution message	Shown when the battery is off-line.
13	BUS TIE caution message	Shown when the related bus tie is open.
14	#1 (#2) DC GEN caution message	Shown whenever a failure de-energizes the related generator relay.
15	#1 (#2) GEN CTL	Shown when the related GEN CONTR circuit breaker on the relay box is open.
16	INV 1 (2) caution message	Shown when the No 1 (or the No 2) inverter is inoperative.
17	EXT PWR ON caution message	Shown when the access door of the external power receptacle is open or the external power is connected.
18	VAC indications	AUX mode only. Show the tension values of the alternate current output (115 V) from the No 1 and No 2 inverters.
19	VDC indications	AUX mode only. Show the tension values of the direct current output from the No 1 and No 2 generators.
20	AMP indications and horizontal bars	AUX mode only. Show the current values of the direct current output from the No 1 and No 2 generators.

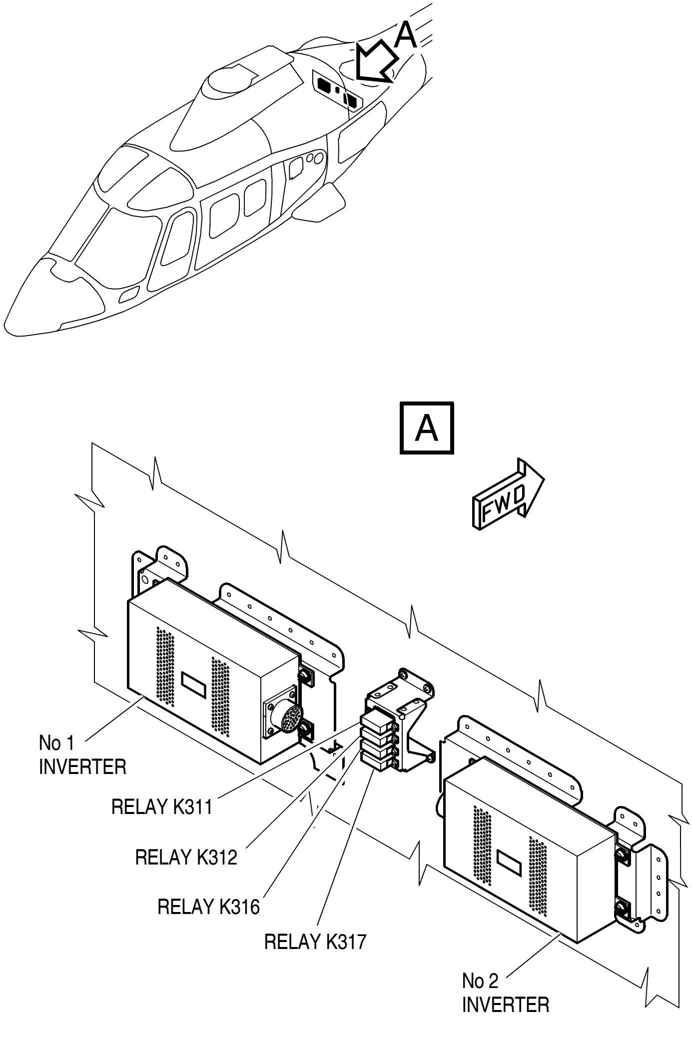

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Figure 7-37 AC Generation System - Component Location

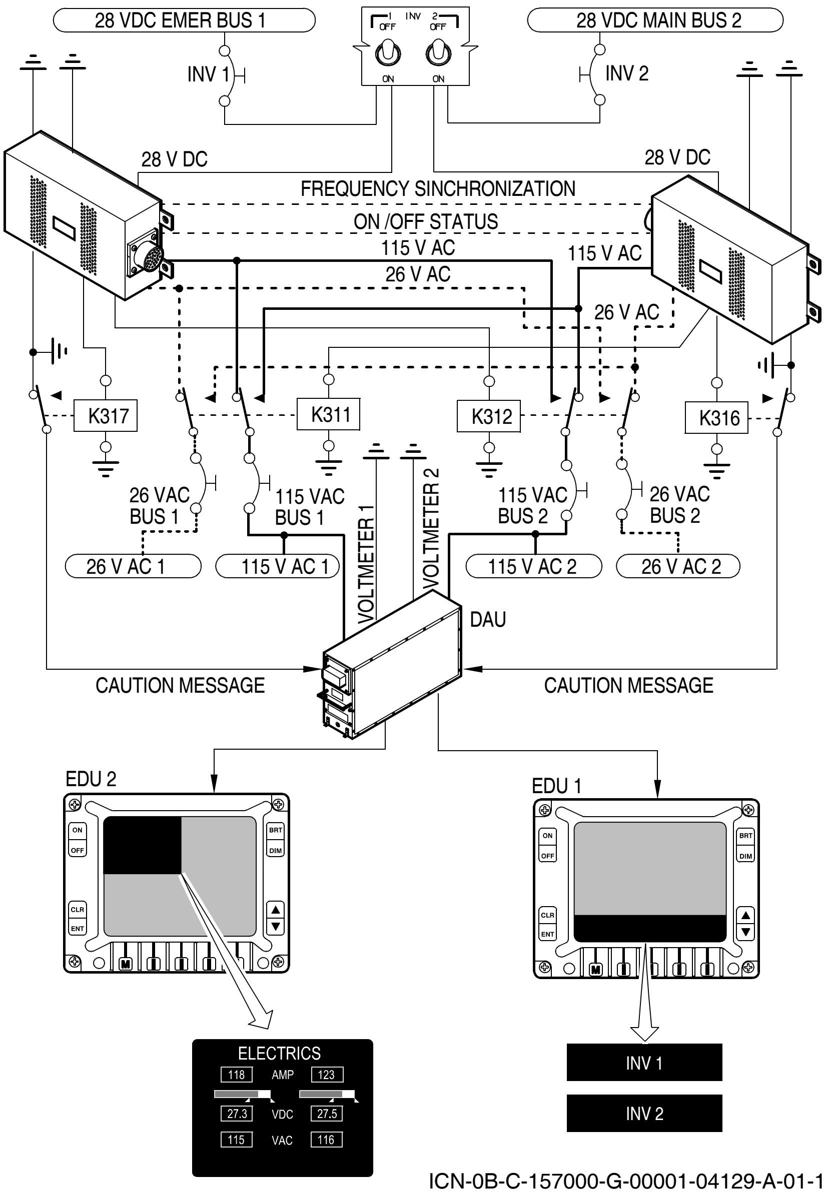

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Figure 7-38 AC Generation System - Schematic Diagram

DC GENERATION SYSTEM

The 28 V dc electrical power is supplied by:

– two generators driven by the engine
– a 24 V dc, 33 Ah nickel-cadmium battery
– an external power source when the helicopter is on the ground.

The main components of the DC generation system are shown in Figure 7-39.

The 28 V dc electrical power is supplied by the two generators when the engines are operating; the generators operate also as starters.
A control and adjustment circuit is associated to each generator; the circuit provides for voltage adjustment, load distribution during parallel operation and over-voltage and reverse current protection.
The generators, which furnish power at rated voltage when the rotor speed reaches 67% rpm, are connected to the related bus through a relay controlled by the GEN 1 or GEN 2 switch.
The #1 DC GEN or #2 DC GEN caution message comes in view on the EDU1 when the failure of one generator occurs. In the event of failure of one generator, the operative generator is capable of supplying all systems.
In event of failure of both generators or when the engines are shutdown, power to the various systems is supplied by the 24 V dc battery connected to the battery bus. This connection is made by a relay controlled by the BAT switch.
The battery, housed in the nose compartment, is provided with a temperature sensor. When the battery temperature exceeds 71 ±3 °C, the sensor sends a signal to the DAU which causes the display of the BATT HOT caution message on the EDU1.
On the ground, and with the battery disconnected, the electrical power may be obtained from an external power source (28 V dc, 400 A minimum) connected to the system through an appropriate receptacle located on the right side of the rear fuselage. Opening of the external power receptacle access door causes the display of the EXT PWR ON caution message on the EDU1.

The system is powered through the circuit breakers that follow:

– GEN 1 BUS (28 V DC BUS #1)
– GEN 2 BUS (28 V DC BUS #2)

– ESS BUS 1 (28 V DC BUS #1)
– ESS BUS 2 (28 V DC BUS #2)
– ESS BUS TIE (28 V DC BUS #1/2)
– STARTER GEN 1 RESET (28 V DC BUS #1)
– STARTER GEN 2 RESET (28 V DC BUS #2).

Refer to the Figure 7-40 for the DC generation system schematic diagram.

AC ELECTRICAL LOAD DISTRIBUTION SYSTEM

The AC power is distributed through circuit breakers connected to two 115 V ac bus bars and two 26 V ac bus bars.

Refer to Figure 7-38 for the bus interconnections.

Refer to Figure 7-8 for the location of the circuit breakers installed on the circuit breaker panel of the overhead console and the to Figure 7-39 for the circuit breakers installed on the rear auxiliary circuit breaker panel in the baggage compartment.

DC ELECTRICAL LOAD DISTRIBUTION SYSTEM

The DC power is distributed through circuit breakers connected to seven bus bars.

Refer to Figure 7-40 for the bus interconnections.

Refer to Figure 7-8 for the location of the circuit breakers installed on the circuit breaker panel of the overhead console and the to Figure 7-39 for the circuit breakers installed on the rear auxiliary circuit breaker panel in the baggage compartment.

AVIONIC EQUIPMENT ELECTRICAL POWER SUPPLY

The DC power necessary for the operation of the communication and navigation equipment is controlled by the MSTR AVNX switch, installed on the overhead console.
The switch has three positions: NORMAL, GND and GND COM.

When the switch is in the NORMAL position, the electrical power is supplied to all the avionics equipment.

When the switch is in the GND position, the equipment connected to the No 1 and No 2 auxiliary bus are not energized.

When the switch is in the GND COM position, the electrical power is supplied only to the ground bus.

The MSTR AVNX switch operation is inhibited during flight through the Weight on Wheel (WOW) switch.

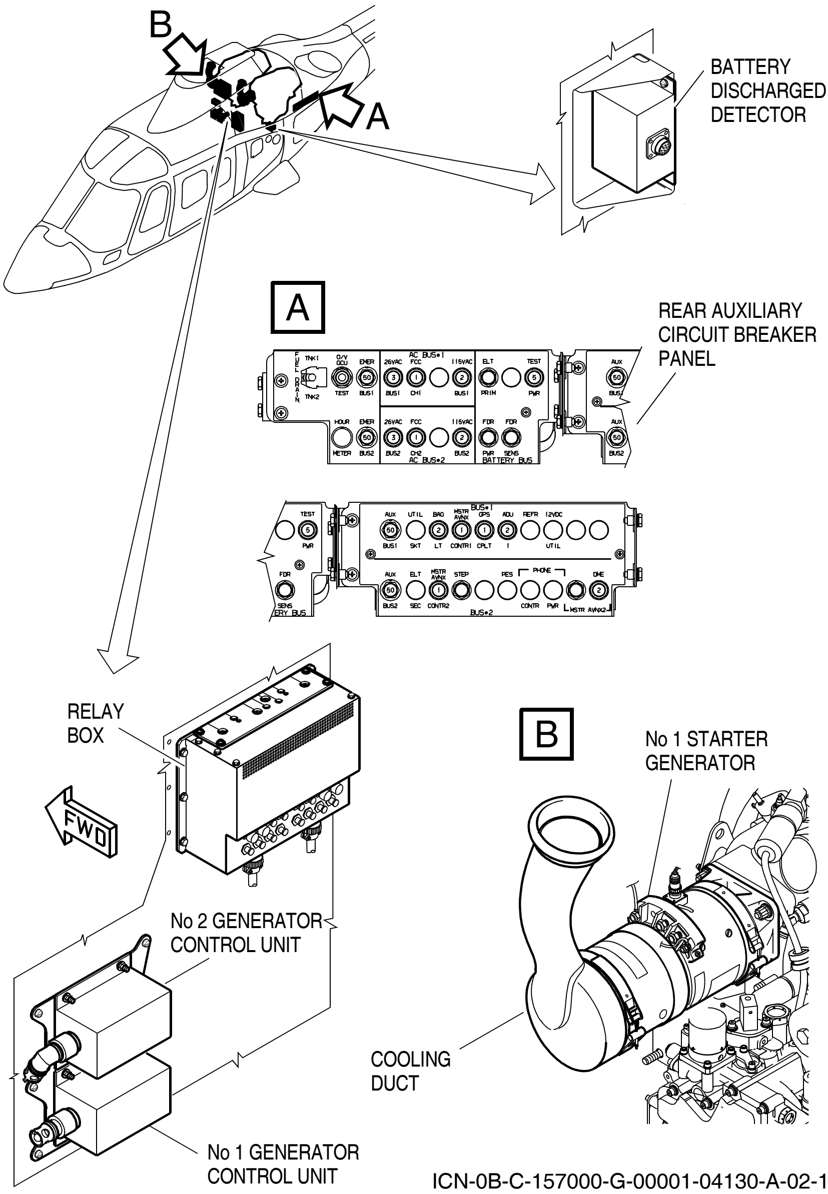

Figure 7-39 DC Generation System - Component Location (Sheet 1 of 2)

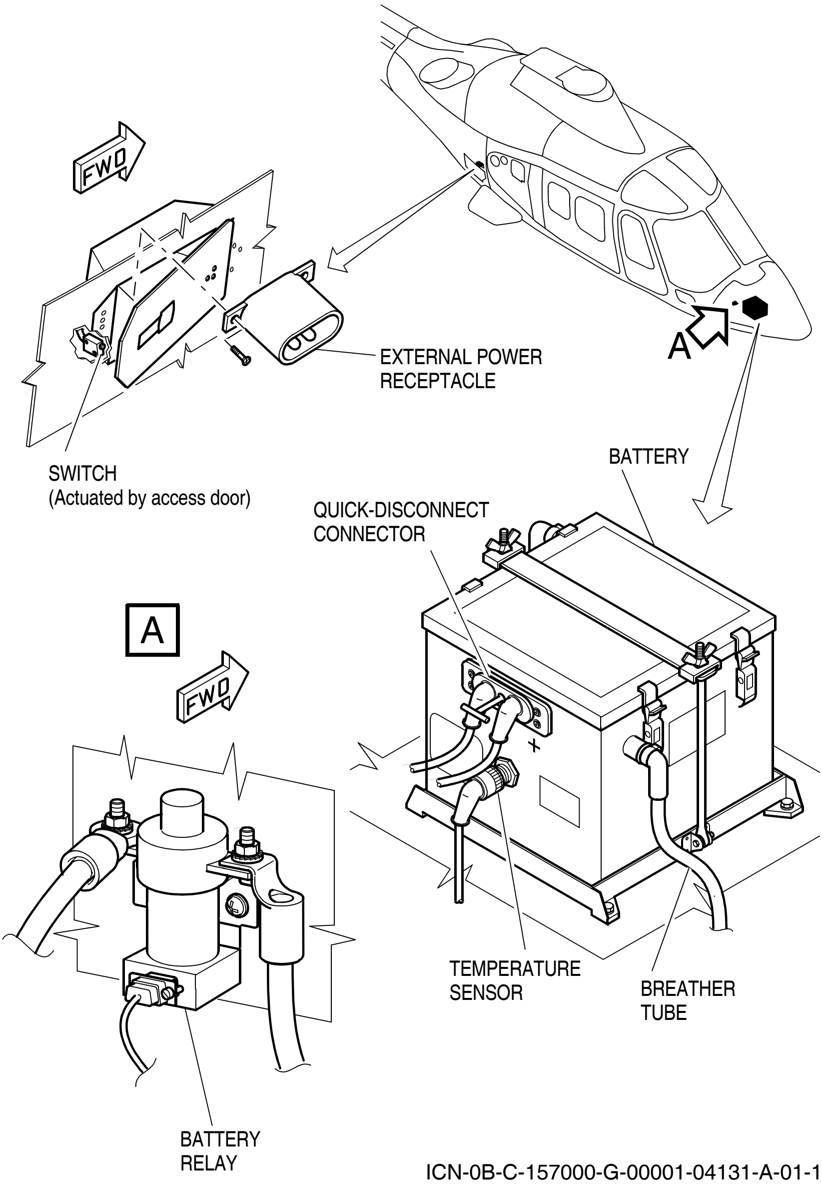

Figure 7-39 DC Generation System - Component Location (Sheet 2 of 2)

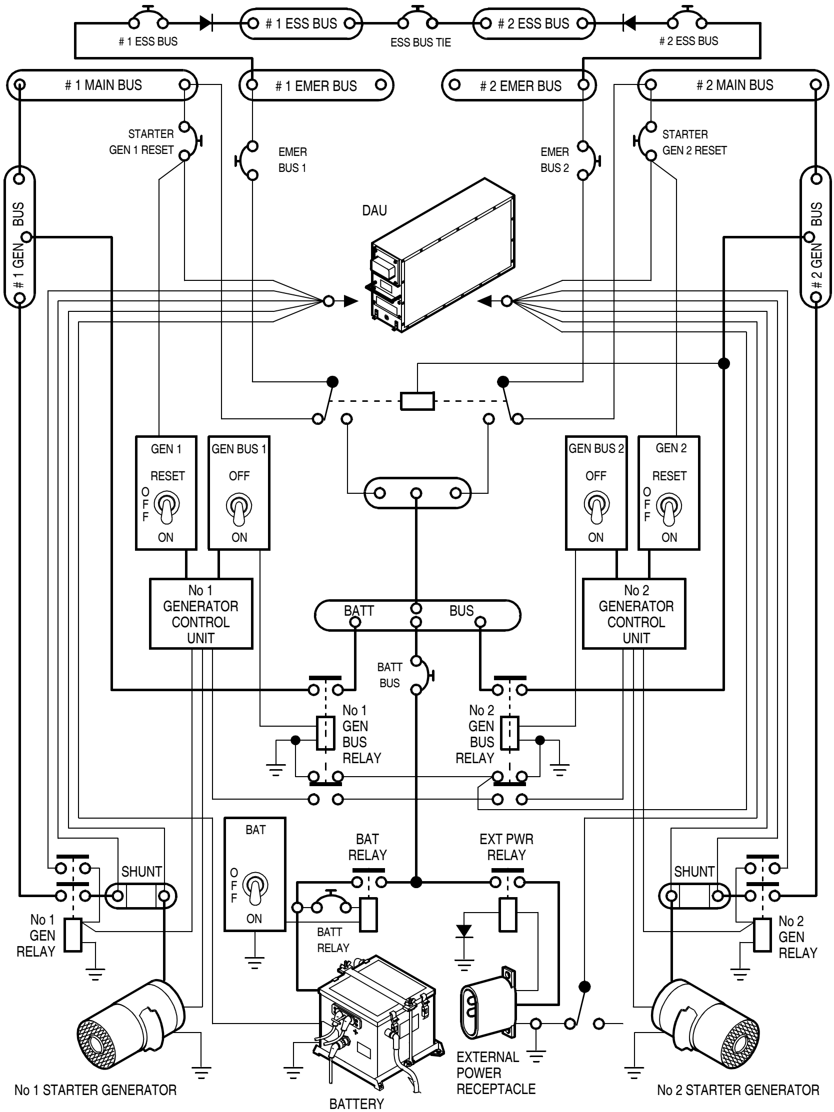

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Figure 7-40 DC Generation System - Schematic Diagram

EQUIPMENT/FURNISHING

(Chapter 25)

GENERAL

The equipment/furnishing system includes:

– The flight compartment furnishing
– The passenger compartment furnishing.

The optional installation/systems that follow can be installed on the helicopter:

– The emergency locator system
– The wire strike protection system
– The first aid kit
– The external hoist installation
– The cargo hook installation
– The pilot external mirror installation
– The safety cargo hook installation
– The copilot external mirror installation (optional).

FLIGHT COMPARTMENT FURNISHING

The flight compartment furnishing includes (Figure 7-41):

– The crew seats
– The crew seat safety belts and inertia-reel shoulder-harnesses.

CREW SEATS

The crew seats are of composite construction and are adjustable forward and aft on rollers in the seat tracks. Two cushions of flexible polyurethane are formed in the seat structure.
An adjustable lumbar support device is mounted on the backrest; it is controlled through the lumbar adjustment knob.
The adjustable seats are provided with a locking device controlled by a lever. The device consists of a plunger engaging one of the four holes on the tracks which forms the seat adjustment system.

The plunger is spring loaded in the locked position.
To allow the seat to slide, the lever must be raised thus disengaging the plunger and releasing the seat.

CREW SEAT SAFETY BELTS AND INERTIA REEL SHOULDER HARNESSES

The pilot and copilot have safety belts and inertia-reel shoulder-harnesses.
The inertia reel incorporates an automatic locking mechanism that prevents the seat occupant from leaning forward in the event of a rapid deceleration.

PASSENGER COMPARTMENT FURNISHING

The passenger compartment furnishings includes (Figure 7-42):

– The passengers' seats
– The safety belts and inertia-reel shoulder-harnesses.

PASSENGER SEATS

Different types of seats can be installed in the passsenger compartment, according to customer's requirements.
The typical configuration consists of two three-seater benches.
The seats are secured to four supports by means of quick-release pins. The supports are bolted to the structure.

PASSENGER SEAT SAFETY BELTS AND INERTIA REEL SHOULDER HARNESSES

The safety belts and inertia-reel shoulder-harnesses are installed on each passenger seat.
The inertia reel is a mechanical restraining device that prevents the passengers from leaning forward during flight.

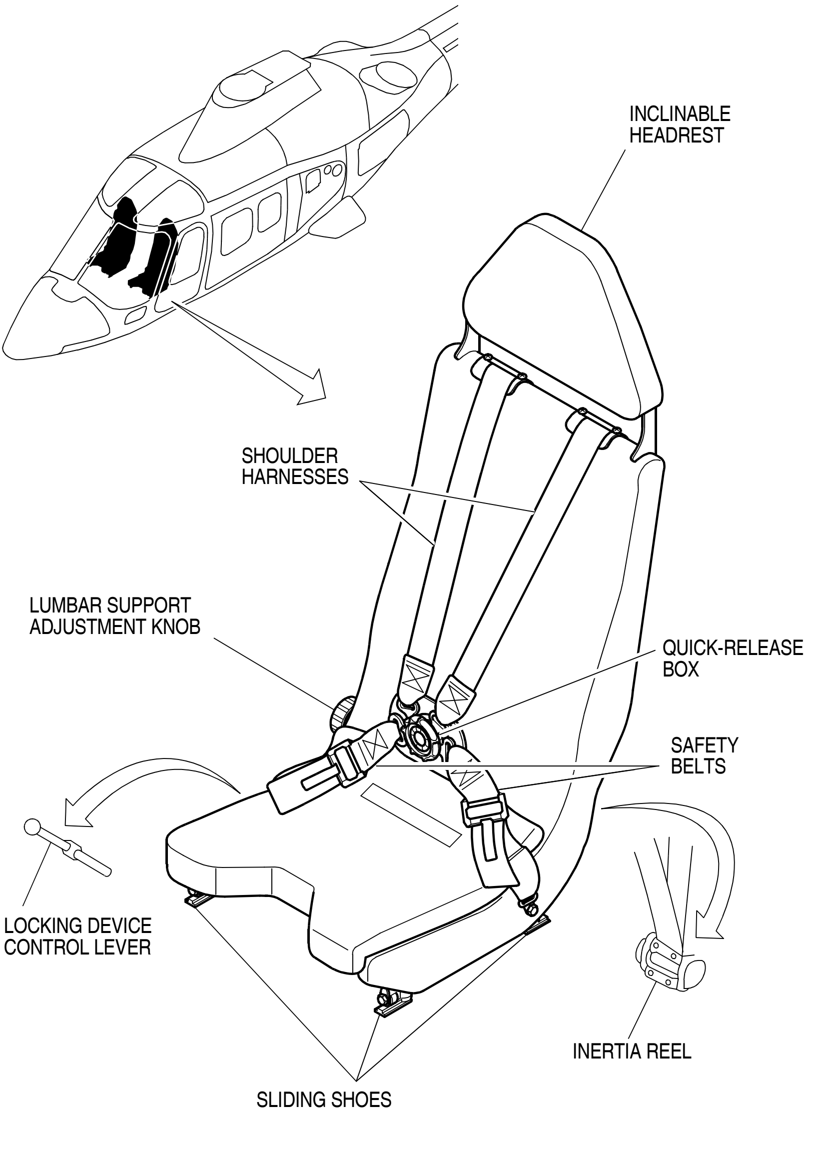

Figure 7-41 Flight Compartment Furnishing

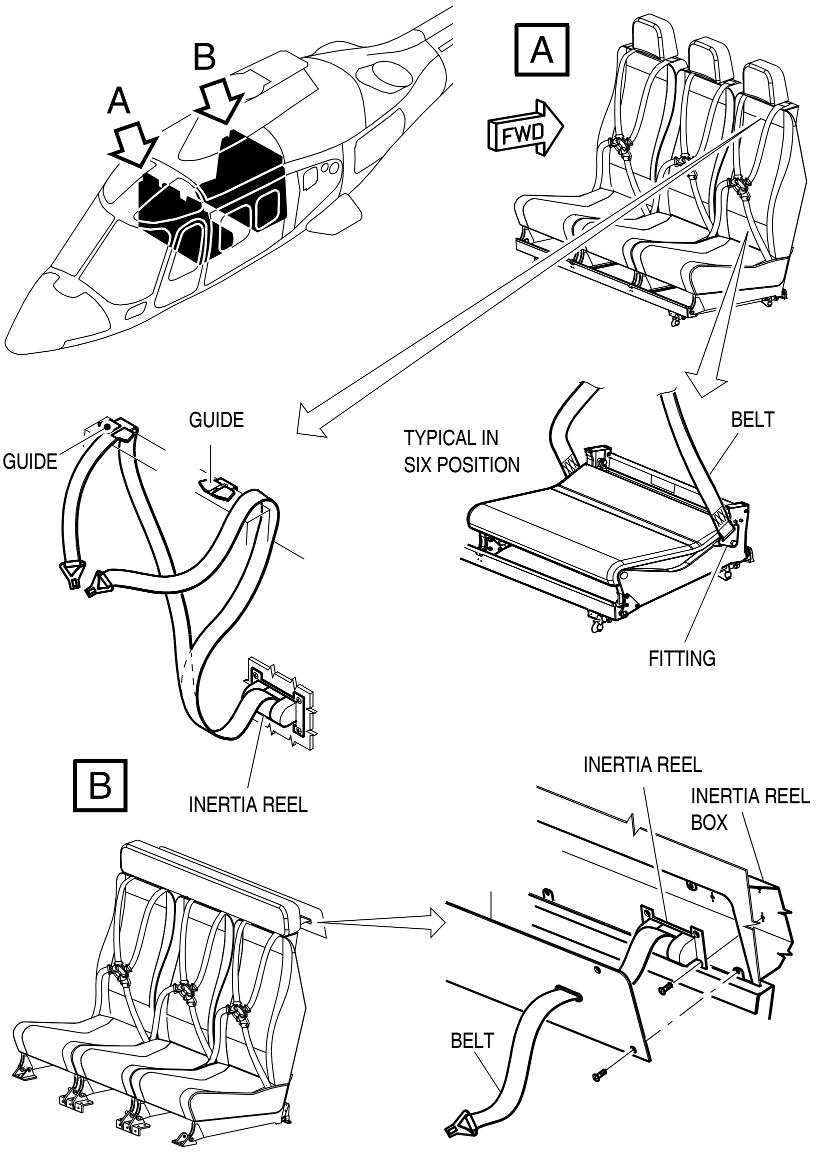

Figure 7-42 Passenger Compartment Furnishing (Typical)

EMERGENCY LOCATOR SYSTEM

The emergency locator system consists mainly of an Emergency Locator Transmitter (ELT), an antenna and a switch on the instrument panel.
The ELT is installed in the tail boom. The antenna is located on the tail boom.
The ELT automatically activates in the event of a crash and transmits the standard swept tone on 121.5, 243.0 and 406.0 MHz radio frequencies. During transmission an encoded digital message is sent to the satellite. The information contained in the message is:

– The country code
– The identification code
– The manufacturer code
– The operator code.

The ELT continues to operate until the battery discharges, that is during 72 hours at least.
A switch in the cockpit permits the ELT to be tested and reset. It is not possible to turn off the system from the cockpit.
The system is powered through the circuit breaker that follows:

– ELT PRIMARY (28 V dc BATTERY BUS).

The main components of the emergency locator system are shown in Figure 7-43.

EMERGENCY LOCATOR SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-44.

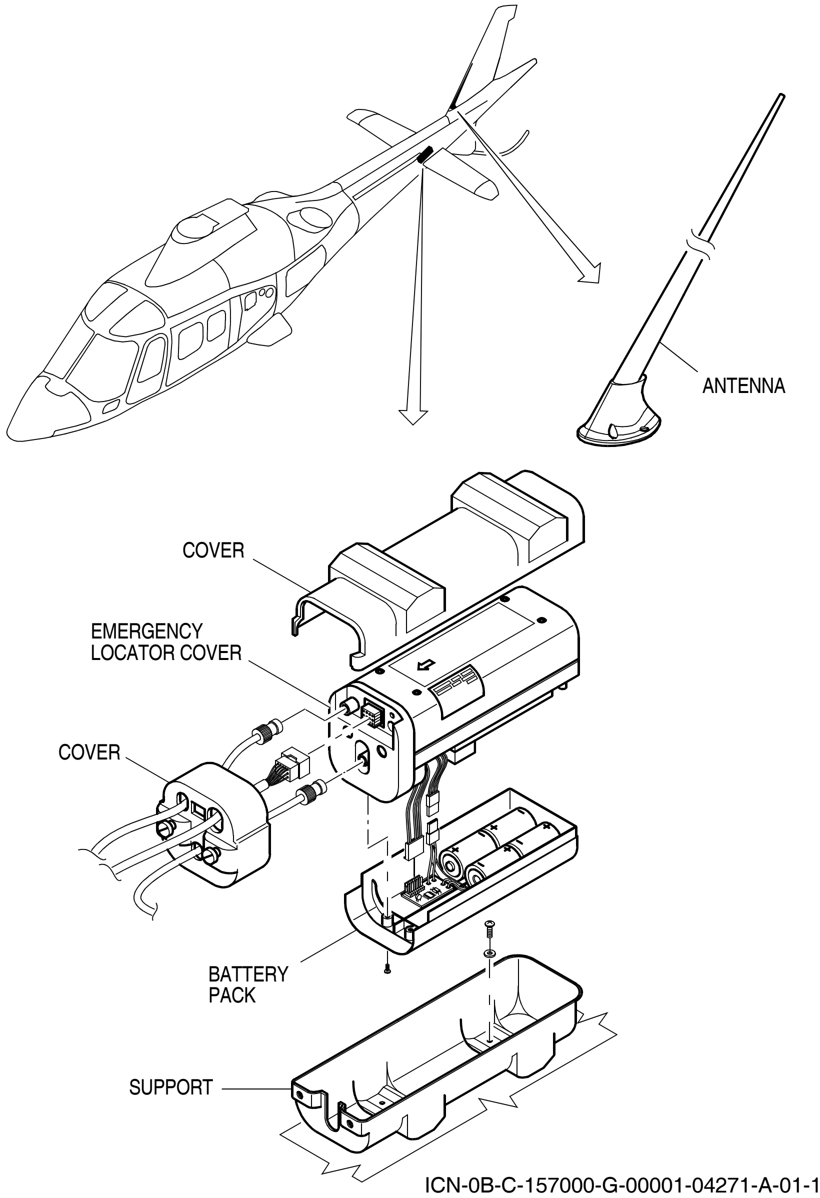

Figure 7-43 Emergency Locator System - Components Location

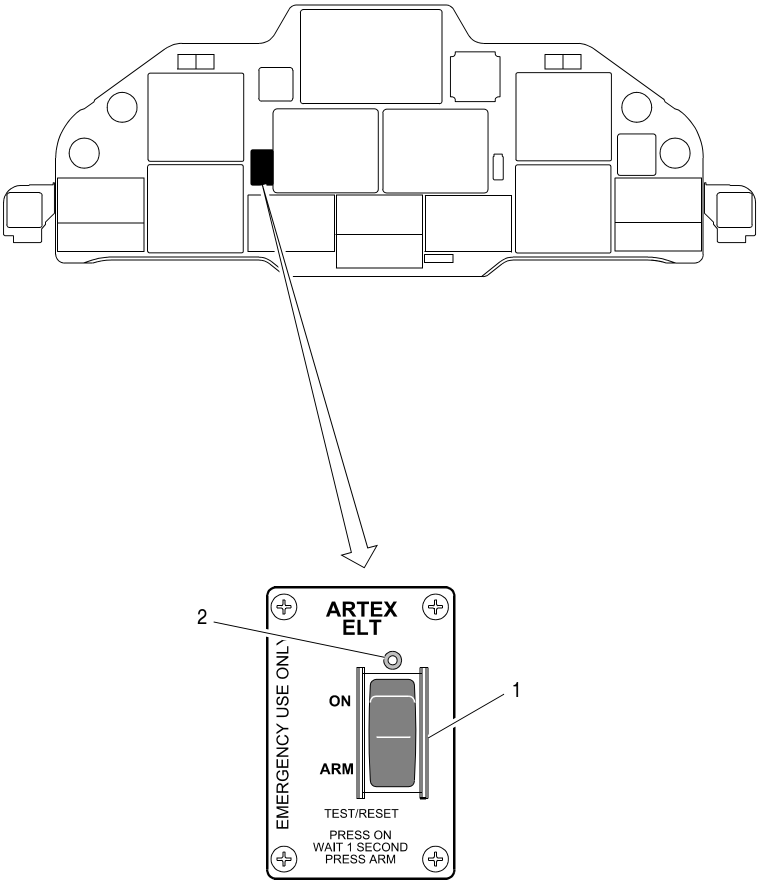

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Figure 7-44 Emergency Locator System - Controls and Displays

Key to Figure 7-44

Ref.	Control/Display	Function
1	ELT switch	ARM - The ELT is activated to operate in normal automatic condition. ON position - pressed (for more than 1 second) activates the test/reset of the ELT. The swept tone signal and the flashing light operate.
2	Flashing light	Comes on when the ELT switch is held in the ON position.

WIRE STRIKE PROTECTION SYSTEM

The wire strike protection system prevents damage to the helicopter if it strikes aerial wires that are horizontal in relation to the ground.

The primary components of the system are:

– The nose cutter
– The nose deflector
– The two windshield deflectors
– The upper deflector
– The upper cutter.

The wire is cut when it hits the upper cutter or the nose cutter. The helicopter in flight supplies the kinetic energy necessary to cut the cable.

The wire strike protection system gives the best protection when the helicopter is in straight and level flight. If the attitude of the helicopter is different from straight and level flight, the protection is worse.

The nose cutter can be locked in one of its two positions:

– The forward-tilted position
– The backward-tilted position.

The forward-tilted position is the flight position. The backward-tilted position is used when the nose bay radome must be opened, or the helicopter must be towed.

When the nose cutter is in the backward-tilted position, a safety pin with a red flag must be put into the cutter.

The main components of the wire strike protection system are shown in Figure 7-45.

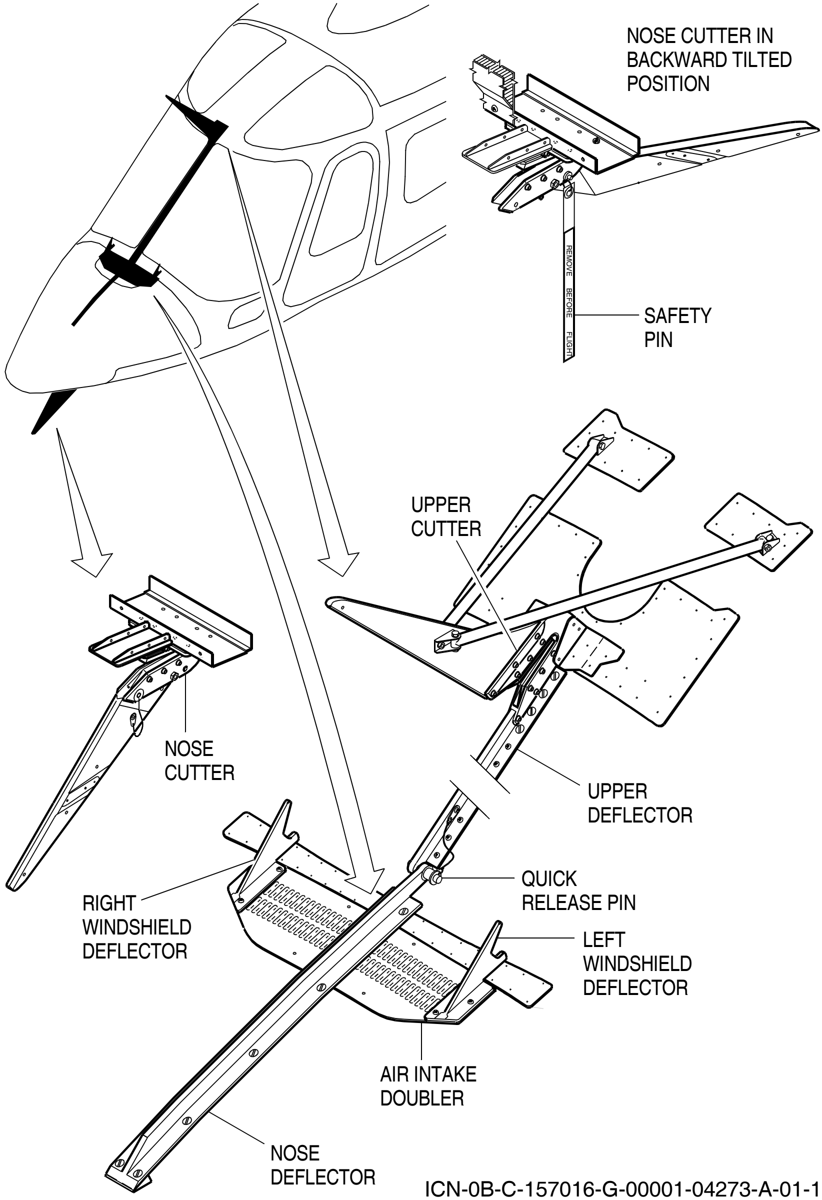

Figure 7-45 Wire Strike Protection System - Components Location

FIRST AID KIT

The first aid kit (Figure 7-46) is installed below the center seat of the front seat row in the cabin.

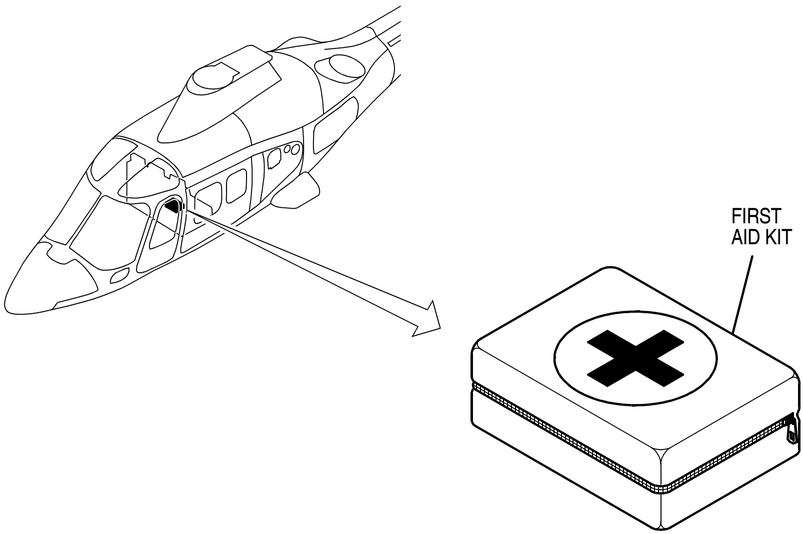

Figure 7-46 First Aid Kit

EXTERNAL HOIST INSTALLATION

The external hoist installation is used for rescue operations and to lift and lower cargo loads in areas where the helicopter cannot land.

The primary components of the external hoist installation are:

– The electrical hoist motor and winch assembly
– The support assembly
– The step bar protection
– The control pendant
– The cable payout display
– The hoist control panel
– A radio-ICS system to allow communcation among crew members and the Human External Cargo (HEC)
– The safety shoulder harness
– The manual cable cutter.

The hoist motor and winch assembly is installed on the right side of the helicopter, above the passengers' compartment. The support assembly attaches it. The hoist motor and winch assembly includes the components that follow:

– An electric motor
– A storage drum
– A hoist cable and hook
– A hook assembly
– A level wind screw
– A limit switch
– A oil pump
– An automatic brake
– An override limit switch
– A traction sheave
– A cable guide
– A cable cutter
– 88.4 m (290 ft) of load cable (this length is the usable length).

The automatic brake holds the load at the set height and prevents slippage also if a power failure occurs. The limit switch assembly gives automatic acceleration/deceleration of hoist cable and stops the hoist cable at various preset positions.

The hoist motor and winch assembly also has a cable foul protection. It is a switch installed on the hoist flanges parallel to the drum axis. When a cable foul occurs, the switch stops the supply of electrical power to the hoist motor. At the same time it causes the HOIST CABL LKD caution message to come into view on the pilot's EDU.

In an emergency, an electrically-operated cable cutter can cut the cable. A control on the handgrip of the pilot's collective lever operates the cable cutter. When the safety guard of the hoist-cable cut-switch is up, the HOIST CUT ARMD caution message is in view on the pilot's EDU. The cable cutter cuts the cable, and the cable flies away from the helicopter. Damage to the fuselage and tail blades is thus prevented.

The control pendant is installed on the right side of the passengers' compartment. It lets the hoist operator to move freely when he/she operates the hoist.

The hoist control panel is installed on the interseat console. It includes the on/off HOIST switch and a TEST push-button switch. When the HOIST switch is at ON, the HOIST ON advisory message is in view on the co-pilot's EDU.

The installation also includes a step bar protection, a safety shoulder harness for the hoist operator and a manual cable cutter. The hoist operator operates the manual cable cutter when necessary.

The system is powered through the circuit breakers that follow:

- HOIST PWR (28 V DC BATT BUS)
- HOIST CONTR (28 V DC BATT BUS)
- HOIST CABLE CUT 1 (28 V DC ESS BUS #1)
- HOIST CABLE CUT 2 (28 V DC ESS BUS #2).

These circuit breakers are installed on the overhead console.

The main components of the system and their location are shown in Figure 7-47.

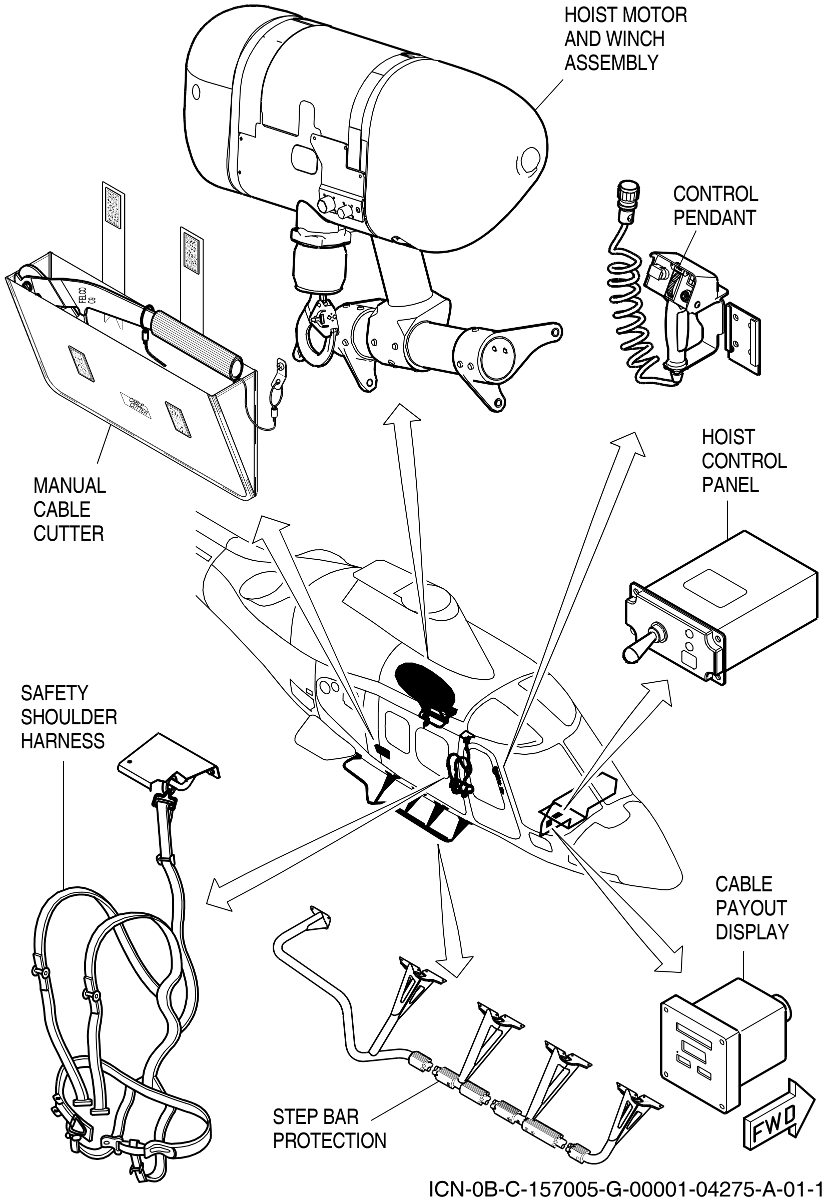

Figure 7-47 External Hoist Installation - Components Location

EXTERNAL HOIST INSTALLATION CONTROLS AND DISPLAYS

Refer to Figure 7-48

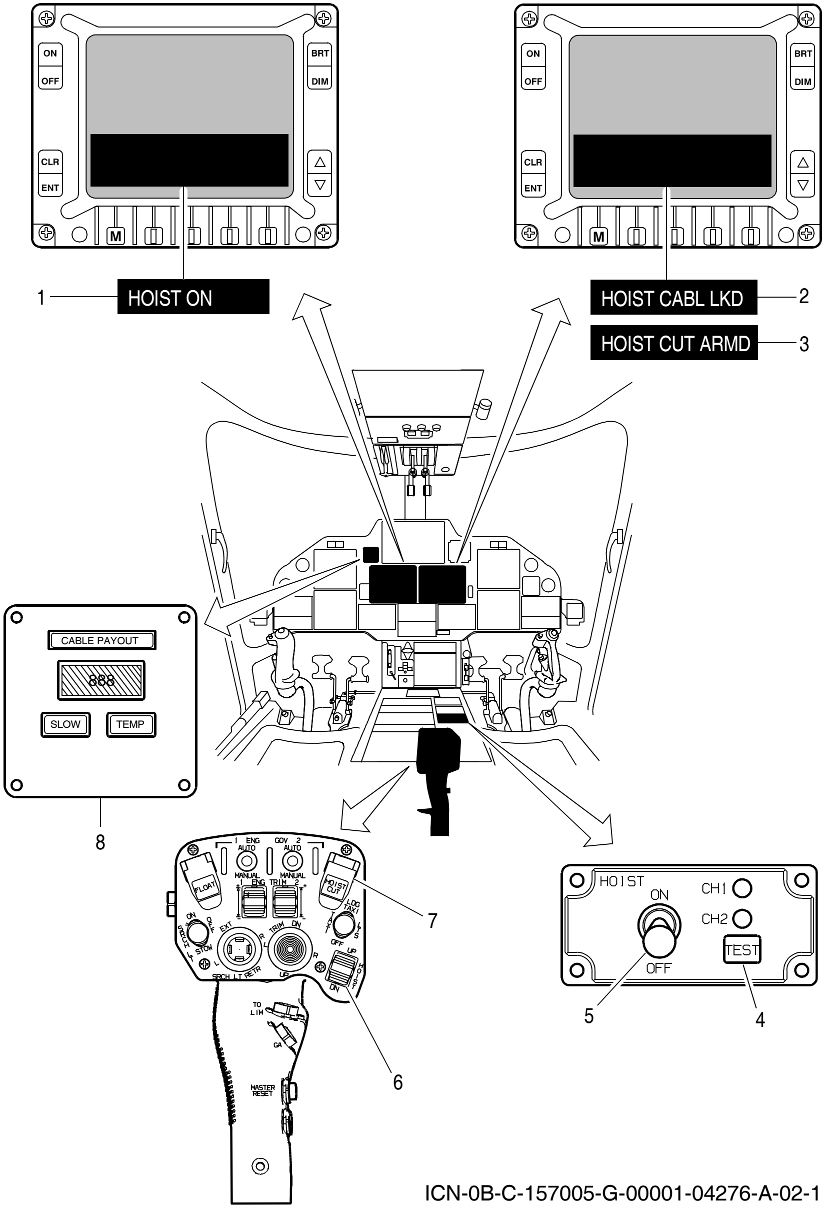

Figure 7-48 External Hoist Installation - Controls and Displays (Sheet 1 of 2)

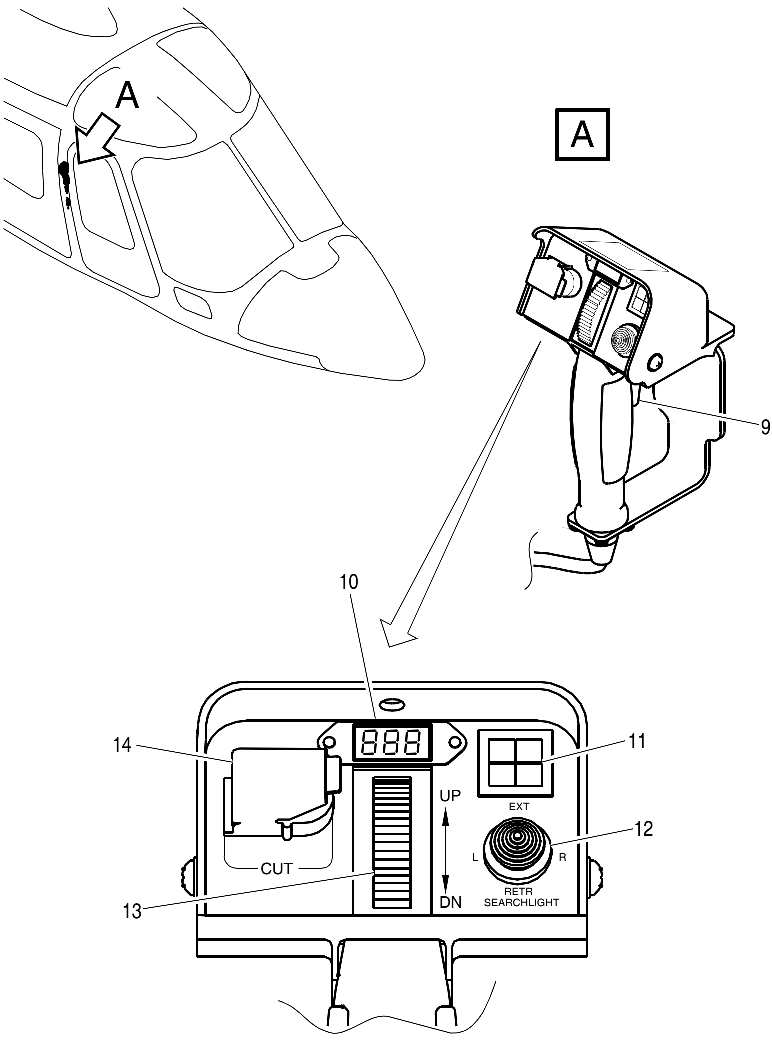

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Figure 7-48 External Hoist Installation - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-48

Ref.	Control/Display	Function
1	HOIST ON advisory message	Shown when the external hoist system is ready for operation.
2	HOIST CABL LKD caution message	Shown when there is a foul condition on the hoist drum.
3	HOIST CUT ARMD caution message	Shown when the cable cutter is ready for operation.
4	TEST push-button	When pressed performs the cable cut squib test (cartridge).
5	HOIST switch	ON - The system is energized. OFF - The system is de-energized.
6	Hoist control switch	Up - Controls the up movement of the load cable at stable speed. Down - Controls the down movement of the load cable at stable speed.
7	Hoist cable cut switch	Guard down - The cable cutter is safe. Guard up - The cable cutter can be operated to cut the load cable in an emergency.
8	Cable payout display indicator	Shows the length of the cable unwound from the hoist in feed or meters. When the system is first supplied with electrical power, the display shows 888. Then it shows the length of the unwound load cable (000 is shown if all the cable is wound on the drum). SLOW light - When on, it shows that the limit cable occured. TEMP light - When on, it shows that the motor overtemperature occured.
9	ICS trigger switch	When operated, it connects the hoist operator's headset to the ICS.
10	Cable payout display	Shows the length of the cable unwound from the hoist in feed or meters. When the system is first supplied with electrical power, the display shows 888. Then it shows the length of the unwound load cable (000 is shown if all the cable is wound on the drum).

Key to Figure 7-48 (Continued)

Ref.	Control/Display	Function
11	CABLE/SLOW/TEMP lights	CABLE light - When on, it shows that a cable foul occurs and the limit switch de-energizes the hoist motor. SLOW light - When on, it shows that the limit cable occured. TEMP light - When on, it shows that the motor overtemperature occured.
12	EXT/R/RETR/L searchlight switch	Is used to control the searchlight installed in the right lower part of the center fuselage (if installed).
13	Direction/speed control thumb wheel	OFF - The hoist is in the rest position. UP - Operates the switch that controls the up movement of the load cable, and sends a speed command signal to the variable speed controller. DN - Operates the switch that controls the down movement of the cable, and sends a speed command signal to the variable speed controller. The up and down speeds increase the more the thumb wheel is moved away from the center position.
14	Hoist cable cut switch	Guard down - The cable cutter is safe. Guard up - The cable cutter can be operated to cut the load cable in an emergency.

CARGO HOOK INSTALLATION

The cargo hook installation is used to move loads of a maximum weight of 1000 kg (2205 lb) that are hung below the helicopter.

The primary components of the cargo hook installation are:

– The cargo hook
– The support frame
– The gimbal
– The electrical release system
– The emergency manual-release system
– The electronic hook load measuring system.

The main components of the system are shown in Figure 7-49.

The cargo hook is installed on the support frame. The support frame is attached to four supports that are attached to the bottom of the center fuselage, near the center of gravity of the helicopter. Two pins connect the support frame to the rear supports. Two quick-release pins connect the support frame to the front supports. A spring system attached to the cargo hook stows the hook when it is not used.

The electrical release system is an electrical circuit that includes two cargo hook switches and an electro-mechanical device. The switches are installed on the handgrip of the pilot's and co-pilot's cyclic stick. The electro-mechanical device controls the opening of the hook. The cargo hook switch has a guard. When the guard is up, the switch is energized and a HOOK ARMED caution message is shown on the pilot's EDU. When the switch is operated, the cargo hook opens and releases the cargo sling. A HOOK OPEN caution message comes in view on the pilot's EDU.

The installation is powered through the circuit breaker that follows:

– CARGO HOOK PRI (28 V DC BATT BUS).

This circuit breaker is installed on the overhead console.

If an electrical failure occurs, the crew can operate the emergency cargo-release handle (the CARGO HOOK handle that is installed between the crew's seats) to release the load. Or a ground assistant can turn the manual release lever on the hook to release the load.

The system that measures the load weight includes a load cell and a digital load indicator. The load cell is installed between the gimbal and

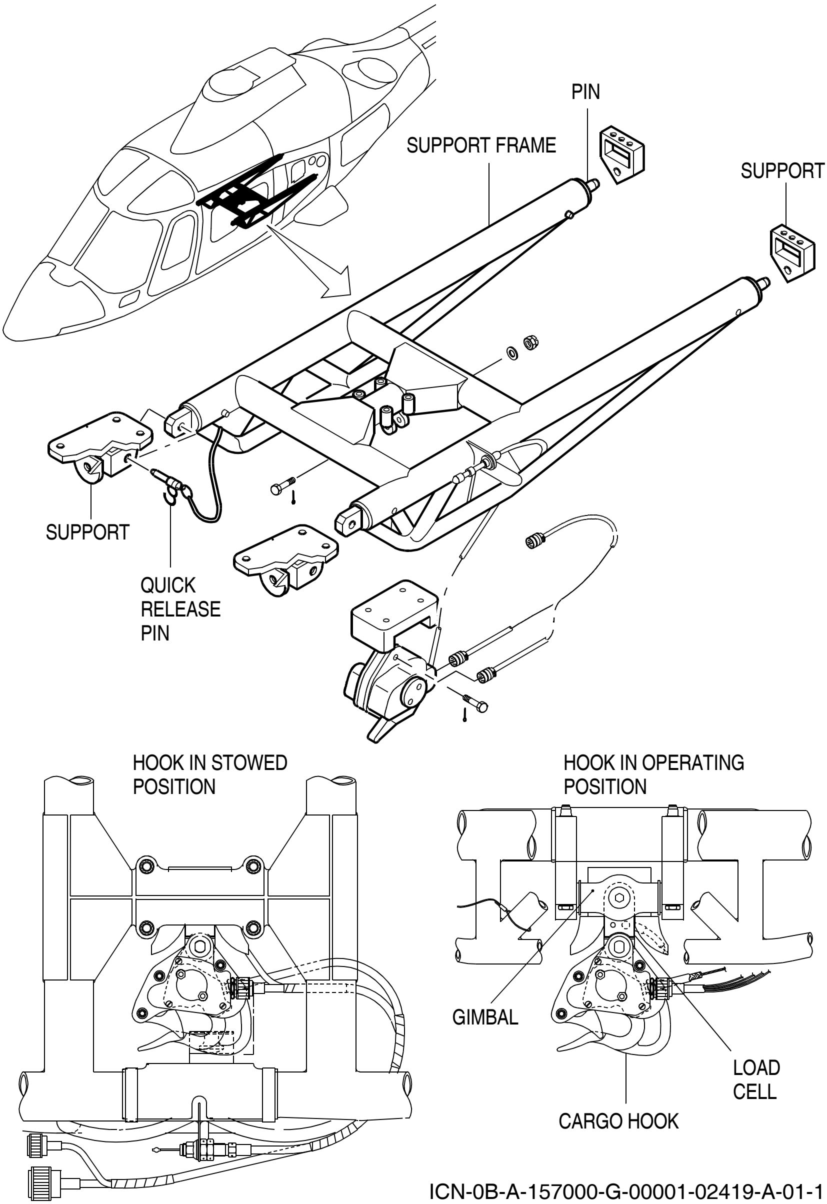

Figure 7-49 Cargo Hook Installation - Components Location

the cargo hook, the indicator is on the instrument panel. The indicator shows the weight of the load hung below the hook or the total weight (this weight includes the weight of the load, the hook and the cable). The indication on the indicator is divided by ten.

The indicator has two modes of operation: the run mode and the set-up mode.

RUN MODE - When power is first supplied to the system, the indicator shows the run mode. Then it does a self-test of the digits and indications. If an error is found, a message ("Err1" to "Err4") is shown.
Two types of information are shown in the run mode: indicator zero and indicator un-zero.
The zero function is used when the tare weight must be subtracted from the total weight (the tare weight is the weight of the hook and cable). The button on the right on the indicator must be pushed to reset the indication.
The un-zero function is used to add the tare weight that was subtracted with the zero function, to the weight applicable at the time.

SET-UP MODE - The set-up mode is used to set the indicator thus that it agrees with that helicopter and that hook load cell. The left and right buttons on the indicator are used to record the data. Refer to the table in Figure 7-50 for the menus, the related functions and the visualizations of the set-up mode.

CARGO HOOK INSTALLATION CONTROLS AND DISPLAYS

Refer to Figure 7-51.

Menu	Function	Display
Push the left button and scroll the menu pages.	Push the right button to see or change the data.	Push the left and right buttons at the same time to return to the run mode.
DAMP	Dampening level. It makes stable the indications on the display in accordance with the pilot's preference and the turbulence conditions.	Set the dampening level from 0 to 9 and push the right button. The best level is between 5 and 6.
CODE	Calibration code. It causes the indicator to agree with the load cell.	Mandatory data. The weight shown is not accurate if the correct code is not recorded.
0 in	Installation zero. It causes the indicator to agree with that load cell and that helicopter when the hook has no load.	Recommended data. The weight shown is not accurate if the display is not set to zero after you record the calibration code.
LOAD	Load cell calibration with a known weight.	Alternative procedure if the calibration code is not known.
SCALE	Not applicable	Not applicable
LB KG	Set the unit of measure for the weight (kg or lb).	Scroll the menu pages until the necessary symbol appears. Push the right button to set the unit of measure.
XX - V	Hardware and software version	It cannot be changed

Figure 7-50 Cargo Hook Installation - Set-up Modes

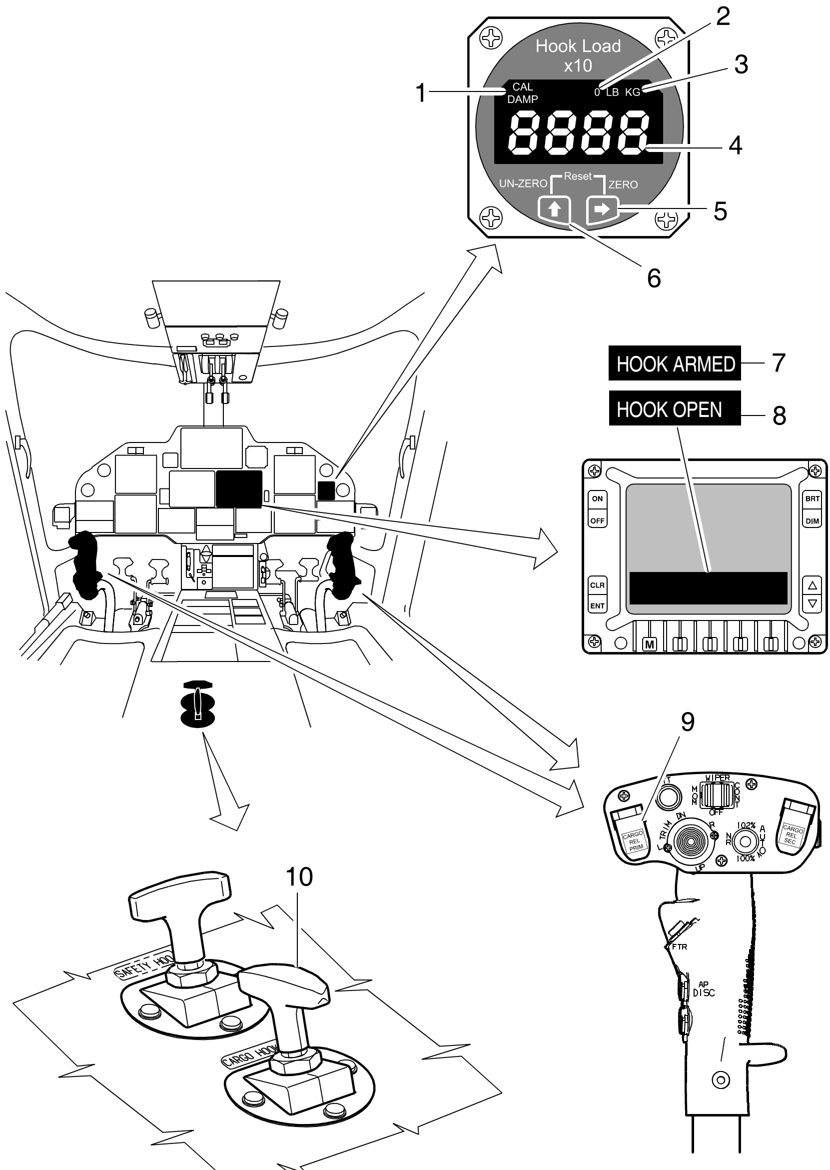

Figure 7-51 Cargo Hook Installation - Controls and Displays

Key to Figure 7-51

Ref.	Control/ Display	Function
1	CAL and DAMP legends	CAL - Shows the selection of the CAL menu page. DAMP - Shows the selection of the DAMP menu page.
2	Zero and un-zero legends	Zero - Shown when the right button is pushed to set the indicator to zero. Un-zero - Shown when the tare weight is added to the load weight.
3	Weight unit legend	Shows the set unit of measure.
4	Display	The shown data must be multiplied by ten to have the correct value of the load weight.
5	ZERO button	In the run mode, when first pushed, it sets to zero the digits on the display. If pushed again the display flashes.In the set-up mode, when pushed, it sets and changes the digits.
6	UN-ZERO button	In the run mode, when first pushed, it causes the last digits that were set to zero to come into view again on the display. If pushed again the display flashes.In the set-up mode, when pushed, it scrolls the menu pages.
7	HOOK ARMED caution message	Shown when the hook release switch is armed (guard up)
8	HOOK OPEN caution message	Shown when the cargo hook and the safety cargo hook are open (hook release switch operated).
9	CARGO REL PRIM switch	Guard down - The hook release switch is safe. Guard up - The hook release switch can be operated to open the hooks and release the safety cable and the cargo sling.
10	CARGO HOOK handle	If operated in an emergency, it opens the cargo hook to release the safety cable and the cargo sling.

PILOT EXTERNAL MIRROR INSTALLATION

The pilot external mirror is installed on the right side of the bottom fuselage (Figure 7-52). Only the pilot can see the mirror. Through the mirror the pilot can monitor the operation of the cargo hook, and the condition of the loads hung below the helicopter.

COPILOT EXTERNAL MIRROR INSTALLATION

The copilot external mirror is installed on the left side of the bottom fuselage (Figure 7-53). Only the copilot can see the mirror. Through the mirror the copilot can monitor the operation of the cargo hook, and the condition of the loads hung below the helicopter.

SAFETY CARGO HOOK INSTALLATION

The safety cargo hook is installed on the helicopter as a safety device for the primary cargo hook.

The installation includes:

– The hook
– The crossbeam
– The pivot
– The safety cable
– The electrical release system
– The emergency manual-release system.

The main components of the installation are shown in Figure 7-54.

The safety cargo hook is installed on the crossbeam. And the crossbeam is attached to two supports that are attached to the bottom of the center fuselage. These two supports are the same supports that attach the front part of the support frame of the primary cargo hook. Two quick-release pins attach the crossbeam.

The safety cable connects the safety cargo hook to the primary cargo hook. It also has a ring to which the cargo sling is attached

The electrical release system is an electrical circuit that includes the two cargo hook switches and an electro-mechanical device. The switches are installed on the handgrip of the pilot's and co-pilot's cyclic

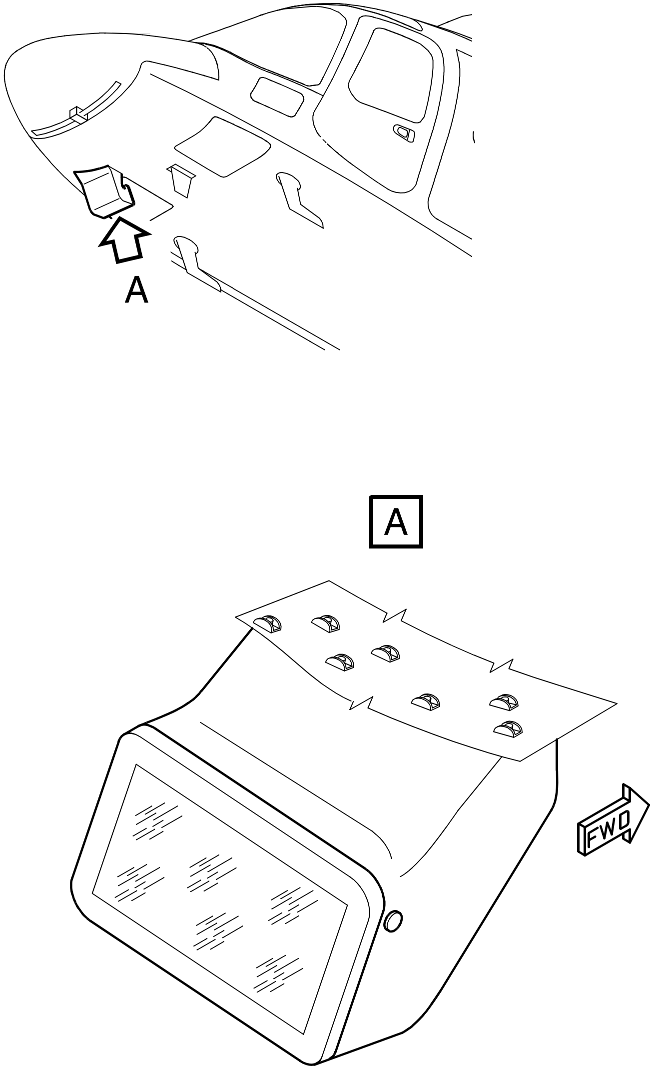

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Figure 7-52 Pilot External Mirror Installation - Components Location

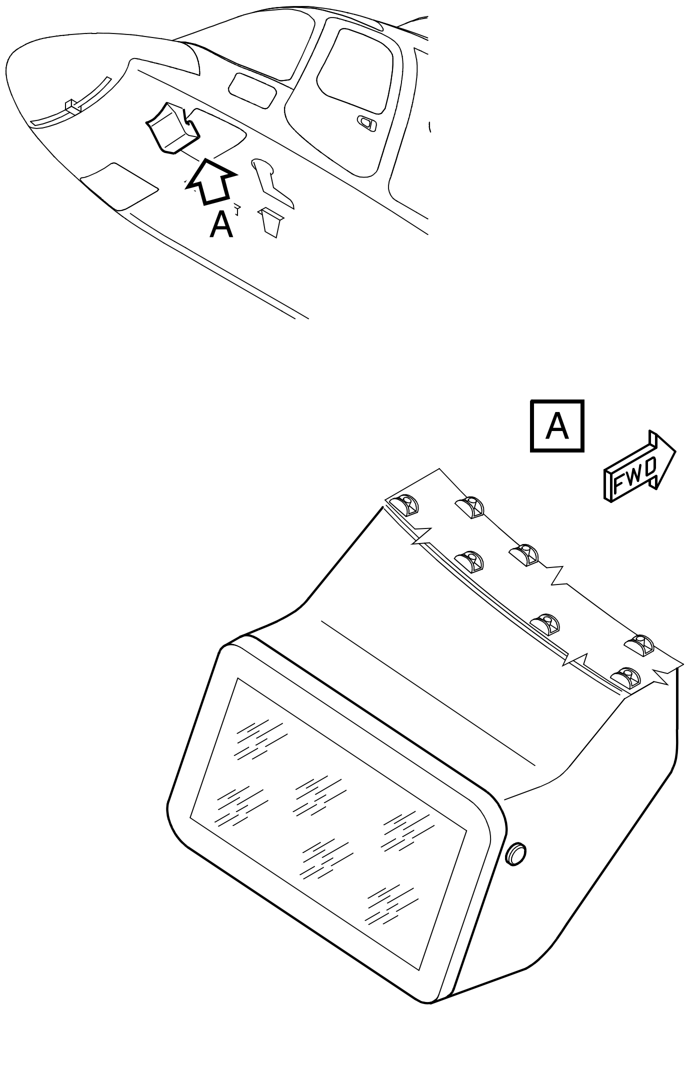

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Figure 7-53 Copilot External Mirror Installation - Components Location

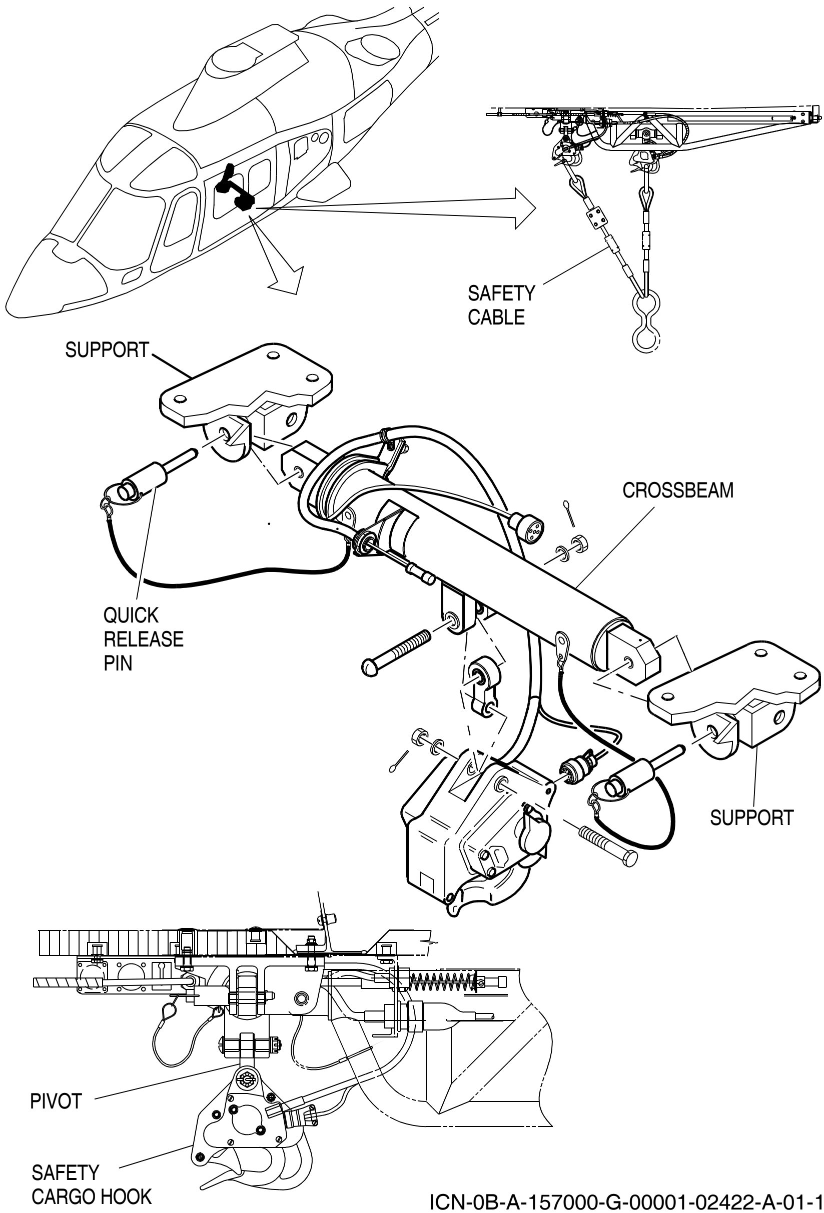

Figure 7-54 Safety Cargo Hook Installation - Components Location

stick. They are the same switches that are included in the primary cargo hook installation. The electro-mechanical device controls the opening of the safety cargo hook.

The installation is powered through the circuit breaker that follows:

– CARGO HOOK SEC (28 V DC BATT BUS).

This circuit breaker is installed on a panel on the right side of the overhead console.

If an electrical failure occurs, the crew can operate the emergency cargo-release handle (the SAFETY HOOK handle that is installed between the crew's seats) to release the safety cable with the cargo sling. Or a ground assistant can turn the manual release lever on the hook to release the load.

Because the safety cargo hook is used with the primary cargo hook, the correct sequence to open the hooks is as follows:

– First pull the SAFETY HOOK handle to open the safety cargo hook.
– Then pull the CARGO HOOK handle to open the primary cargo hook.

SAFETY CARGO HOOK INSTALLATION CONTROLS AND DISPLAYS

Refer to Figure 7-55.

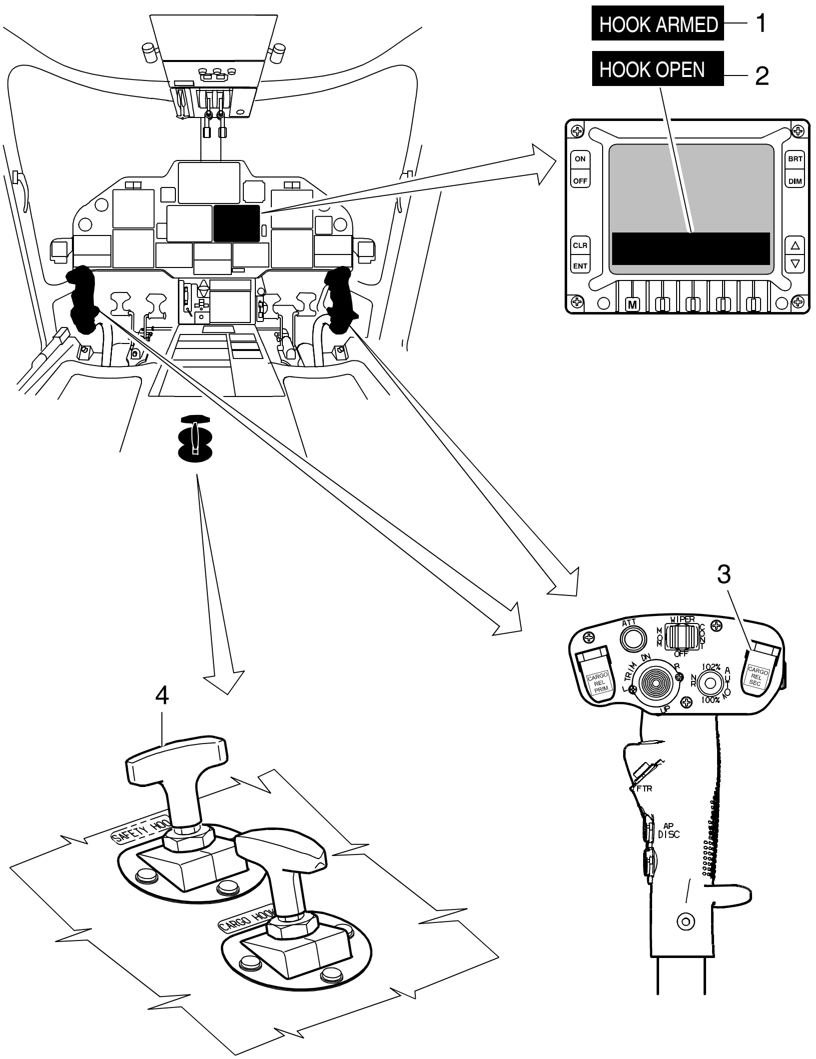

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Figure 7-55 Safety Cargo Hook Installation - Controls and Displays

Key to Figure 7-54

Ref.	Control/ Display	Function
1	HOOK ARMED caution message	Shown when the hook release switch is armed (guard up)
2	HOOK OPEN caution message	Shown when the primary and safety cargo hooks are open (hook release switch operated).
3	CARGO REL SEC switch	Guard down - The hook release switch is safe.
		Guard up - The hook release switch can be operated to open the hooks and release the safety cable and the cargo sling.
4	SAFETY HOOK handle	If operated in an emergency, it opens the safety cargo hook to release the safety cable and the cargo sling.

ADDITIONAL EXTERNAL MIRRORS INSTALLATION

The additional external mirrors are installed on the right and the left side of the helicopter nose (Figure 7-55A).
Through the mirrors the pilot can monitor the operation of the cargo hook, the hoist operator and the condition of the loads hung below the helicopter.

The installation consists of the right and the left external mirror, the right and the left support assembly, a switch on dedicated control panel located on the interseat console, a slew switch on the pilot cyclic grip and the necessary hardware and wires.
The right and the left support assembly, composed each one of two support rods, holds the external mirrors. Theree quick release pins attach the support rods to the fuselage structure.
The operator can control the right mirror from the cabin. With the switch on the interseat console, the operator can set the mirror heating on or off. With the slew switch on the pilot cyclic grip, the operator can adjust the position of the mirror.

The additional external mirrors installation is powered through the toggle-type circuit breaker that follows:

– MIRROR.

ADDITIONAL EXTERNAL MIRRORS INSTALLATION CONTROLS AND DISPLAYS

Refer to Figure 7-55B.

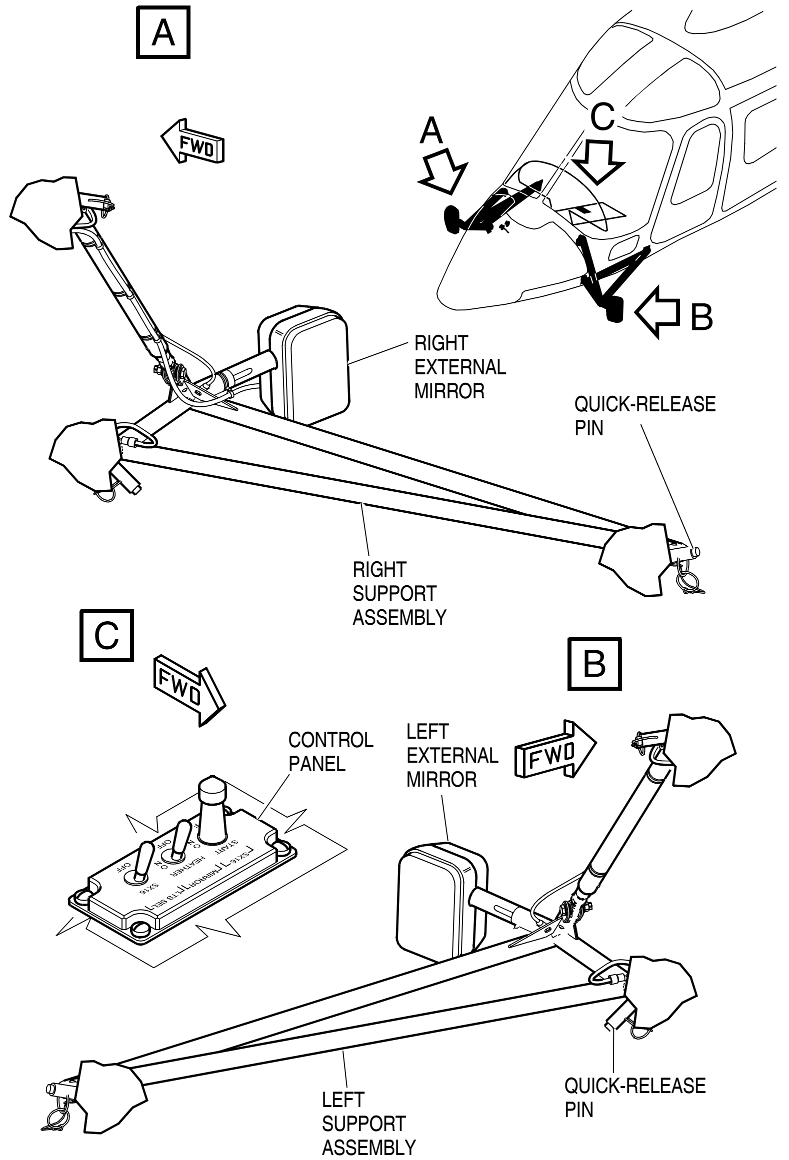

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Figure 7-55A Additional External Mirrors Installation - Component Location

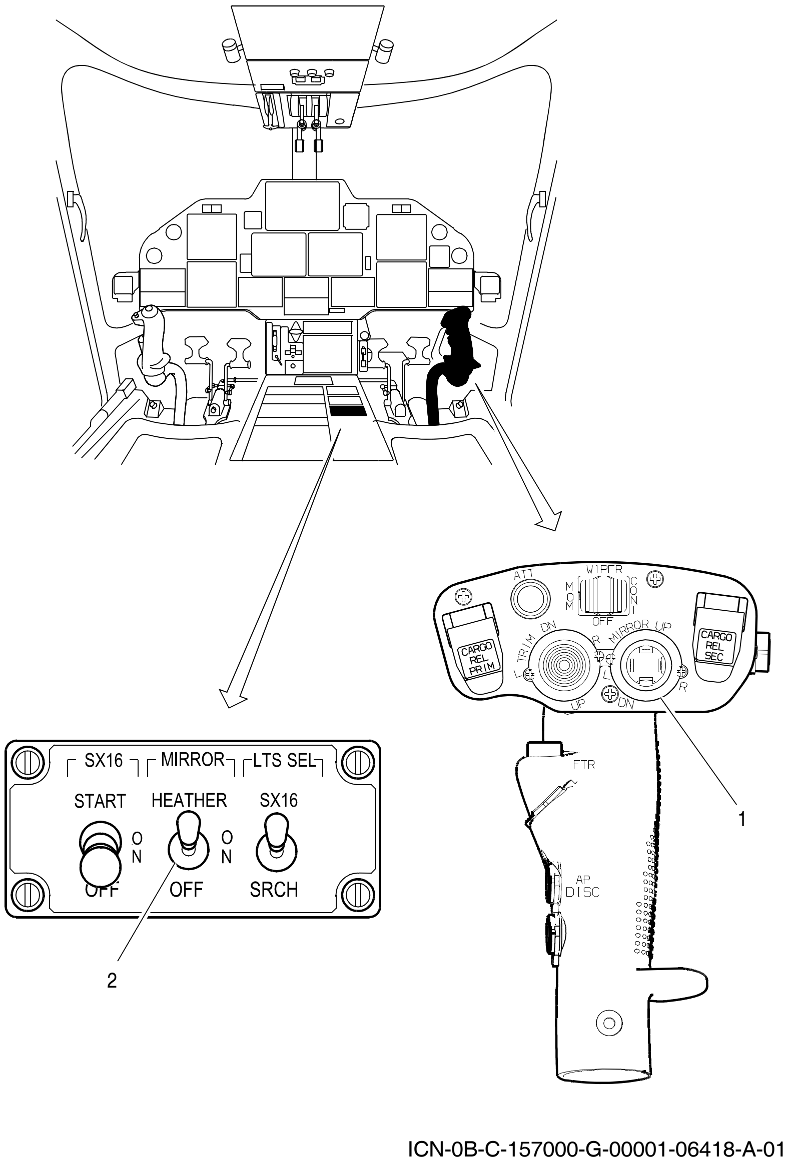

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Figure 7-55B Additional External Mirrors Installation - Controls and Displays

Key to Figure 7-55B

Ref.	Control/Display	Function
1	MIRROR slew switch	The 8-way directional slew switch moves the right external mirror up, down, right or left.
2	MIRROR HEATER switch	ON position - Energizes the heating circuit of the right external mirror. OFF position - De-energizes the heating circuit of the right external mirror.

FIRE PROTECTION

(Chapter 26)

GENERAL

The fire protection system, detects and indicates the presence of fire or overheating in the engine compartments and distributes fire extinguishing agent to all protected areas.

The fire protection system includes:

– The detection system

– The extinguishing system.

The schematic diagram of the fire protection system is shown in Figure 7-56.

A portable fire extinguisher is installed in the cockpit.

FIRE PROTECTION SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-57.

DETECTION SYSTEM

The detection system detects and indicates to the pilot the presence of fire or overheating in the engine compartments. The system consists of two independent circuits (one for each engine) each comprising a fire wire detector installed on the engine firewalls and connected to the DAU by means of a wiring harness. Each fire detector consists of a responder connected to a sensing element. When the sensing element is exposed to high temperatures, a pressure switch in the responder closes the warning circuit, causing the ENG 1 (2) FIRE warning message to appear on the pilot's EDU and the an aural warning message ("engine one (two) fire") to be heard in the pilots' headsets. The second pressure switch is used to monitor the integrity of the sensing element and to alert the pilot in the event of failure of the

detection system through the display of #1 (#2) FIRE DET caution message.

The main components of the detection system are shown in Figure 7-58.

The system is powered through the circuit breakers that follow:

– FIRE DET 1 (ESS BUS #1)
– FIRE DET 2 (ESS BUS #2).

EXTINGUISHING SYSTEM

The fire extinguishing system consists of two sub-systems, each of which can be used to discharge an extinguishing agent into the selected engine compartment.
Each system consists of a bottle containing the extinguishing agent (HALON 1301) that is located behind the engine compartment and the necessary tubing to carry the extinguishing agent into the engine compartments.
The extinguishing agent is discharged through a forward and an aft outlet. The installation also includes a control panel located on the overhead console (Figure 7-57).
Each bottle is provided with an electronic pressure sensor that monitors the pressure in the bottle and transmits a signal to the DAU, which then causes the FIRE BTL #1 (#2) caution message to be shown on the pilot's EDU when a low pressure is detected. Each bottle is also connected to a red colored disc-type discharge indicator, visible from the exterior of the helicopter. If the red disk is visible and flush, the bottle is fully charged; otherwise the bottle is empty. An interconnection pipe connects the two bottles to enable the bottle on one side of the helicopter to be used to extinguish a fire on the other side of the helicopter.

The main components of the extinguishing system are shown in Figure 7-59.

The system is powered through the circuit breakers that follow:

– FIRE EXTING 1 (ESS BUS #1)
– FIRE EXTING 2 (ESS BUS #2)

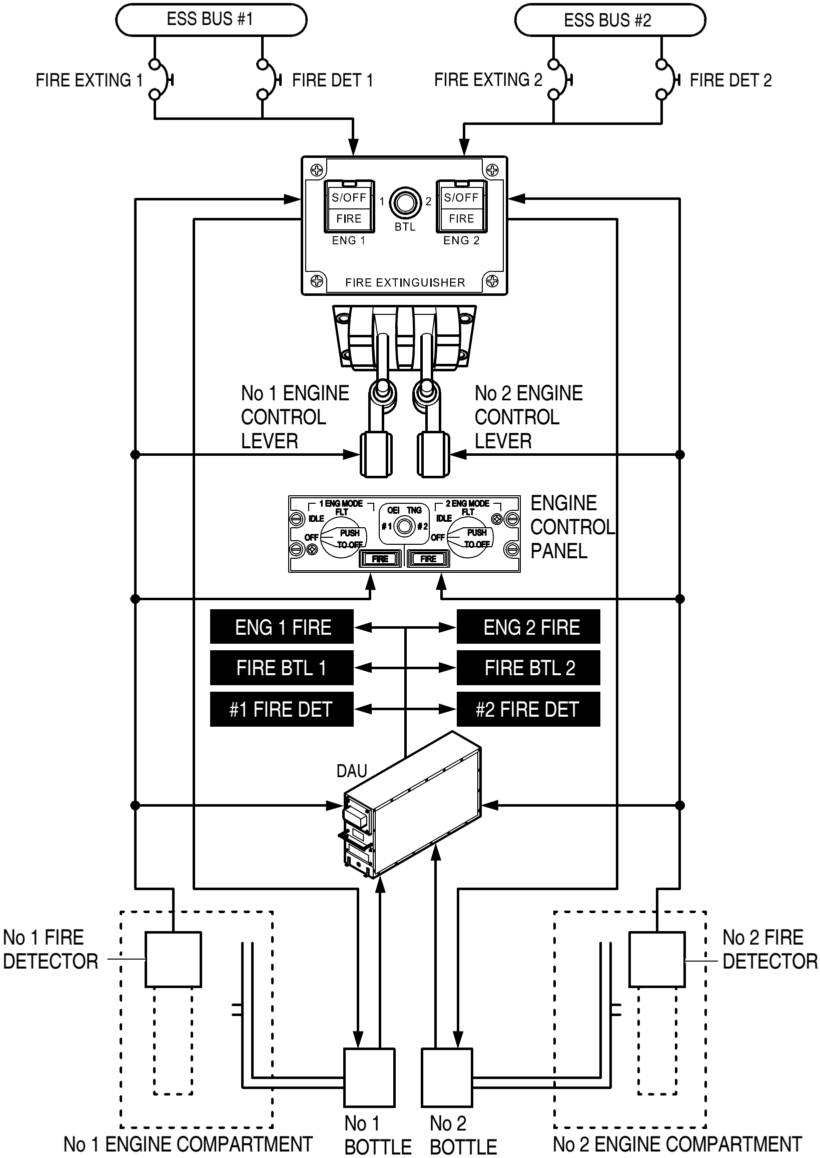

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Figure 7-56 Fire Protection System - Schematic Diagram

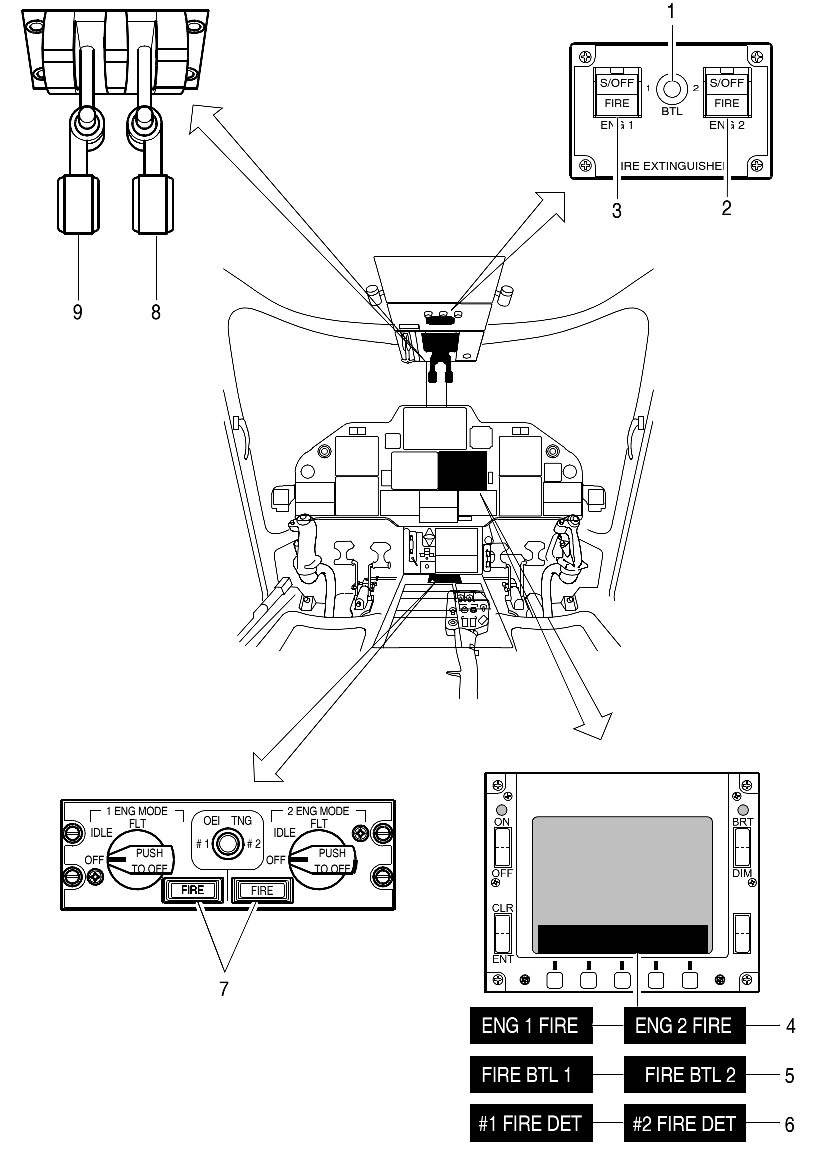

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Figure 7-57 Fire Protection System - Controls and Displays

Key to Figure 7-57

Ref.	Control/Display	Function
1	BTL switch	BLT 1 - The content of the left (No 1) bottle is discharged in selected engine compartment. BLT 2 - The content of the right (No 2) bottle is discharged in selected engine compartment.
2	ENG 2-S/OFF FIRE lighted push-button switch	FIRE light off - No fire or overheating in No 2 engine compartment (normal operation). FIRE light on - Fire or overheating in No 2 engine compartment. S/OFF light off - The squib firing circuits of the discharge valves connected to the No 2 engine are de-armed. S/OFF light on (red light) - The squib firing circuits of the discharge valves connected to the No 2 engine fuel shut-off valve and heating system bleed air shut-off valves are closed.
3	ENG 1-S/OFF FIRE lighted push-button switch	FIRE light off - No fire or overheating in No 1 engine compartment (normal operation). FIRE caption on - Fire or overheating in No 1 engine compartment. S/OFF light off (green light) - The squib firing circuits of the discharge valves connected to the No 1 engine are de-armed. S/OFF light on (green light) - The squib firing circuits of the discharge valves connected to the No 1 engine fuel shut-off valve and heating system bleed air shut-off valves are closed.
4	ENG 1 (2) FIRE warning and aural message	Displayed when an overheat or fire condition is detected in the relative No 1 or No 2 engine compartment. The aural message is sent to the pilots's headsets.
5	FIRE BTL 1 (2) caution message	Shown when a condition of low pressure in the No 1 or No 2 extinguisher bottle exists.
6	#1 (2) FIRE DET caution message	Shown when the No 1 or No 2 fire detection system is inoperative.

Key to Figure 7-57 (Continued)

Ref.	Control/Display	Function
7	FIRE warning light	Comes on (red light), when a fire condition is detected, in the No 1 or No 2 engine compartment.
8	No 2 engine control lever knob	Comes on (red light) when fire is detected in the No 2 engine.
9	No 1 engine control lever knob	Comes on (red light) when fire is detected in the No 1 engine.

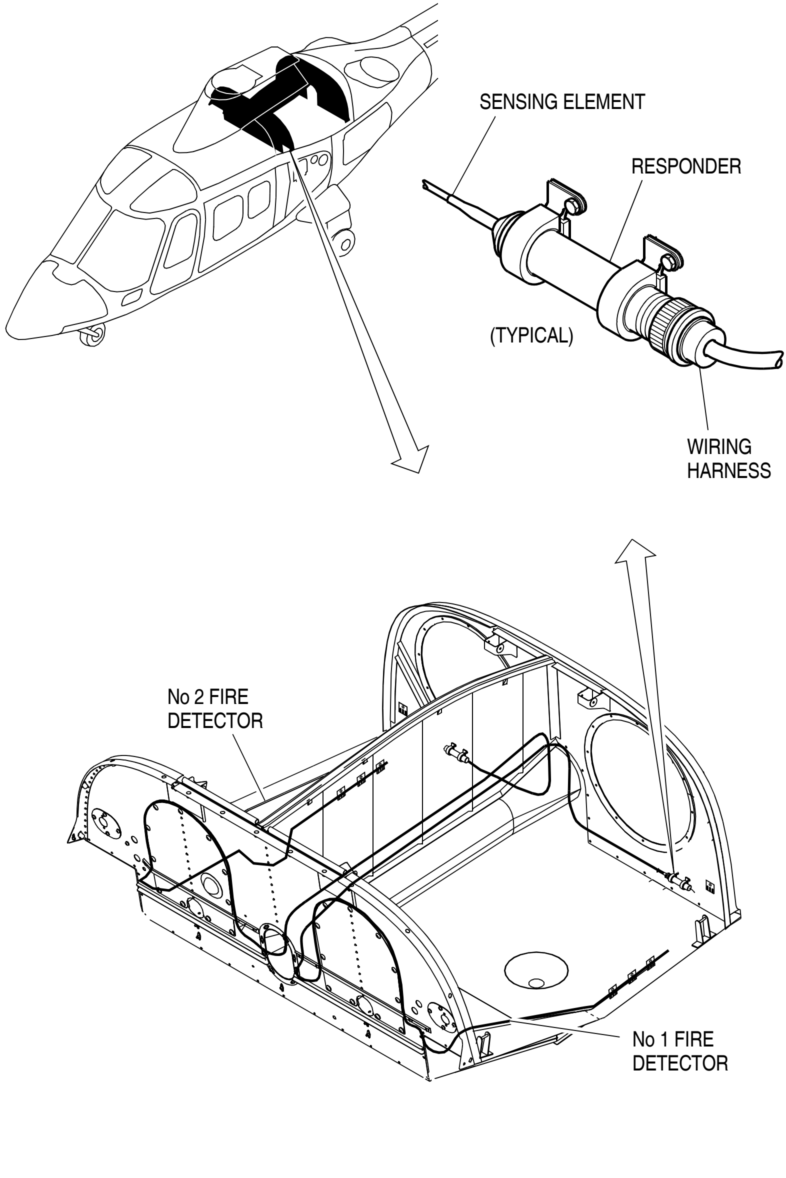

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Figure 7-58 Fire Detection System - Component Location

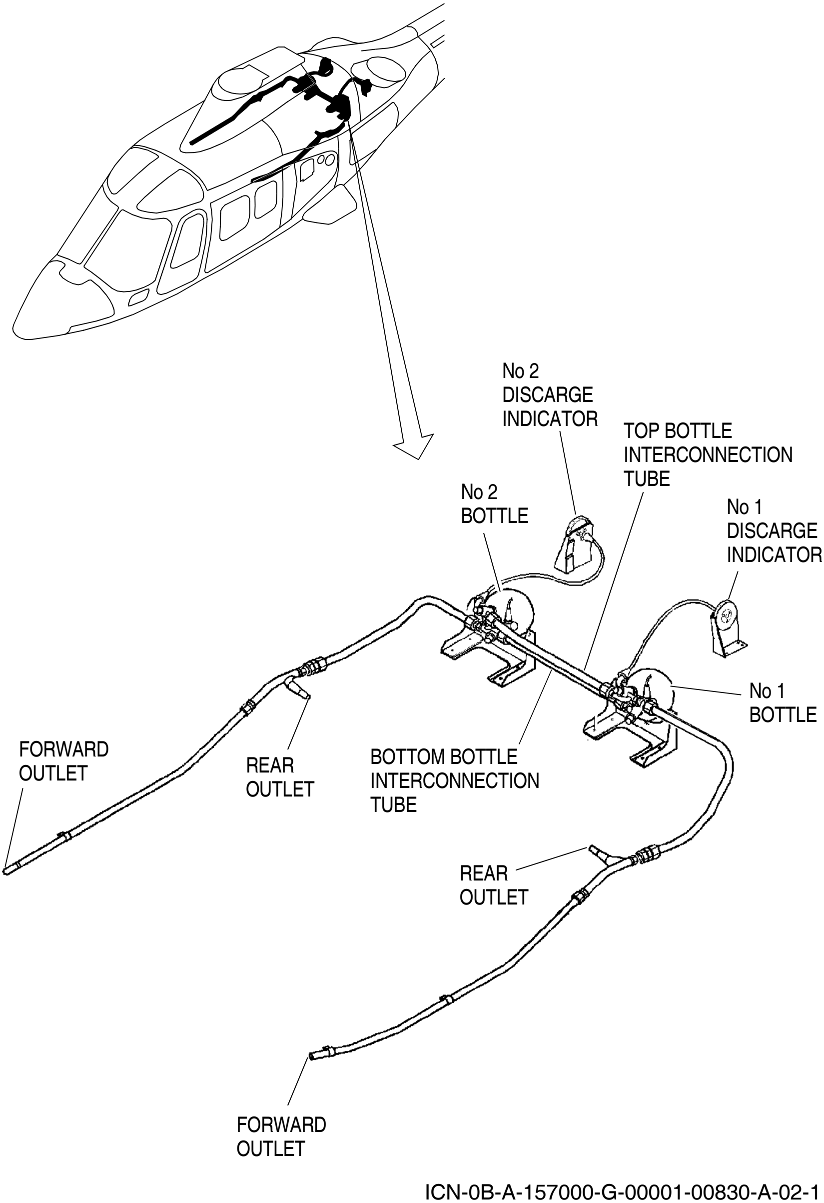

Figure 7-59 Extinguishing System - Component Location

PORTABLE FIRE EXTINGUISHER

A portable, manually operated, fire extinguisher is installed on the right side of the cockpit below the seat of the pilot (Figure 7-60).

A quick release clamp allows for rapid removal of the portable fire extinguisher in the event of a fire.

The extinguishing agent is HALON 1211 and the extinguisher can be used against small carbonaceous fires, flammable liquid fires and electrical fires. The extinguisher has a discharge time of 10 seconds. However once the extinguisher has been used it must be replaced at the first possible opportunity even if there is still some extinguishing agent in the bottle.

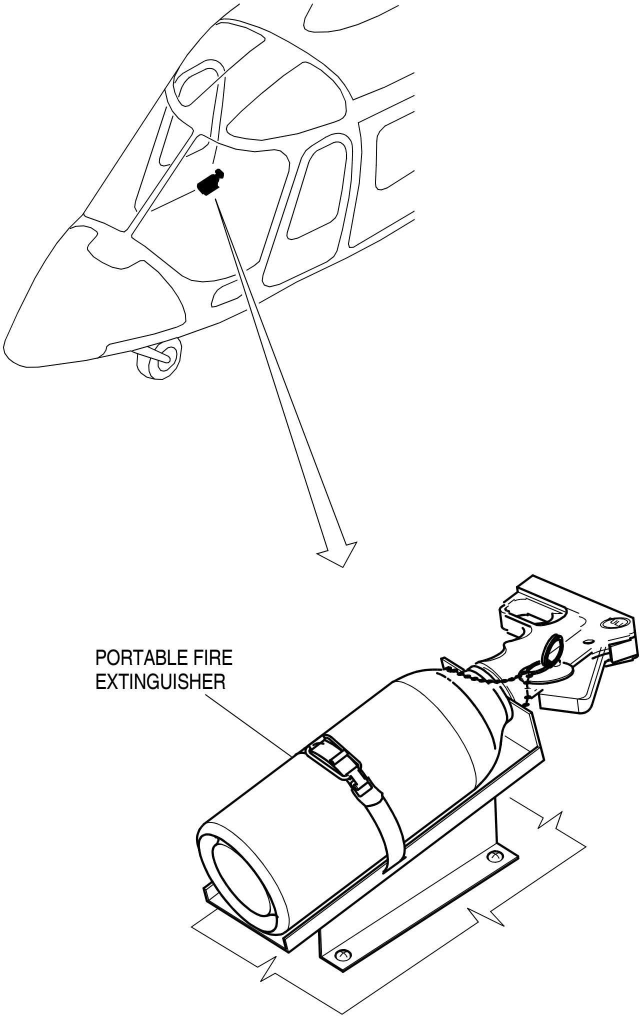

Figure 7-60 Portable Fire Extinguisher

FUEL

(Chapter 28)

GENERAL

This system has the primary function to store and deliver fuel to the engines.

The fuel system includes:

– The storage system
– The distribution system
– The indicating system.

The schematic diagram of the fuel system is shown in Figure 7-61.

Refer to Figure 7-62 for the electrical interfaces of the fuel system.

FUEL SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-63.

STORAGE SYSTEM

The fuel storage system has two main bottom tanks and one main top tank. The tanks are refueled via a filler cap on the right side of the main top tank. Each main bottom tank supplies fuel to its associated engine (right tank to the No 2 engine, left tank to the No 1 engine).
The bottom tanks are gravity-fed from the top tank.
The main tanks are of the bladder type and made of rubberized fabric. Each tank compartment is sealed to prevent fuel from leaking into other parts of the helicopter in the event of leaks. All fuel tank compartments have drain and venting holes.

The main components of the storage system and their location are shown in Figure 7-64.

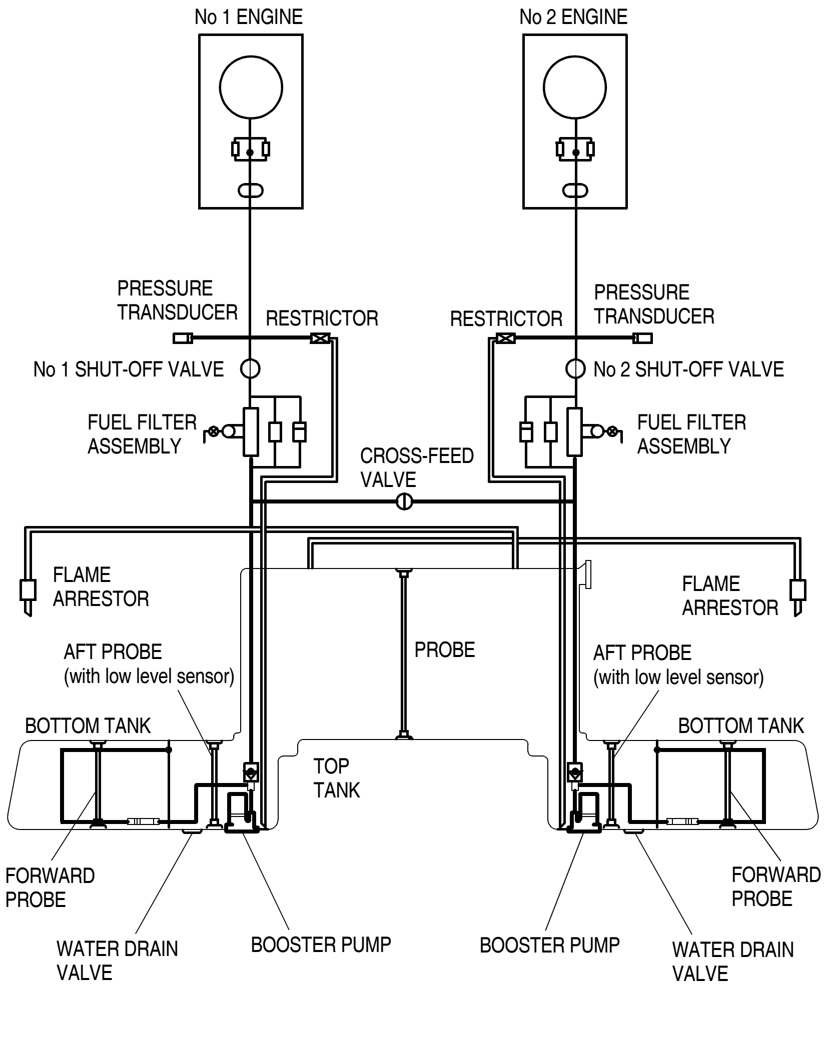

Figure 7-61 Fuel System - Schematic Diagram

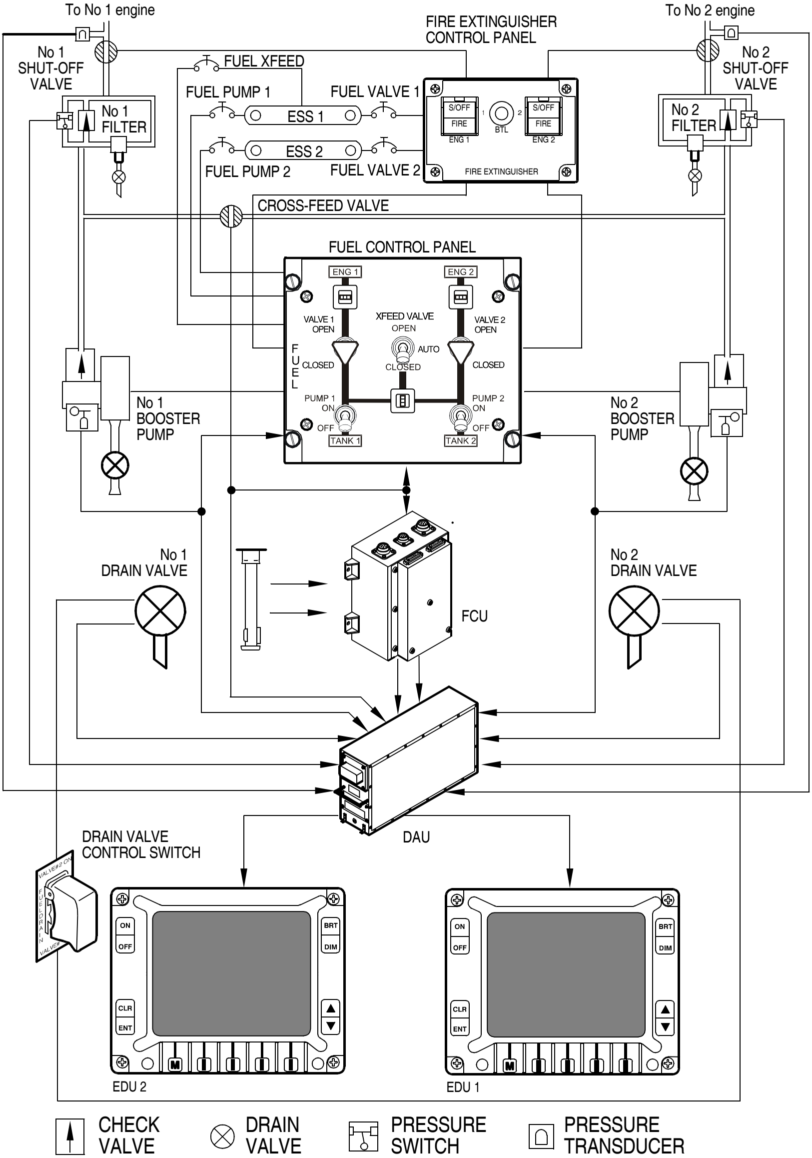

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Figure 7-62 Fuel System - Electrical Interfaces

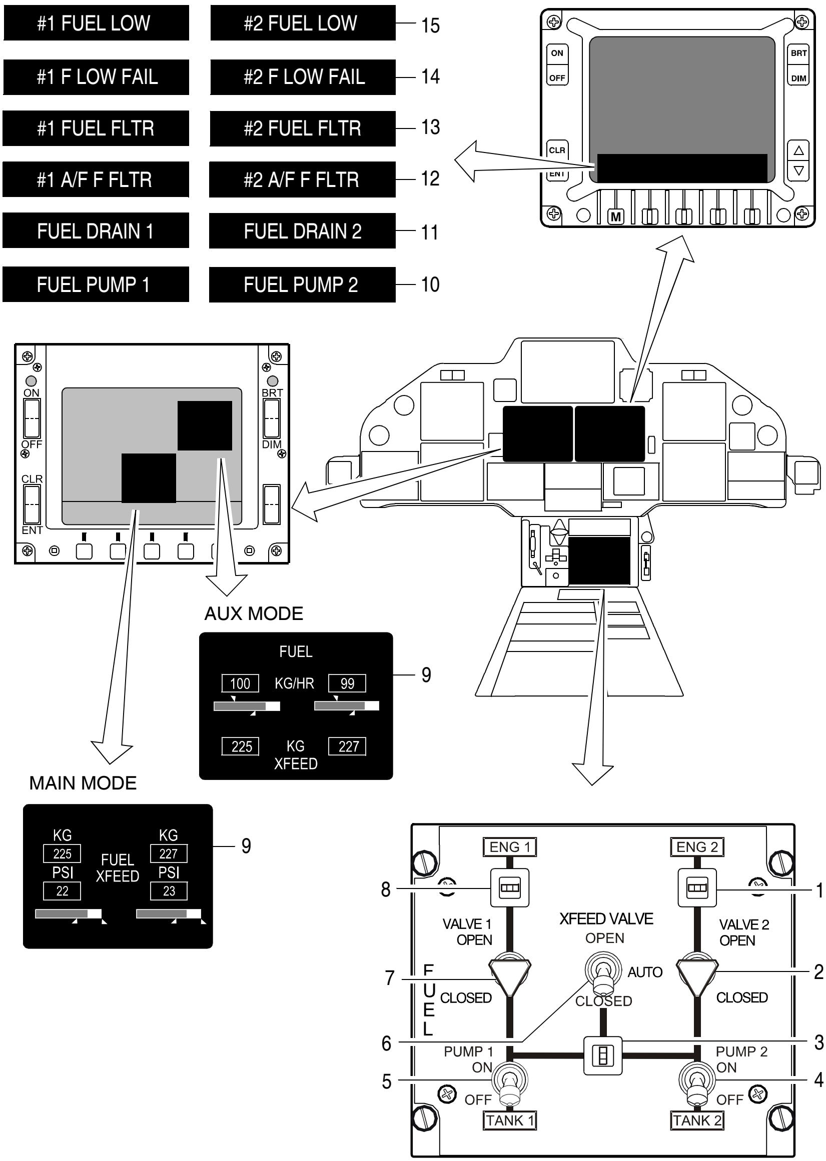

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Figure 7-63 Fuel System - Controls and Displays

Key to Figure 7-63

Ref.	Control/Display	Function
1	ENG 2 shut-off valve position indicator	White line vertical - the No 2 engine shut-off valve is open. White line horizontal - the No 2 engine shut-off valve is closed.
2	VALVE 2 switch	OPEN - the No 2 engine shut-off valve is open. CLOSED - the No 2 engine shut-off valve is closed.
3	XFEED VALVE position indicator	Vertical white line - the cross-feed valve is closed. Horizontal white line - the cross-feed valve is open.
4	PUMP 2 switch	ON - the No 2 engine booster pump is energized. OFF - the No 2 engine booster pump is de-energized.
5	PUMP 1 switch	ON - the No 1 engine booster pump is energized. OFF - the No 1 engine booster pump is de-energized.
6	XFEED VALVE switch	AUTO - the cross-feed valve is closed but opens automatically in case of failure of one booster pump. OPEN - the cross-feed valve is open. CLOSED - the cross-feed valve is closed.
7	VALVE 1 switch	OPEN - the No 1 engine shut-off valve is open. CLOSED - the No 1 engine shut-off valve is closed.
8	ENG 1 shut-off valve position indicator	White line vertical - the No 1 engine shut-off valve is open. White line horizontal - No 1 engine shut-off valve is closed.

Key to Figure 7-63 (Continued)

Ref.	Control/Display	Function
9	FUEL display (MAIN mode)	Shows the fuel quantity and the pressure for the No 1 and No 2 engine circuits. XFEED indication gives the position of the cross-feed valve.
	FUEL display (AUX mode)	Shows the fuel flow rate and the fuel quantity for the No 1 and No 2 engine circuits. XFEED indication gives the position of the cross-feed valve.
10	FUEL PUMP 1 (2) caution message	Shown when the pressure at the outlet port of the relevant booster pump drops below 3.5 psi.
11	FUEL DRAIN 1 (2) caution message	Shown when the related fuel drain switch is in open position.
12	# 1 (# 2) A/F F FLTR caution message	Shown when the differential pressure between inlet and outlet ports of the relevant airframe fuel filter is at 1.4 psi indicating a clogged filter element.
13	# 1 (# 2) FUEL FLTR caution message	Shown when the differential pressure of the related engine fuel filter is between 9.1 and 11.2 psi.
14	# 1 (# 2) F LOW FAIL caution message	Shown when there is a failure in the fuel low level sensor. No fuel low indication is available.
15	# 1 (# 2) FUEL LOW caution message	Shown when the quantity of fuel remaining in each bottom tank is less than 32 kg.

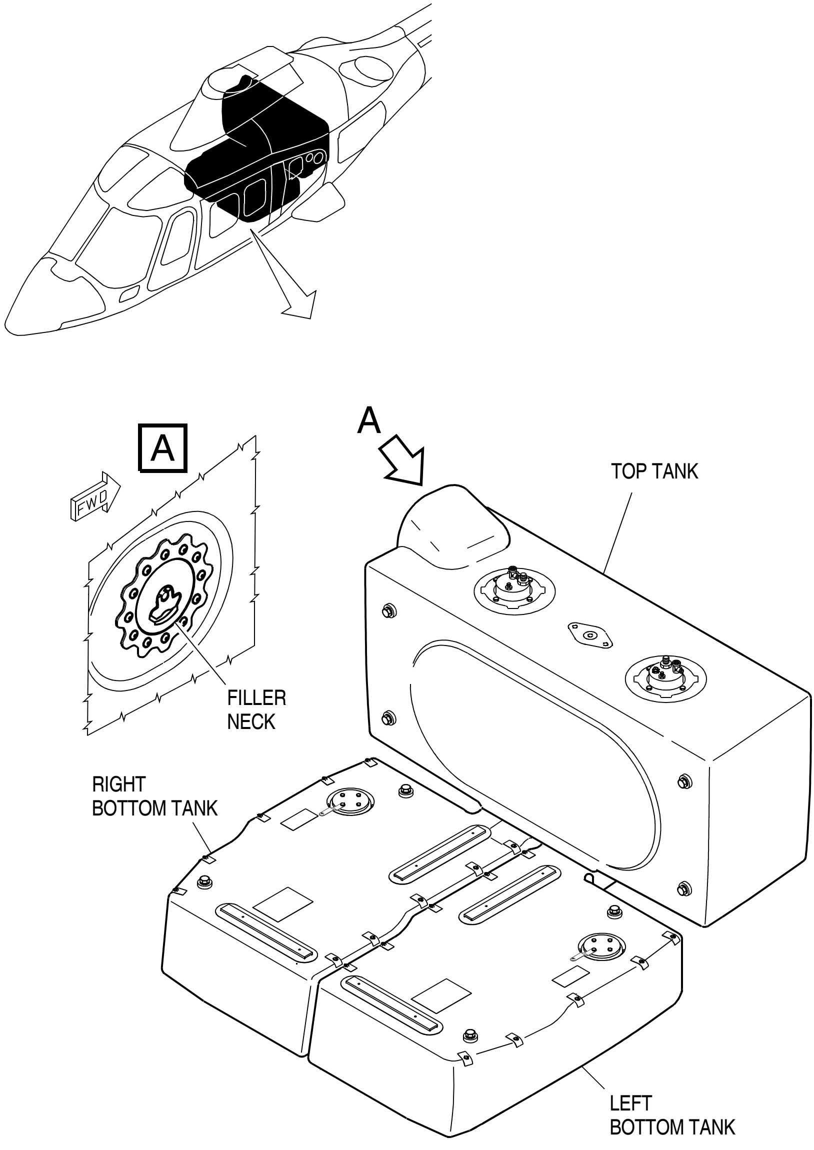

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Figure 7-64 Storage - Component Location

DISTRIBUTION SYSTEM

The distribution system allows the flow of fuel from the booster pumps to the fuel pump and filter group of each engine. The distribution system consists of two independent circuits each of which supplies the associated engine. When activated, a cross-feed valve allows the fuel from one circuit to supply both engines.

A fuel control panel, located on the instrument panel provides the primary interface between the pilots and the fuel system. The panel controls the booster pumps, the shut-off valves and the cross-feed valve (Figure 7-63).

The main components of the distribution system are:

- The booster pumps
- The fuel filters (airframe fuel filters)
- The shut-off valves
- The cross-feed valve.

The main components of the distribution system are shown in Figure 7-65.

The electrically powered booster pumps are of the submerged centrifugal type and supply fuel to the related engine via supply lines. The nominal fuel flow rate for each pump is 458 kg/h. A pressure switch is fitted to each pump. If the pressure of a pump falls below 3.5 ±0.4 psi, the switch activates the FUEL PUMP 1 (2) caution message. At the same time the cross-feed valve will open automatically.

A fuel filter is installed on each engine supply line. It is fitted with a 10 micron filter element. A differential pressure switch senses the loss of pressure across the filter. When the loss of pressure is between 9.1 to 11.2 psi the pressure switch activates a caution message (# 1 or # 2 FUEL FLTR). A by-pass valve is incorporated in the filter. If the filter becomes blocked, unfiltered fuel will still flow to the engines.

A fuel shut-off valve is installed on each fuel supply line and allows the pilot to cut-off the fuel flow to the engines. If one of the engines fails, fuel to the operational engine can be supplied by both fuel tanks. This is done by closing the shut-off valve of the failed engine and setting the cross-feed valve to the OPEN position.

The cross-feed valve is fitted to the cross-feed line that connects the two fuel supply lines. It is controlled by the “XFEED VALVE” switch

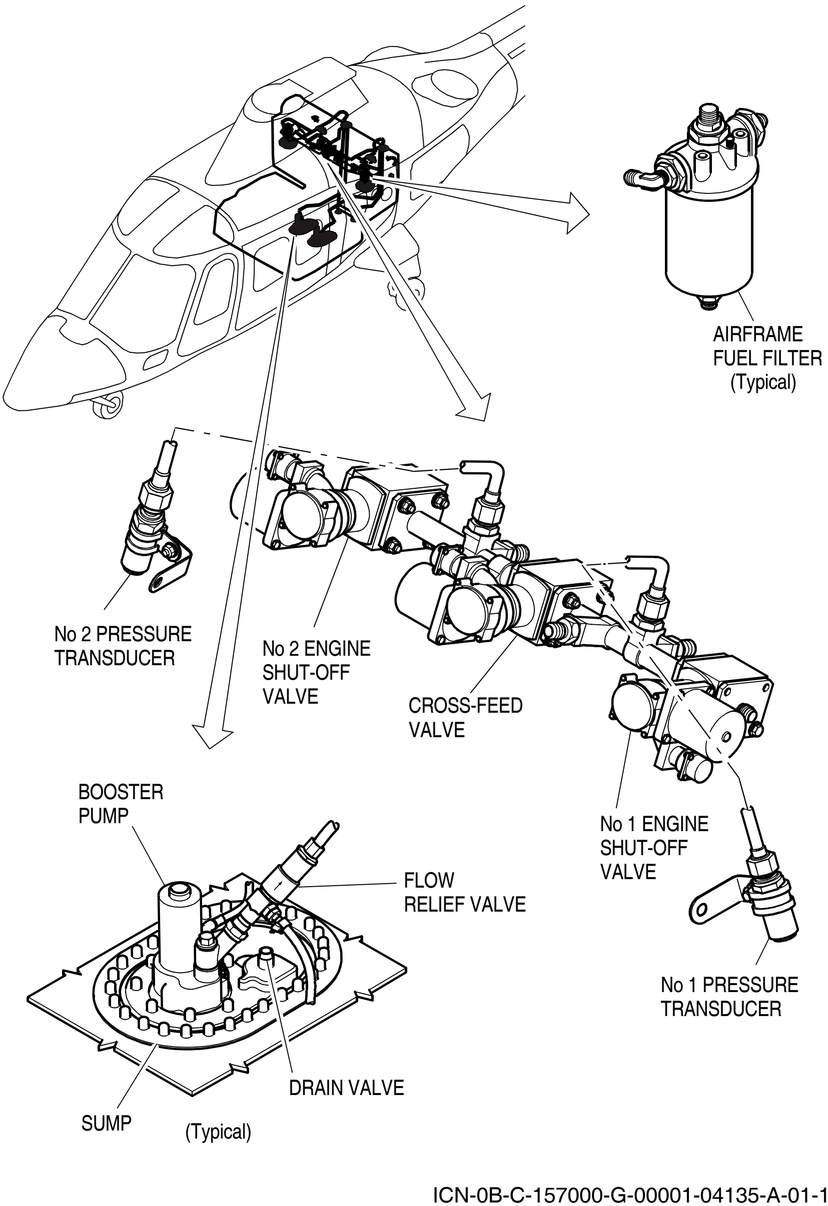

Figure 7-65 Distribution System - Component Location

that is located on the fuel control panel, and by the pressure switches on the booster pumps.

The system is powered through the circuit breakers that follow:

– FUEL XFEED (28 V DC ESS BUS #1)
– FUEL PUMP 1 (28 V DC ESS BUS #1)
– FUEL VALVE 1 (28 V DC ESS BUS #1)
– FUEL PUMP 2 (28 V DC ESS BUS #2)
– FUEL VALVE 2 (28 V DC ESS BUS #2).

INDICATING SYSTEM

The main components of the indicating system are shown in Figure 7-66.

This system gives a continuous indication of the amount of fuel contained in the fuel tanks and of the pressure in the system.

The system includes the units that follow:

– The Fuel Computing Unit (FCU)
– The fuel quantity probes
– The fuel pressure transducers.

Five variable-capacitance fuel-quantity probes are installed inside the tanks (two probes in the left bottom tank, two in the right bottom tank and one probe in the top tank). When the amount of fuel in the tank changes, the probes send an electrical signal to the FCU, which processes it and then sends proportional output currents to the DAU. The latter sends the fuel quantity signals to the co-pilot’s EDU where fuel quantity is continuously displayed.

The rear probes of the bottom tanks have a low-level sensor. When the fuel quantity in the related tank becomes less than 33 kg, the sensor sends a signal to the DAU that causes the #1 (#2) FUEL LOW caution message to come in view on the pilot’s EDU.

A pressure transducer, installed on the fuel supply lines of each engine, sends pressure information signal to the DAU, which then sends them to the co-pilot’s EDU.

The system is powered through the circuit breakers that follow:

– FUEL QTY 1 (28V DC ESS BUS #1)
– FUEL QTY 2 (28V DC ESS BUS #2).

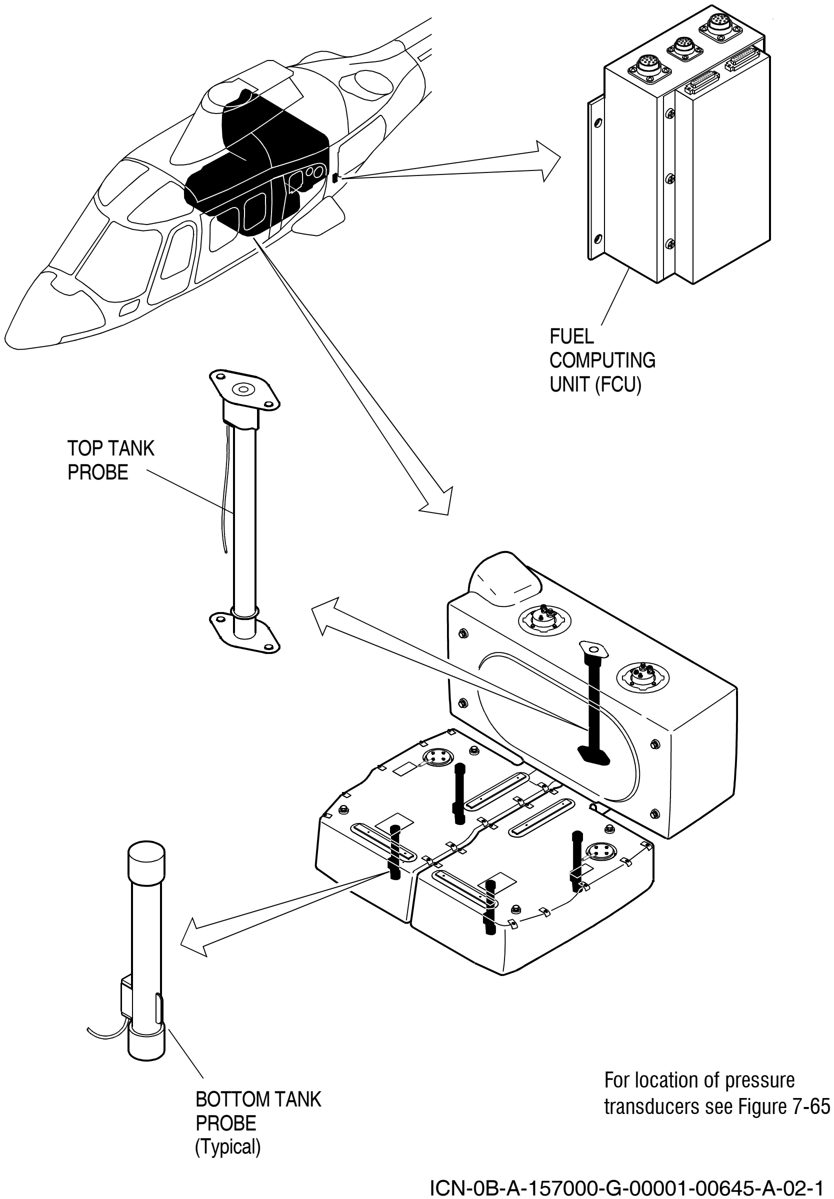

Figure 7-66 Indicating System - Component Location

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HYDRAULIC POWER

(Chapter 29)

GENERAL

The hydraulic power system includes:

– The main hydraulic system
– The utility hydraulic system
– The indicating system.

HYDRAULIC POWER SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-67.

MAIN HYDRAULIC SYSTEM

The main hydraulic system includes two independent sub-system:

– The No 1 main hydraulic system
– The No 2 main hydraulic system.

The main components of the main hydraulic systems is shown in Figures 7-68 and 7-70.

The schematic diagram of the main hydraulic system are shown in Figures 7-69 and 7-71.

Both systems (No 1 and No 2) supply the hydraulic power for operation of the flight controls. Each system, operating at a pressure of 1500 psi, includes a suction circuit, a pressure circuit, a return circuit and a bypass circuit.
In normal operation, the fluid is pressurized by the pump that is driven by the transmission, and supplied to the main rotor and the tail rotor servoactuators after passing through a filter group and an accumulator. The pump gets the hydraulic fluid from the reservoir. The pump is fitted with a flow regulating device that returns the excess fluid to the reservoir.

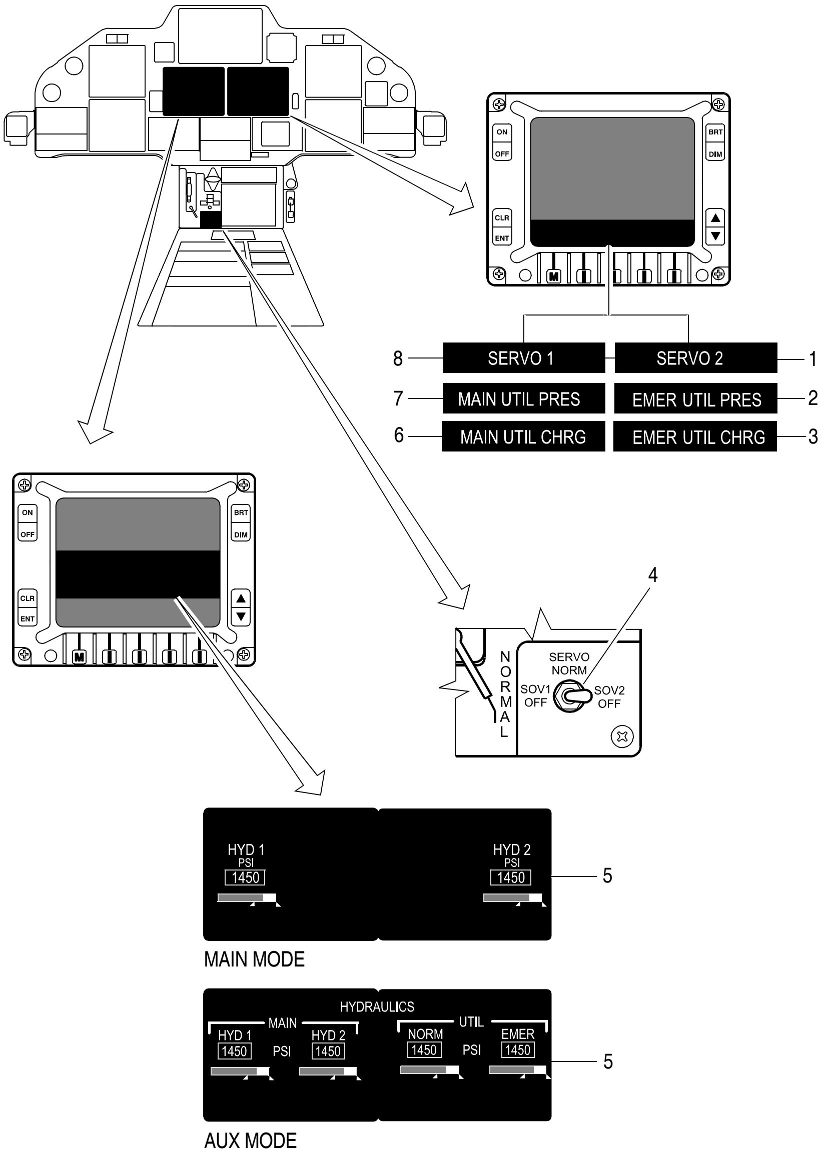

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Figure 7-67 Hydraulic Power System - Controls and Displays

Key to Figure 7-67

Ref	Control/Display	Function
1	SERVO 2 caution message	Shown when the pressure in the No 2 main hydraulic system drops below 1200 psi.
2	EMER UTIL PRESS caution message	Shown when the pressure in the emergency utility system drops below 1140 psi.
3	EMER UTIL CHRG caution message	The solenoid valve of the emergency utility hydraulic system is open. The system is inoperative.
4	SERVO switch	NORM - The No 1 and the No 2 main hydraulic systems are operating. 1 OFF - No 1 system is off. 2 OFF - No 2 system is off.
5	HYD 1, HYD 2, NORM, EMERG indications and horizontal bars	MAIN mode - HYD 1 and HYD 2 indications show the pressure values of the No 1 and No 2 main hydraulic systems. AUX mode - HYD 1 and HYD 2 indications show the pressure values of the No 1 and No 2 main hydraulic systems. NORM and EMERG indications show the pressure values of the normal and emergency circuits of the utility hydraulic system.
6	MAIN UTIL CHRG caution message	The solenoid valve of the main utility hydraulic system is open. The system is inoperative.
7	MAIN UTIL PRES caution message	Shown when the pressure in the main utility accumulator drops below 500 psi.
8	SERVO 1 caution message	Shown when the pressure in the No 1 main hydraulic system drops below 1200 psi.

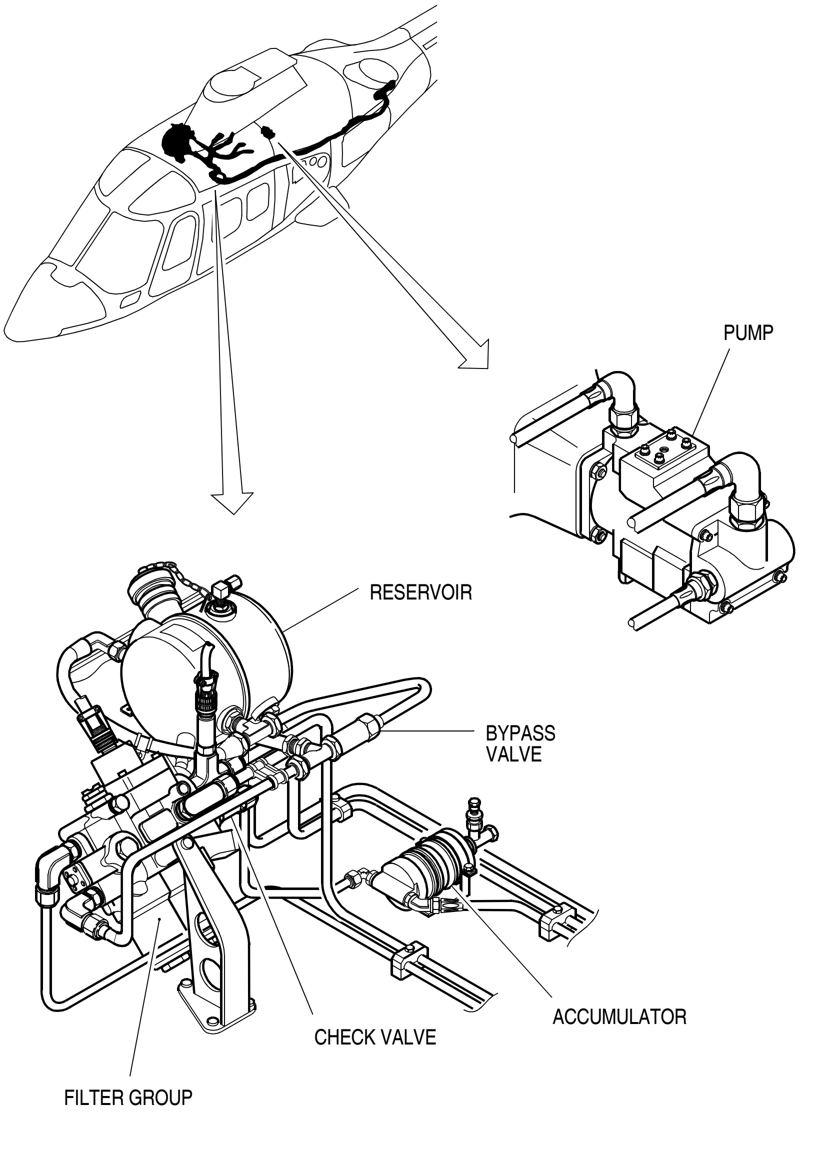

Figure 7-68 No 1 Main Hydraulic System - Component Location

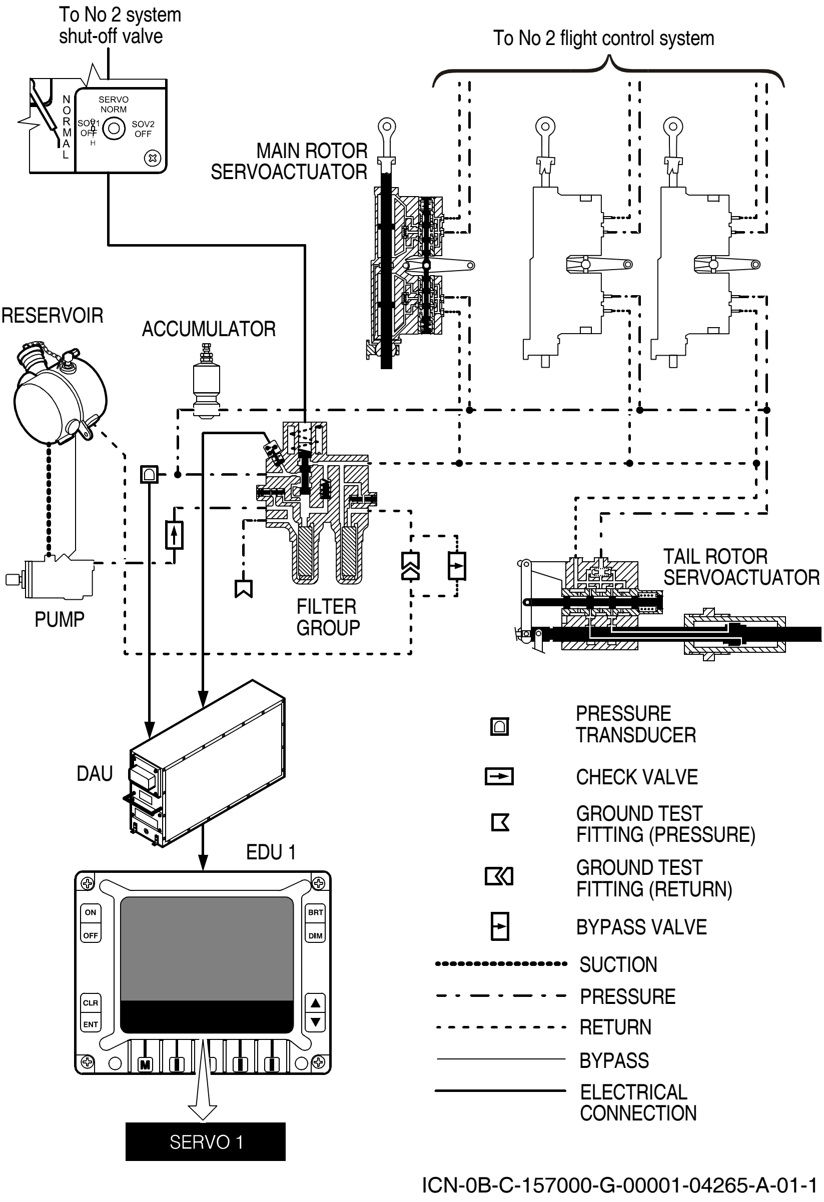

Figure 7-69 No 1 Main Hydraulic System - Schematic Diagram

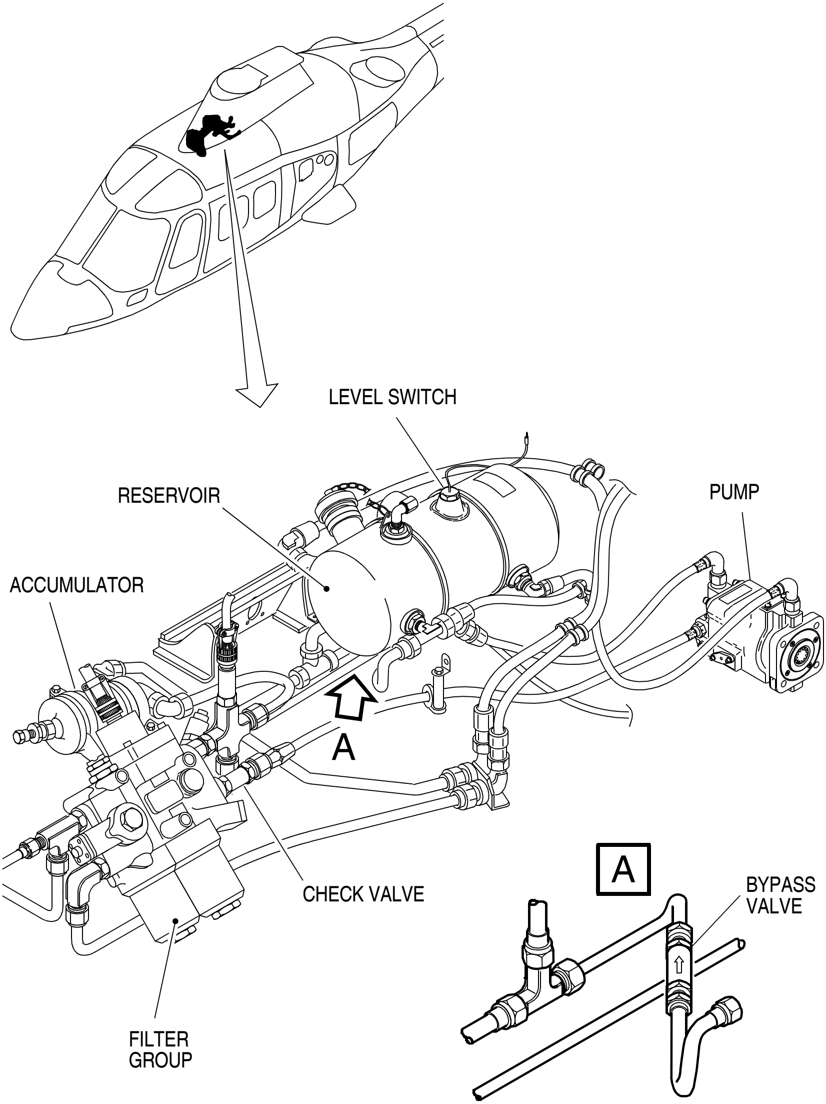

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Figure 7-70 No 2 Main Hydraulic System - Component Location

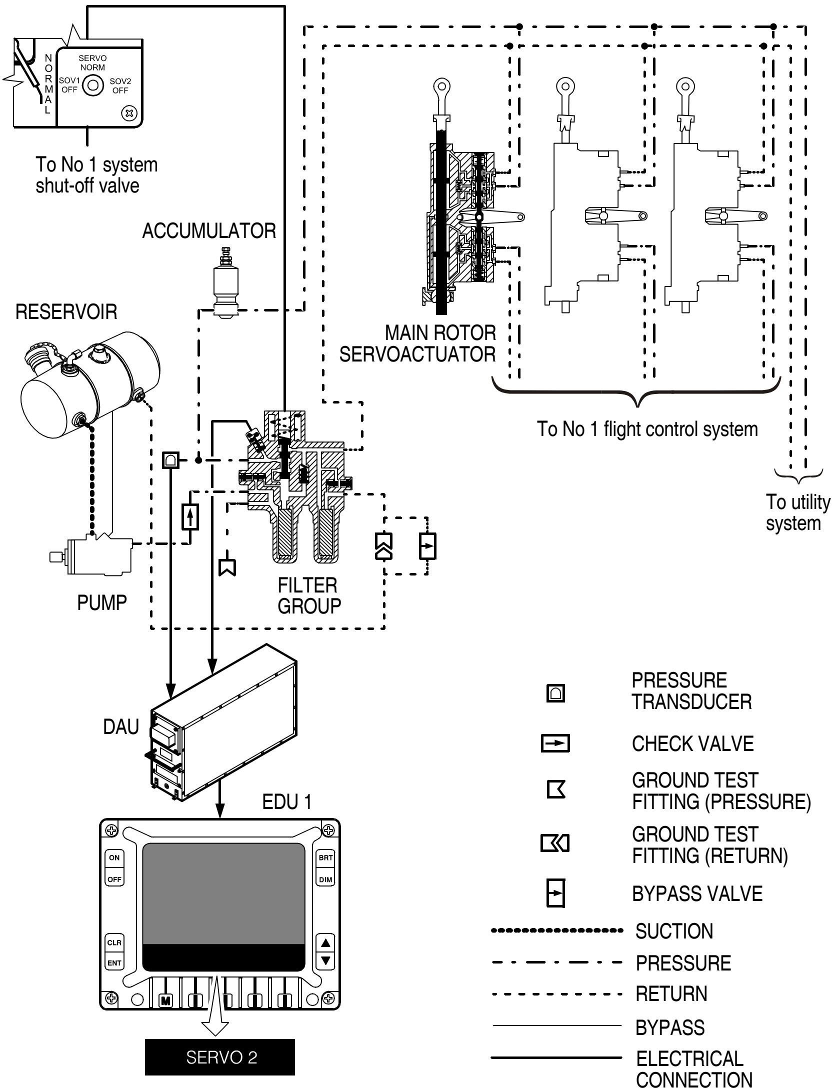

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Figure 7-71 No 2 Main Hydraulic System - Schematic Diagram

The filter group consists of two filters (one in the pressure line and one in the return line), a pressure relief valve, a shut-off valve and a pressure switch.
The shut-off valve operation is controlled by the SERVO switch installed on the landing gear control panel of the front console.
This three position switch, installed on the landing gear control panel, is used to shut off one of the systems.
With both systems operating (SERVO switch set to NORM), the solenoid shut-off valves are de-energized open.
Moving the switch to either OFF position closes the applicable shut-off valve and shuts off the related hydraulic system.
The electrical system is designed so that it is possible to shut-off one of the systems only if the other system is fully operational.
The status of the hydraulic system is displayed on the hydraulic system sub-page of the pilot’s EDU.

The system is powered through the circuit breaker that follows:

– HYD SYS (28 V DC ESSENTIAL BUS #2).

UTILITY HYDRAULIC SYSTEM

The utility hydraulic system includes two sub-systems:

– The main system
– The emergency system.

The utility hydraulic system is supplied by the No 2 main hydraulic system.

The main components of the main hydraulic systems is shown in Figures 7-68.

The schematic diagram of the utility hydraulic system is shown in Figure 7-73.

The main system supplies hydraulic power for operation of the landing gear (through the landing gear control panel), the wheel brakes (through the brake pedals installed on the pilot's rudder pedals), the rotor brake (through the rotor brake control lever) and the nose wheel centering-lock (through the nose wheel lock control lever on the front console).
The emergency system supplies hydraulic power for operation of the landing gear, the nose wheel-centering lock and the wheel brakes.

The main system accumulator is pressurized when the relevant solenoid valve is energized open.
The signal to energize the solenoid valve is supplied by the level switch located in the No 2 system reservoir, only when the fluid level is in a given range. The solenoid valve closes when the accumulator is fully charged.
The emergency system accumulator is pressurized when the relevant solenoid valve is energized open.
The signal to energize the solenoid valve is supplied by the level switch, located in the No 2 system reservoir, only when the fluid level is in a given range, and by the microswitch located on the left and right main landing gears but only when the helicopter is on the ground with Weight On Wheels (WOW).
The solenoid valve closes with a set time delay when the accumulator is fully charged.

The system is powered through the circuit breaker that follows:

– HYD UITL CONTR (28 V DC MAIN BUS #2).

INDICATING SYSTEM

The status of the hydraulic system is displayed on the EDUs (refer to Figure 7-67).

Low pressure indications are provided for each hydraulic system. They consist of caution messages displayed on the pilot's EDU.
Each system is provided with a pressure switch installed in its filter group.
The switches feed electrical signals to the DAU, and the related information is shown on the EDU display.

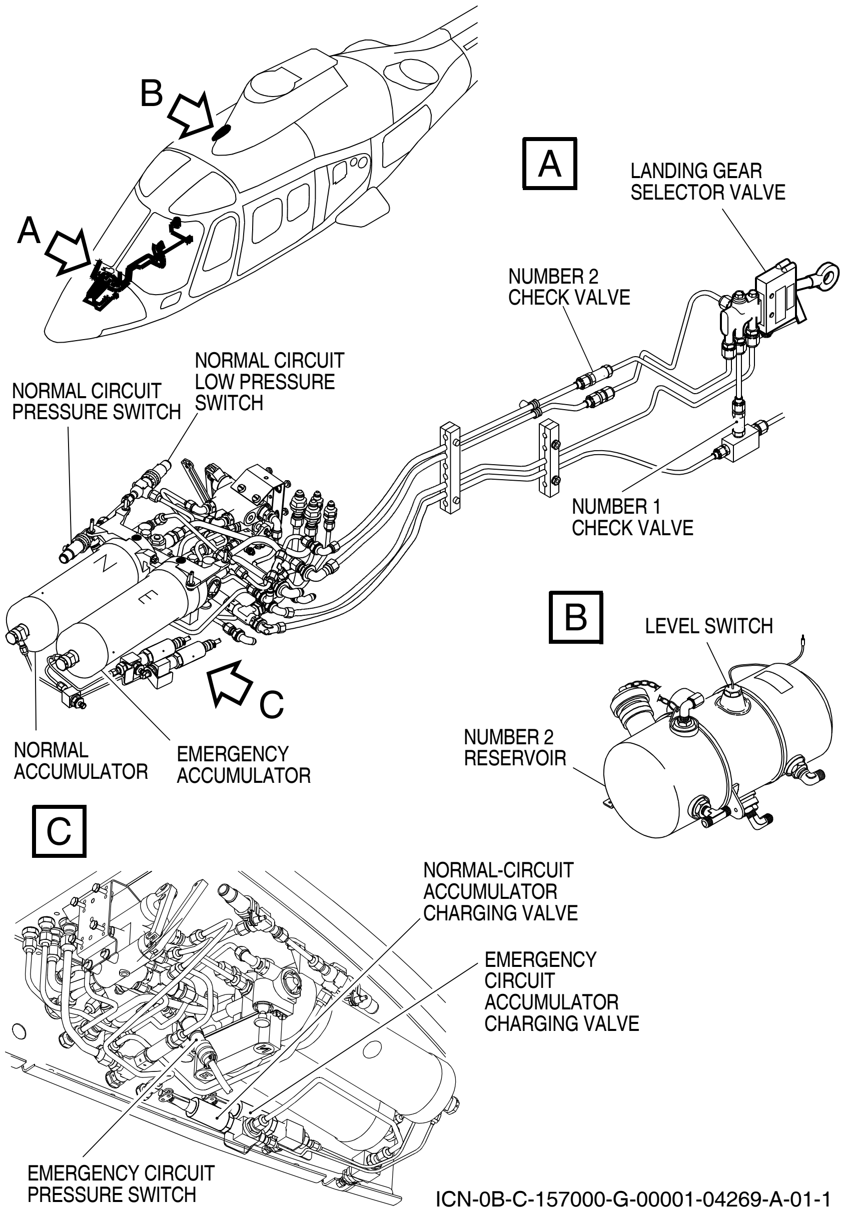

Figure 7-72 Utility Hydraulic System - Component Location

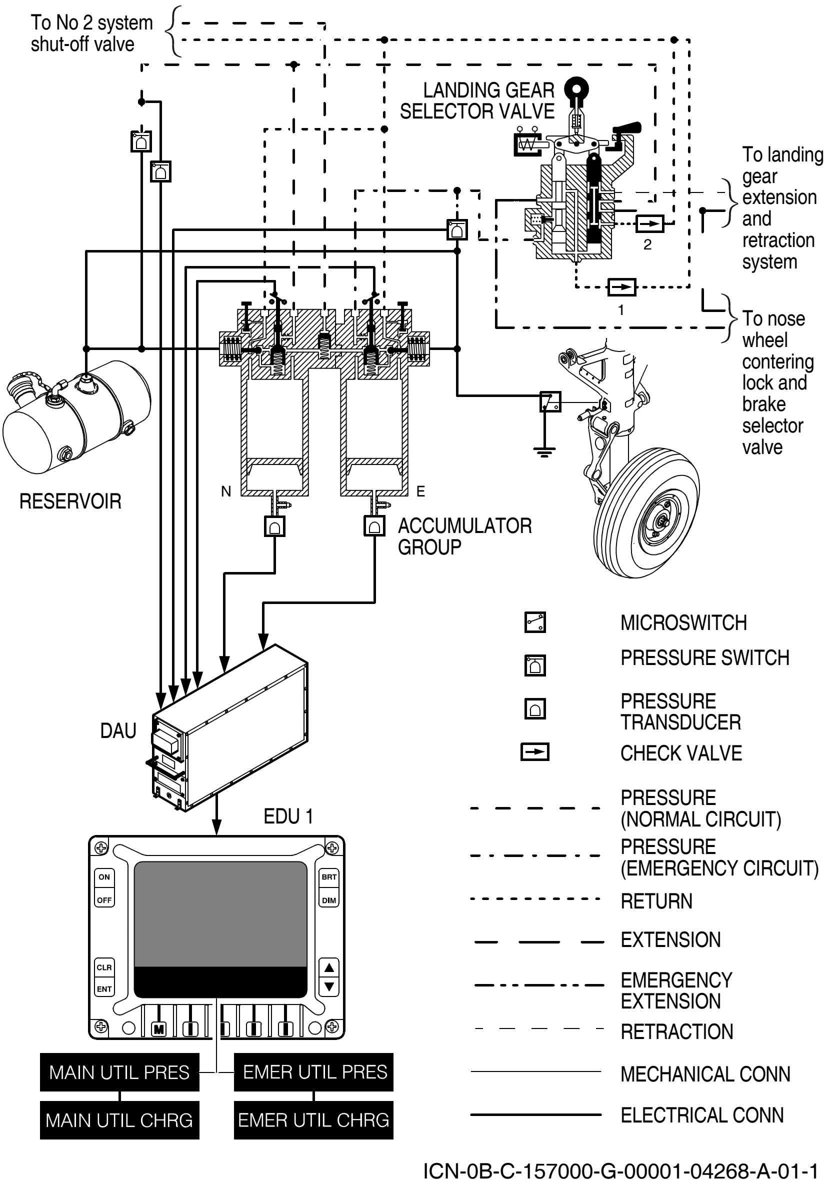

Figure 7-73 Utility Hydraulic System - Schematic Diagram

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ICE AND RAIN PROTECTION

(Chapter 30)

GENERAL

The ice and rain protection system includes:

– The Pitot tube heating system
– The windshield wiping system.
– The windshield washing system (optional).

ICE AND RAIN PROTECTION CONTROLS AND DISPLAYS

Refer to Figure 7-74.

PITOT TUBE HEATING SYSTEM

CAUTION

Operation of the heating elements with an ambient air temperature above 25 °C can damage the Pitot tubes.

This system prevents the build up of ice on the Pitot tubes.
The Pitot tube heating system is controlled through two switches located on the overhead console. Each Pitot tube heating can be individually selected and switched either OFF or ON. These switches also act as circuit breakers to protect the heating elements from overvoltage. Operation of the heating elements is shown on the co-pilot's EDU by two advisory messages (PITOT 1 HEAT and PITOT 2 HEAT).
Refer to Figure 7-75 for the schematic diagram of the Pitot tube heating system.

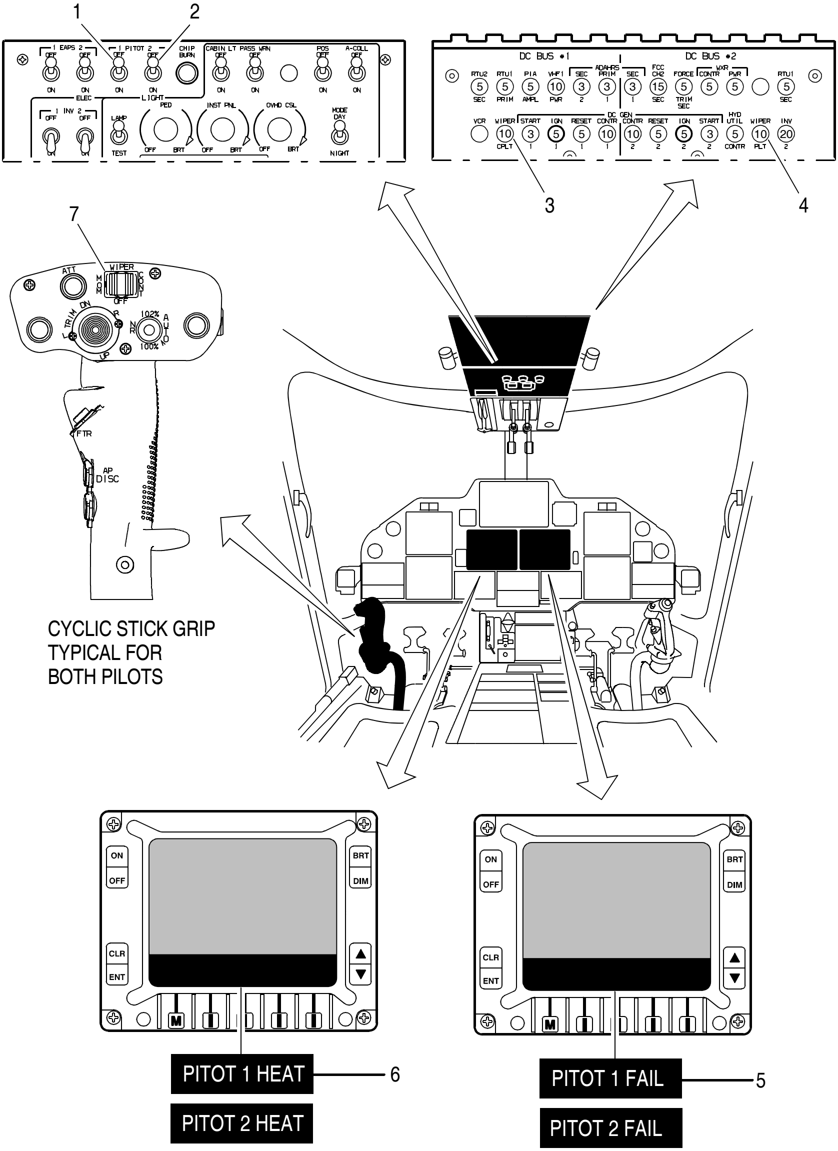

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Figure 7-74 Ice and Rain Protection - Controls and Displays

Key to Figure 7-74

Ref.	Control/Display	Function
1	PITOT 1 switch	OFF position - the heating element of the No 1 Pitot tube is de-energized. ON position - the heating element of the No 1 Pitot tube is energized.
2	PITOT 2 switch	OFF position - the heating element of the No 2 Pitot tube is de-energized. ON position - the heating element of the No 2 Pitot tube is energized.
3	WIPER CPLT switch	OFF position - the co-pilot's windshield wiping circuit is de-energized. ON position - the co-pilot's windshield wiping circuit is energized. Operation of the wiper blade is possible with the wiper selector.
4	WIPER PLT switch	OFF position - the pilot's windshield wiping circuit is de-energized. ON position - the pilot's windshield wiping circuit is energized. Operation of the wiper blade is possible through the wiper selector.
5	PITOT 1 (2) FAIL caution message	Comes in view on the pilot EDU when the related Pitot tube is not energized with the Pitot switch at ON.
6	PITOT 1 (2) HEAT advisory message	Comes in view on the co-pilot EDU when the related PITOT switch is at ON.
7	Wiper selector	OFF (middle position) - the pilot's or co-pilot's wiper blade is at rest. LOW (left position) - the pilot's or co-pilot's wiper blade operates at low speed. HIGH (right position) - the pilot's or co-pilot's wiper blade operates at high speed.

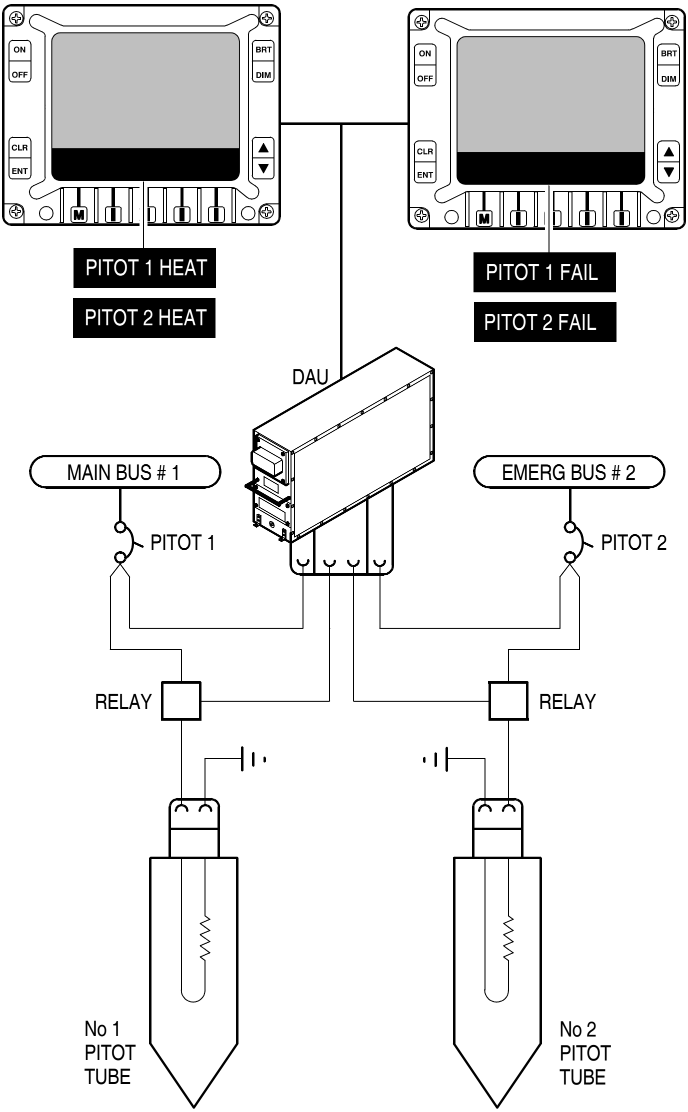

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Figure 7-75 Pitot Tubes Heating System - Schematic Diagram

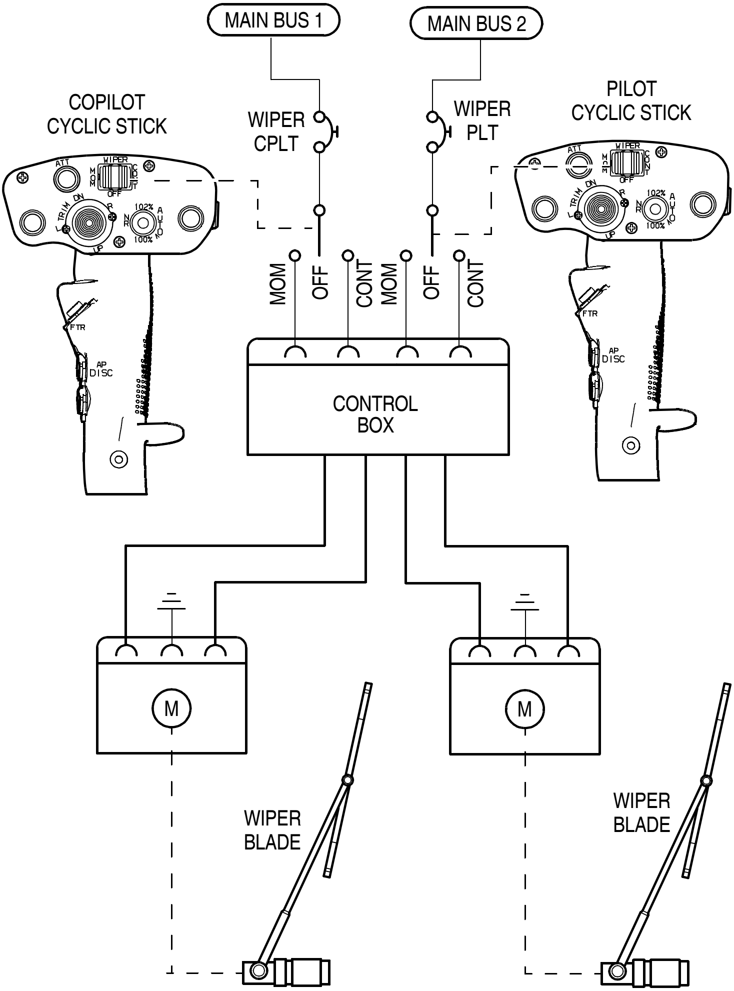

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Figure 7-76 Windshield Wiping System - Electrical Diagram

WINDSHIELD WIPING SYSTEM

The windshield wiping system consists of two identical installations, one for the pilot's windshield and the other for the co-pilot's windshield.
Each installation consists of a wiper blade actuated through an arm by an electric motor-converter. A selector switch, on the cyclic stick grip permits the windshield wipers to be operated at low or high speed.

CAUTION

Operation of wiper blades on a dry windshield may result in scratching of the windshield.

The system is powered through the toggle-type circuit breakers that follow:

– WIPER CPLT (MAIN BUS #1)
– WIPER PLT (MAIN BUS #2).

Refer to Figure 7-76 for the windshield wiping system electrical diagram.

WINDSHIELD WASHING SYSTEM

The windshield washing system is composed of a pilot and co-pilot washer pump and washer bottle. The washer pump is linked to nozzles placed onto the wipers. The washer bottle and the washer pump are installed close to the lower window.

Each windshield washer has a dedicated WSHLD WASHER control push-button located on the cyclic grip that energizes the washer pump to the relevant side.

The system is powered through the circuit breakers that follow:

– WIPER CPLT (MAIN BUS #1)
– WIPER PLT (MAIN BUS #2).

Refer to Figure 7-76A for windshield washing system component location.

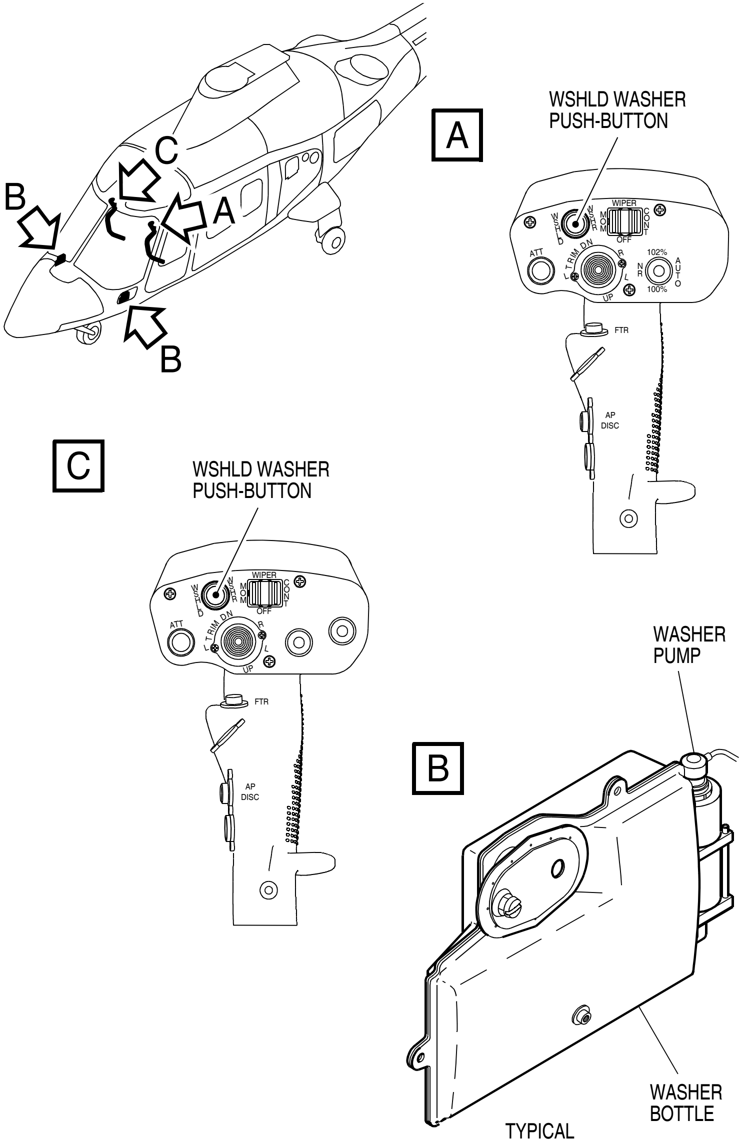

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Figure 7-76A Windshield Washing System - Component Location

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INDICATING/RECORDING

(Chapter 31)

GENERAL

The indicating/recording system includes:

– The instrument panel and consoles installation
– The independent instruments
– The central warning and caution system.

INSTRUMENT PANEL AND CONSOLES INSTALLATION

This installation accommodates all instruments and control panels of the helicopter, and consists of:

– The instrument panel
– The overhead console
– The front console
– The interseat console.

The layout of the each panel and console is shown in Figure 7-4 thru Figure 7-8.

INDEPENDENT INSTRUMENTS

The independent instruments comprise two digital chronometers and two magnetic compasses.

A digital chronometer is located on each side of the instrument panel. The chronometer displays time, date and elapsed time onto a liquid crystal display. Settings are controlled via three push-buttons on the front of the chronometer. Time can be displayed in 12 or 24 hour formats. The display shows hours and minutes or months and days in clock mode and minutes and seconds or hours and minutes in the timer mode.
The chronometer has the following modes and options:

– MODE. Selects between ‘clock’ or ‘timer’ mode.
– RESET. Pressed once when in ‘clock’ mode allows months, days, hours and minutes to be changed. Pressed once in the ‘timer’ mode resets the time to zero.
– START-STOP. Starts and stops the timing clock when in the TIMER mode.

A standby magnetic compass is located on each side of the instrument panel. The compass indicates the helicopter’s heading with reference to magnetic north. The compass consists of a rotating compass card and two magnetic bars attached to the compass card as a compensation system. The compass card and magnetic bars are contained in a sealed case filled with damping fluid. The compass card is graduated in 5-degree intervals and shows letters and numerals every 30 degrees. The heading is read against a lubber line through a transparent window. The compensation system permits error correction by changing the position of the magnetic bars via a screw on the outside of the instrument.

Refer to Figure 7-77 for the location and controls of the independent instruments.

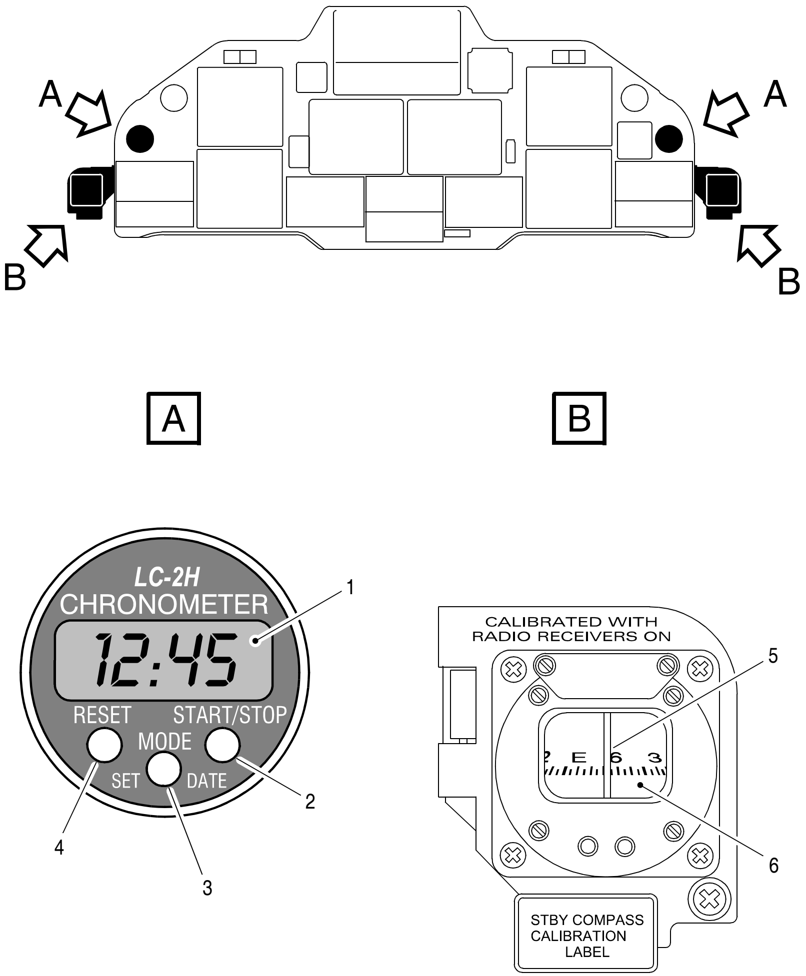

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Figure 7-77 Independent Instruments - Location and Controls

Key to Figure 7-77

Ref.	Control/Display	Function
1	Display	In clock mode, shows hours and minutes or month and day. In timer mode, shows minutes and seconds or hours and minutes.
2	START/STOP push-button	– starts timer count when timer is selected. – stops timer count when timer is selected.
3	MODE push-button	Pressed momentarily - selects the clock or timer mode.
4	RESET push-button	SET function: – Pressed once when the clock mode is selected, allows months, days, hours and minutes to be changed. – Pressed again at the end of time or date adjustment, keeps the changed time or date. RST function - pressed once when the timer mode is selected, resets the timer to zero.
5	Lubber line	Allows the reading of the helicopter magnetic heading on the compass card.
6	Compass card	Indicates the helicopter heading with respect to the magnetic North Pole.

INTEGRATED DISPLAY SYSTEM (IDS)

The Integrated Display System (IDS) monitors the engines and airborne systems and provides the pilots with all the corresponding parameters and values and warning, caution, advisory and status messages.
The system mainly consists of a Data Acquisition Unit (DAU) and two Electronic Display Units.

The system interfaces with these equipment:

– Engine Control Units (ECU1 and ECU2)
– Fuel Computer Units (FCU1 and FCU2)
– Engine fire detectors
– Automatic Flight Control System (AFCS)
– Aural Warning Generator (AWG)
– Master Warning Lights (MWL)
– Master Caution Lights (MCL)

The data, received and processed by the DAU, are shown on the EDUs. In case of IDS failure or when the system functions are degraded, the IDS caution message is shown on the EDU1.

Refer to Figure 7-78 for the block diagram of the Integrated Display System.

The system is powered through the circuit breakers that follow:

– DAU CH-A (28 V DC BATT BUS)
– DAU CH-B (28 V DC ESS BUS #1)
– EDU PRIM (28 V DC BATT BUS)
– EDU SEC (28 V DC ESS BUS #1).

DATA ACQUISITION UNIT (DAU)

The DAU is installed in the forward avionic compartment on a proper support. The DAU processes all the data received from the helicopter sensors and shows the results on the Electronic Display Units (EDU1 and EDU2). The DAU power, interfaces and processing circuits are redundant to reduce the risk of loss of the functions to a minimum. For this reason the DAU has two independent processing channels, named channel A and channel B. Each channel has a full capability of

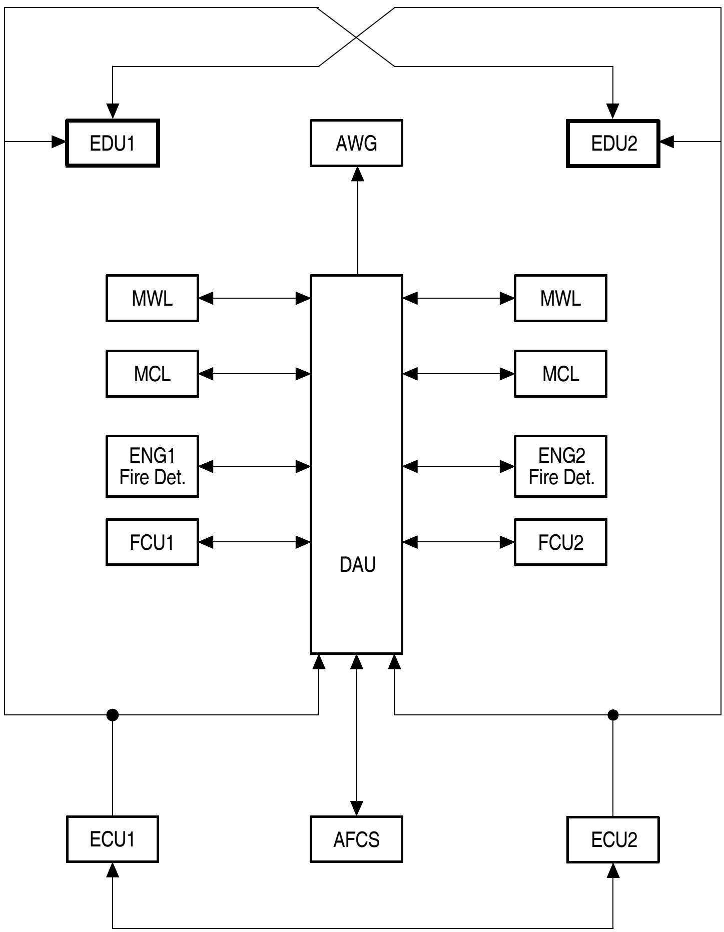

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Figure 7-78 Integrated Display System - Block Diagram

processing the data received from the sensors and interfaces with both EDUs. The channel A is the usual active channel, while the channel B remains in standby mode.
When the channel A has a malfunction, the channel B automatically becomes the active channel without loosing the data visualization.

The DAU performs the following functions:

- Monitors the information provided by the ECUs and by all the analog and discrete sensors and transmits it to both EDUs.
- If any ECU data is declared invalid, automatically makes available the information provided by the corresponding engine analog sensor or by a specific algorithm for the visualization.
- Activates the MASTER WARNING or MASTER CAUTION indicators in the event of any warning or caution condition, respectively.
- Activates the AWG to generate and send the precursor tones or the aural messages related to the detected warning and/or caution conditions.
- Interfaces its two channels with the two channels of the AFCS through ARINC 429 busses for the mutual exchange of information.

When the primary data provided by one or both channels are invalid or the DAU functions are degrading, the DAU MISCMP-P caution message is shown on the EDU1.
The main connector of the DAU includes an RS-232C port which permits the connection to a ground station for maintenance and data loading/unloading operations.

ELECTRONIC DISPLAY UNIT (EDU)

The EDUs are installed on the instrument panel and are identified as Primary EDU (EDU1), on the pilot side, and Secondary EDU (EDU2), on the co-pilot side. During normal operation both displays are operative.

The front panel of each EDU has a color Liquid Crystal Display (LCD) and a set of controls on the bezel. In the top area of the display all engines and systems information is shown in graphical form. In the bottom area the warning, caution, advisory and status messages are shown. When the push-button identified with the letter M is pushed, the bottom area of the display shows a menu. The menu pages can be

scrolled with the scroll rocker switch. The visualization of a message always prevails on the visualization of the menu pages.

The EDUs present the following information:

EDU1

- N1
- TOT
- TRQ
- NR
- N2
- Caution, warning and advisory messages

EDU2

- Engine 1 oil pressure
- Engine 1 oil temperature
- Transmission oil pressure
- Transmission oil temperature
- OAT
- Engine 2 oil pressure
- Engine 2 oil temperature
- Main hydraulic pressure 1 & 2
- Fuel Quantity
- Fuel Pressure
- Utility hydraulic pressure
- DC current
- DC voltage
- AC voltage
- Fuel flow
- Normal utility hydraulic pressure
- Emergency utility hydraulic pressure
- Advisory and status messages

Note

Fuel flow indication should not be used for flight planning.

The bottom area of each EDU shows, depending on the selected display format, different menu options or up to 9 active messages, as follows:

– WARNING messages (red)

– CAUTION messages (yellow)

– ADVISORY messages (green)

– STATUS messages (cyan).

The warning messages take precedence over the caution and advisory messages, and both the warning and caution messages remain in view until the condition that caused them to be shown has been corrected. The warning, cautions, advisory and status messages are listed in Table 7-1

Table 7-1 Warning, Caution, Advisory and Status Messages

	WARNING Messages	
ENG 1 FIRE	ROTOR HIGH	ENG 2 FIRE
ENG 1 OUT	ROTOR LOW	ENG 2 OUT
#1 ECU FAIL	XMSN OIL PRES	#2 ECU FAIL
#1 OIL PRES	XMSN OIL HOT	#2 OIL PRES
	BATT HOT	
	BATT DISCH	
	ELECTRICAL	
	ROTOR BRK ON	
	CAUTION Messages	
#1 OIL HOT	XMSN OVTRQ	#2 OIL HOT
#1 OIL CHIP	TGB OIL CHIP	#2 OIL CHIP
#1 DCU	XMSN OIL CHIP	#2 DCU
#1 ECU MAINT	TRQ LIMITER	#2 ECU MAINT
#1 PMS	RPM SELECT	#2 PMS
#1 PLA	RMS MESSAGES	#2 PLA
#1 FIRE DET	FLOATS ARMED	#2 FIRE DET
#1 FUEL LOW	HOOK ARMED	#2 FUEL LOW
#1 F LOW FAIL	HOOK OPEN	#2 F LOW FAIL
#1 FUEL FLTR	LANDING GEAR	#2 FUEL FLTR
#1 A/F F FLTR	AWG FAIL	#2 A/F F FLTR
FUEL PUMP 1	CABIN DOOR	FUEL PUMP 2
FUEL DRAIN 1	IDS	FUEL DRAIN 2
#1 DC GEN	DAU MISCMP-P	#2 DC GEN
#1 GEN CTL	BUS TIE	#2 GEN CTL
INV 1	EXT PWR ON	INV 2
SERVO 1	ROTOR BRK	SERVO 2
MAIN UTIL PRES	PARK BRK ON	EMER UTIL PRES
MAIN UTIL CHRG	BATT OFF	EMER UTIL CHRG
ADC 1		ADC 2
AHRS 1		AHRS 2
#1 HOT START		#2 HOT START
		BAG DOOR

Table 7-1 Warning, Caution, Advisory and Status Messages

Caution Messages (Continued)		
#1 OVSPD	MGT MATCH	#2 OVSPD
#1 OVSPD DET	MAIN RTR SERVO	#2 OVSPD DET
#1 OVSPD TEST		#2 OVSPD TEST
#1 ECU DATA	REAR BAY FAN	#2 ECU DATA
#1 TOT LIMITER		#2 TOT LIMITER
#1 PLA MOTOR		#2 PLA MOTOR
PITOT 1 FAIL		PITOT 2 FAIL
FIRE BTL 1		FIRE BTL 2
1 AP OFF	ATT OFF	2 AP OFF
1 AP FAIL	AP CAS FAIL	2 AP FAIL
1 AP HOT	AP DGRD	2 AP HOT
AP AHRS 1 FAIL	APMS PNL FAIL	AP AHRS 2 FAIL
1 AP P FAIL	MISTRIM	2 AP P FAIL
1 AP R FAIL	P TRIM FAIL	2 AP R FAIL
1 AP Y FAIL	R TRIM FAIL	2 AP Y FAIL
1 AP TEST FAIL	Y TRIM FAIL	2 AP TEST FAIL
	C TRIM FAIL	
ADVISORY Messages		
PITOT 1 HEAT	LANDING LT ON	PITOT 2 HEAT
	C/Y TRIM OFF	VENT ON
	P/R TRIM OFF	LIMITER ON
STATUS Messages		
CHECK SETUP	EXCEEDANCE	MAINT LOG A
CHECK DATA	MAINTENANCE	MAINT LOG B

After the initialization process is completed, the EDUs can access different functions/modes according to the operational conditions or crew requests as follows:

– START. This key selects the START mode on the EDU1.
– CRUISE. This key selects the CRUISE mode on the EDU1.
– MAIN. This key selects the MAIN mode on the EDU2.

- AUX. This key selects the AUX (auxiliary) mode (i.e. hydraulic, electrical and fuel information) on the EDU2.
- TEST. This key initiates the BIT of the IDS system. It can be operated only when the helicopter is on the ground (WOW active).
- POWER CHECK. This key initiates the Power Assurance Check recording.
- EXIT. This key permits the exit from the menu selection and the return to the primary mode of operation.
- DAU (CH-A). This key permits the manual selection of the DAU Channel A as primary source. At initial power-up, the legend CH-A is displayed in green to indicate that this is the active channel. Once selected, a white box surrounds the green legend. The channel failure is indicated in yellow.
- DAU (CH-B) - This key permits the manual selection of the DAU Channel B as primary source. At initial power-up, the legend CH-B is displayed in white to indicate that this is the hot-standby channel. Once selected, a white box surrounds the green legend. The channel failure is indicated in yellow.
- DATA - This key gives access to the SENSORS DATA page containing status and information about each strain-gage sensor and collective pitch position sensor interfaced with the DAU. It can be operated only when the helicopter is on the ground (WOW active).This page permits the calibration (CAL) sequence of all strain-gage sensors to be initiated.
- MAIN TEST - This key performs an extensive EDU display test, and gives access to the IDS STATUS page. It can be operated only when the helicopter is on the ground (WOW active).
- RTN - This key permits return to the previous mode of operation.

When the Maintenance (MAINT) menu is accessed, the following modes are available:

- FAULT. This key gives access to the Fault Log.
- CUM. This key gives access to the Cumulative Log.
- EXCEED. This key gives access to the Exceedance Log.
- PWR. This key gives access to the Power Assurance Log.

Refer to Figure 7-79 for the EDU1 and EDU2 menu tree.

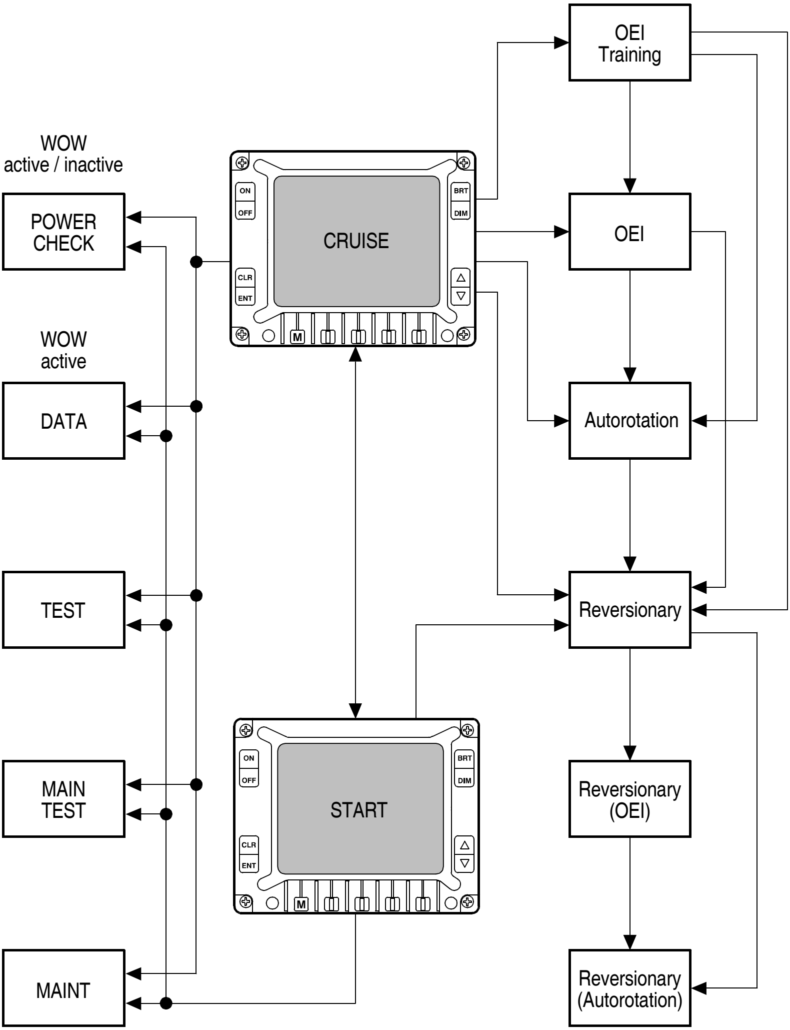

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Figure 7-79 Integrated Display System - EDU Mode Tree (Sheet 1 of 3)

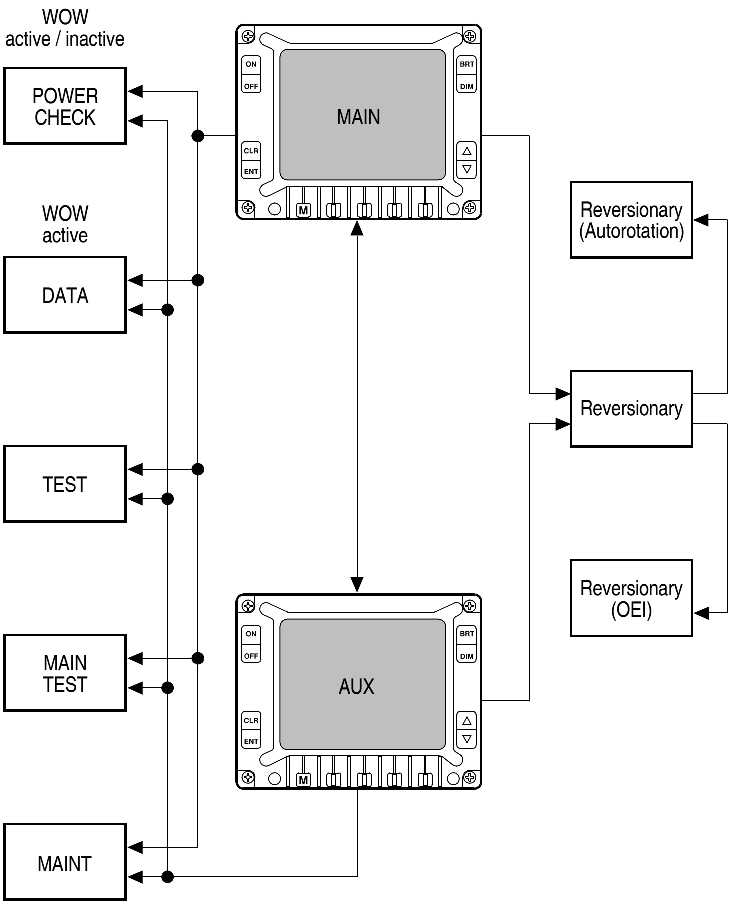

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Figure 7-79 Integrated Display System - EDU Mode Tree (Sheet 2 of 3)

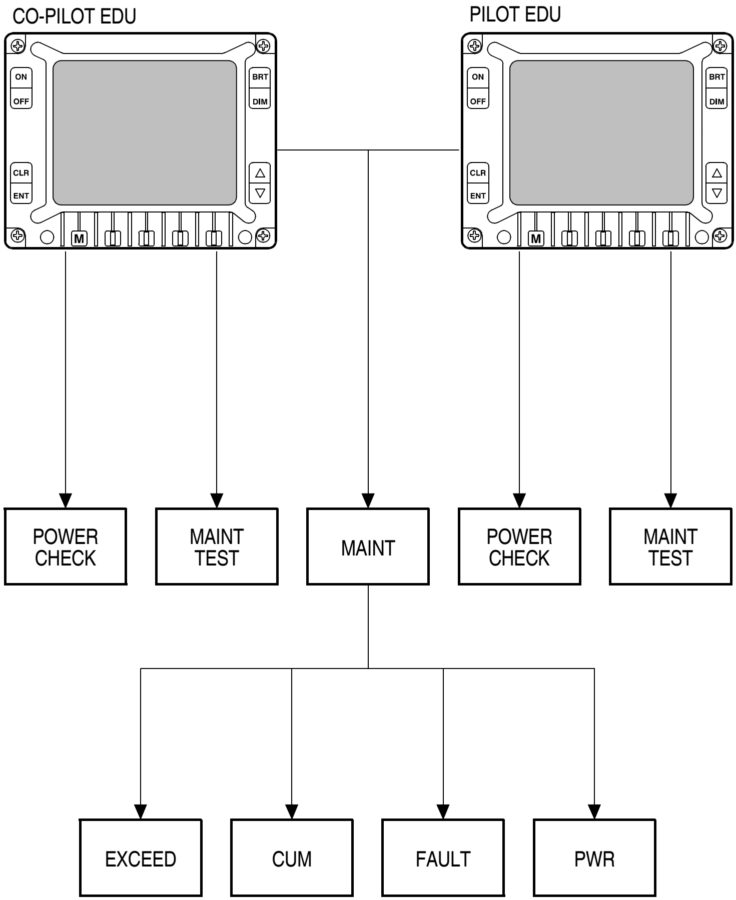

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Figure 7-79 Integrated Display System - EDU Mode Tree (Sheet 3 of 3)

The START mode of EDU1 is displayed after the initialization process or when selected by the START key on EDU1.

The CRUISE mode is displayed on EDU1 when selected by the CRUISE key on EDU1 or automatically when NR exceeds 85% at the end of the start sequence. If the DAU fails, both EDUs will display the CRUISE mode as secondary data will no longer be available. The cruise (AEO) format is also presented in the REVERSIONARY mode.

The OEI mode is a subset of the CRUISE mode and is automatically displayed if one engine becomes inoperative. The OEI format is also presented in REVERSIONARY mode.

The AUTOROTATION mode is automatically displayed when:

– both engines fail
– both engine mode switches are set to IDLE or OFF
– both engine torque inputs are less than 2%
– in presence of a NR/N2 split.

The AUTOROTATION format is also presented in REVERSIONARY mode.

For the Category A operations only, the OEI TRAINING mode can be selected by the pilot at any time. This mode is automatically reverted to the CRUISE mode (either OEI or AEO, as applicable) if a single or dual engine failure condition occurs or if the ECU suppresses the training command.

Refer to Figure 7-80 for the EDU1 and EDU2 formats.

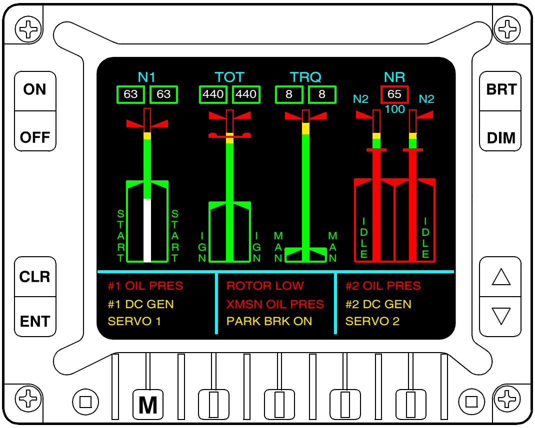
The MAIN mode of EDU2 is displayed after the initialization process or when selected by the MAIN key on EDU2 and/or when any warning or caution message appears.

The AUXILIARY mode of EDU2 is displayed by pressing the AUX key on EDU2. The AUXILIARY page reverts automatically to the MAIN page when any warning or caution message appears.

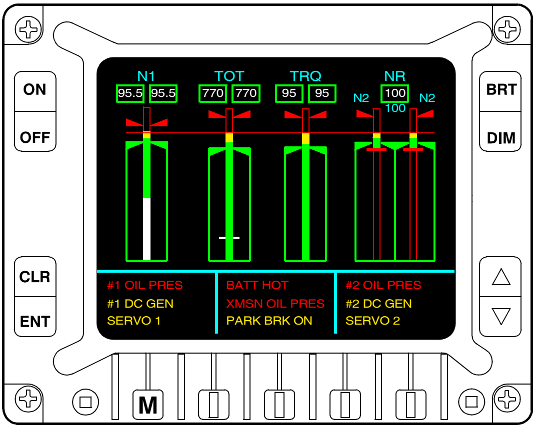
An EDU will automatically revert to the REVERSIONARY mode if the other EDU fails or is switched OFF manually using the ON/OFF switch.

CONTROLS AND DISPLAYS OF THE INTEGRATED DISPLAY SYSYTEM

Refer to Figure 7-81.



EDU1 - START Mode (AEO) (Typical)



EDU1 - CRUISE Mode (AEO) (Typical)

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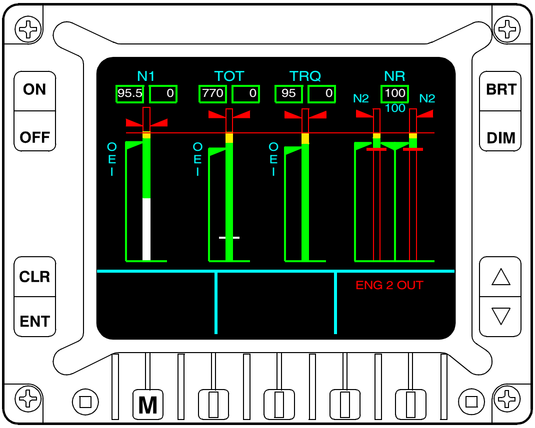
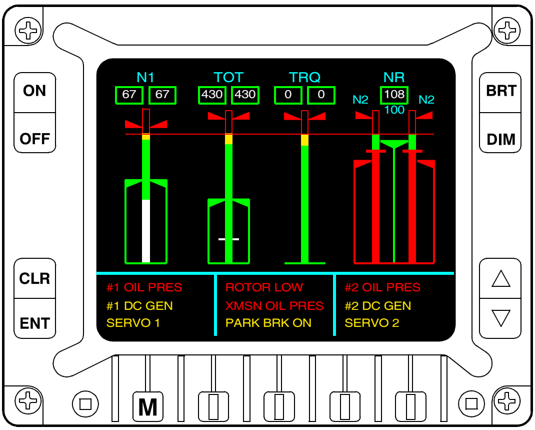
Figure 7-80 EDU1 and EDU2 Formats
(Sheet 1 of 4)



EDU1 - OEI Mode (Typical)



EDU1 - AUTOROTATION Mode (Typical)

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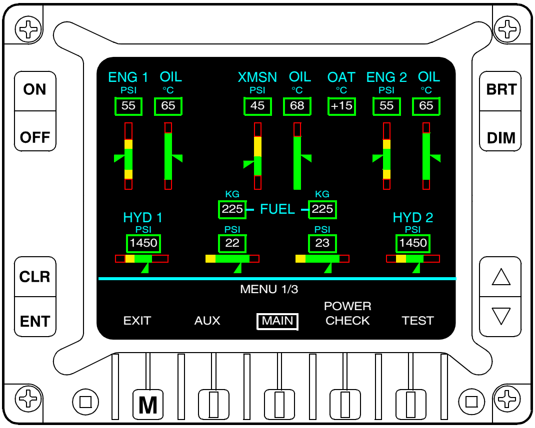
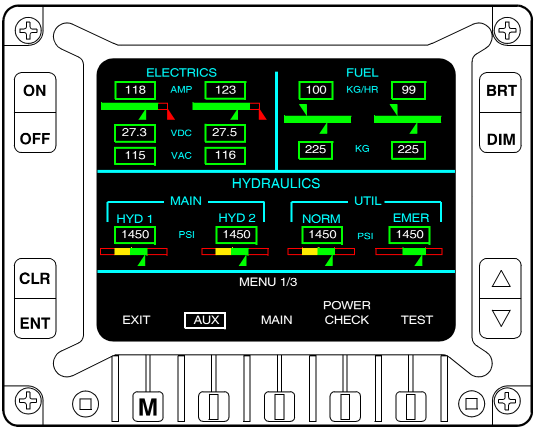
Figure 7-80 EDU1 and EDU2 Formats (Sheet 2 of 4)



EDU2 - MAIN Mode (Typical)



EDU2 - AUXILIARY Mode (Typical)

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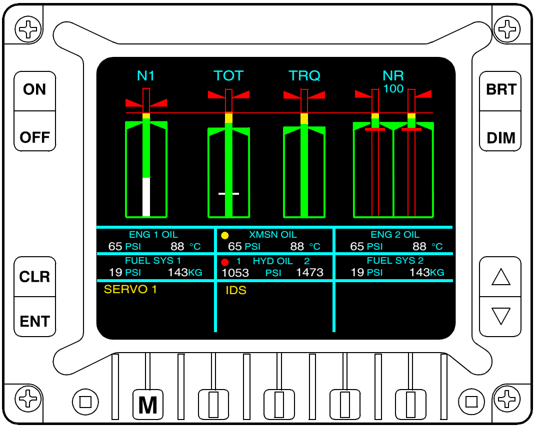
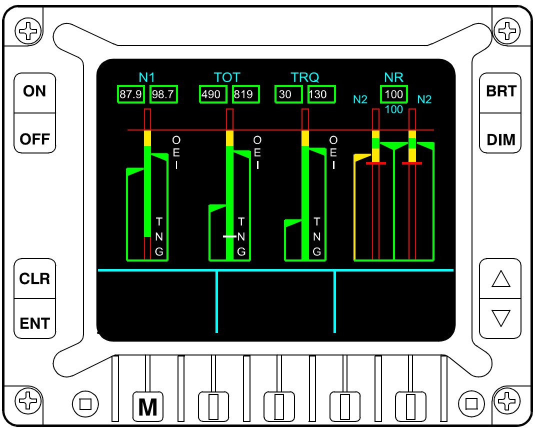
Figure 7-80 EDU1 and EDU2 Formats (Sheet 3 of 4)



EDU1 and EDU2 - REVERSIONARY Mode (Typical)



EDU1 - OEI TRAINING Mode

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Figure 7-80 EDU1 and EDU2 Formats (Sheet 4 of 4)

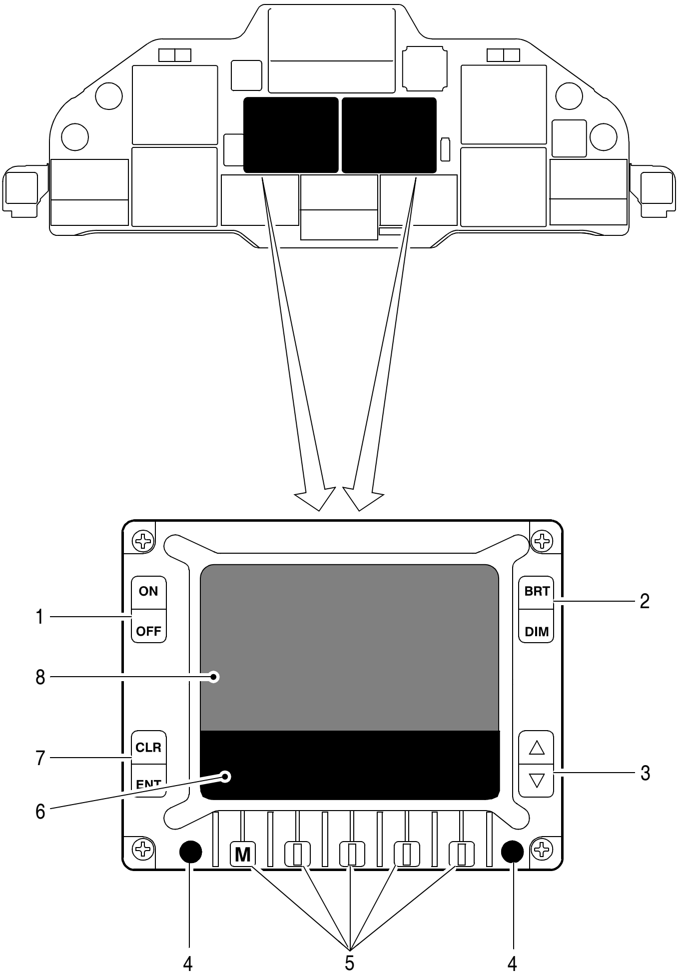

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Figure 7-81 Integrated Display System - Controls and Displays

Key to Figure 7-81

Ref.	Control/Display	Function
1	ON - OFF rocker switch	OFF - the EDU is inoperative. When the switch of one EDU is set to OFF, the reversionary mode is commanded on the other EDU. ON - the EDU is operative.
2	DIM - BRT rocker switch	BRT - adjusts the display luminance to the maximum. DIM - the display luminance is dimmed.
3	Scroll rocker switch (vertical arrows)	Permits caution, warning and advisory messages or new options to be scrolled. The up and down arrows are used respectively to scroll forward and backward one page at a time through the multiple pages of the same mode.
4	Light sensor	Automatically adjusts the display luminance according to the ambient light.
5	Soft keys (momentary push-button switches)	Permit menu selection or recall. One key is engraved with a “M” to identify the Menu/ Mode recall function. When this key is pressed, the corresponding display format is recalled on the EDU1 or EDU2. The remaining keys are used in conjunction with the information being shown on the EDU display and their functions are defined by the system software. According to the selected display formats, these keys are used either to select an option or to control a cursor that is used for data entry.
6	Lower display area	Warning, caution, advisory messages and status information are shown in this area.
7	ENT CLR switch	Permits the status messages to be cleared. When CLR is pressed, all the messages are reordered according to their priority.
8	Upper display area	Analog and digital information is shown in this area.

ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

The Electronic Flight Instrument System (EFIS) provides the pilots with the flight and navigation data. The system mainly consists of four multifunction displays installed on the instrument panel (Figure 7-82).
Two displays are vertically aligned on the pilot side of the instrument panel. The other two displays are vertically aligned on the copilot side of the instrument panel.
The upper display on pilot side is referred to as the Pilot Primary Flight Display (PFD2), the lower display as the Pilot Multi-Function Display (MFD2). On co-pilot side, the instruments are named Copilot Primary Flight Display (PFD1) and Copilot Multi-Function Display (MFD1).

In normal condition the upper displays are configured to operate as Electronic Attitude and Director Indicator (EADI), while the lower displays are configured to operate as Electronic Horizontal Situation Indicator (EHSI).

The front of each instrument is provided with multifunction push-buttons arranged on the both sides of the bezel and with two knobs located in the lower part of the bezel.
In order to collect all the data necessary for the flight and the navigation, the EFIS is interfaced with the following systems (Figure 7-83):

– Air Data and Attitude Heading Reference Systems (ADAHRS)
– Automatic Flight Control System (AFCS)
– VOR1/ILS1 and VOR2/ILS2 systems
– ADF system
– DF system
– DME system
– Radio-altimeter system

The data are transmitted through a dual digital data bus and then, adequately analyzed and processed, are shown on the PFDs and MFDs.

The system is powered through the circuit breakers that follow:

– EADI PLT (28 V DC MAIN BUS #1)
– EADI PLT (28 V DC GROUND BUS)

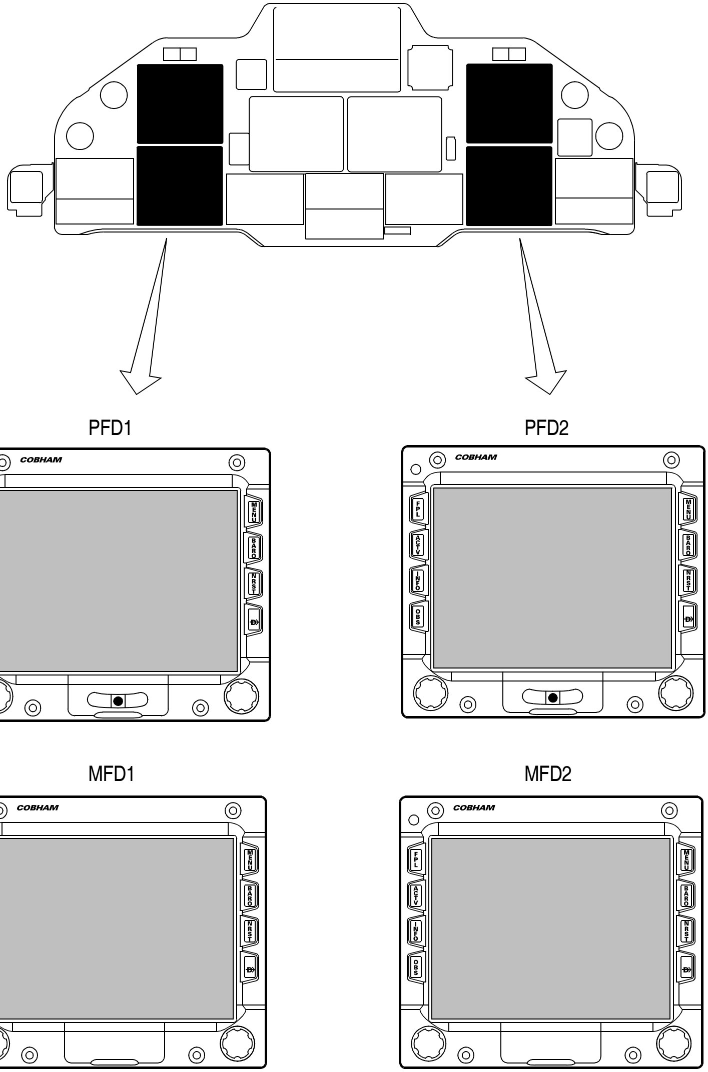

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Figure 7-82 Electronic Flight Instrument System - Components Location

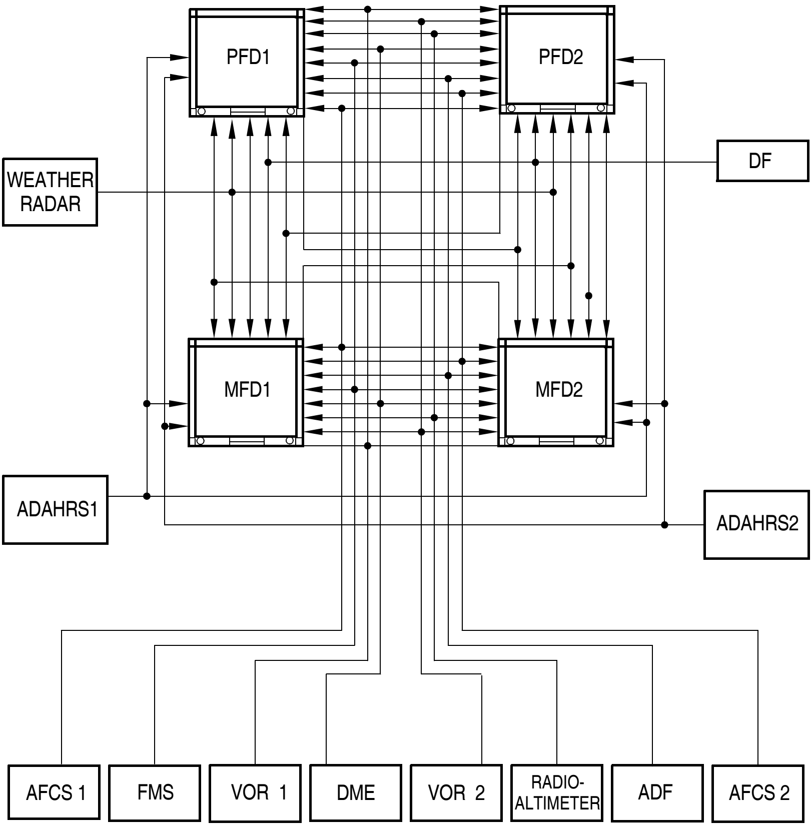

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Figure 7-83 Electronic Flight Instrument System - Block Diagram

– EHSI PLT (28 V DC ESS BUS #1)
– EHSI PLT (28 V DC MAIN BUS #2)
– EADI CPLT (28 V DC MAIN BUS #1)
– EHSI CPLT (28 V DC MAIN BUS #1)
– EHSI CPLT (28 V DC MAIN BUS #2)

OPERATION OF THE EFIS

The data from the different systems are sent to the EFIS, analyzed, ordered in digital format and then shown in graphical and/or alphanumerical form according to the selections made with the controls on each instrument bezel.

The PFD1 and PFD2 function is to show the primary flight information that follows:

– Primary flight data (airspeed, altitude, vertical speed, heading, pitch and roll angles)
– EFIS in command indication
– AFCS datums and annuciations
– Different cues (GS, LOC, VNAV, FD bars, collective cue, cyclic cue and other indications according to the selected mode)
– Decision height selection
– Alert and warnings related to the EFIS functions
– Reconfiguration state
– Caution messages for failures and discrepancies.

The MFD1 and MFD2 operate usually as Electronic Horizontal Situation Indicator, but can operate in other different modes according to the operational conditions or crew requests.
The MFDs show, when in the EHSI mode, a conventional HSI page including heading, bearing and distance indications.

In case of failure, the PFDs and MFDs can be reconfigured automatically. In case of a PFD failure, the on-side MFD automatically reconfigures as a PFD. In case of a MFD failure, the PFD continues to operate normally, since all the PFD information are sufficient to safely continue the flight.

The MFDs provide also TAWS (Terrain Awareness and Warning System) functions. The functions include the Forward Looking Terrain Awareness (FLTA). This is a warning function that uses a terrain database and an obstruction database to alert the pilot to hazardous terrain or obstructions in front of the aircraft.

The TAWS functions give annunciations for conditions that demand pilot awareness. There are three categories of annunciations: warnings, cautions and advisories.

– Warnings are displayed with red flags and an aural annunciation that repeats until the condition goes away or is acknowledged by the pilot.
– Cautions are displayed with yellow flags and a single aural annunciation.
– Advisories are displayed with blue flags and a single aural annunciation.

The warning, caution and advisory messages are listed in Table 7-1A.

Table 7-1A Warning, Caution and Advisory Messages

WARNING Messages		
Flag	**Aural Annunciation**	**Description**
OBSTRUCTION	"Warning, Obstruction"	Obstruction within TAWS FLTA warning envelope.
TERRAIN	"Warning, Terrain"	Terrain cell within TAWS FLTA warning envelope.
CAUTION Messages		
Flag	**Aural Annunciation**	**Description**
NO TAWS	Alert tone	TAWS FLTA function inoperative. Indicates that the aircraft is currently beyond extent of terrain database or a failure condition exists that prevents the TAWS FLTA function from operating.
OBSTRUCTION	"Caution, Obstruction"	Obstruction within TAWS FLTA caution envelope.
TERRAIN	"Caution, Terrain"	Terrain cell within TAWS FLTA caution envelope.
TOO LOW	"Too Low Terrain"	Within the GPWS Mode 3 envelope.
ADVISORY Messages		
Flag	**Aural Annunciation**	**Description**
TAWS INHBT	Chime	TAWS Inhibit. TAWS is inhibited through the activation of TAWS inhibit.
TAWS LOW ALT	Chime	TAWS Low Altitude Mode.

EFIS CONTROLS AND DISPLAYS

The multifunction push-buttons and the knobs on the instrument bezels are used to navigate the different menu and have different functions according to the instrument configuration and the selected format of data visualization.

There are two types of menu functions: top-level menu functions, corresponding to the display push-button hard labels, and soft menu functions annunciated by menu tiles that appear on the screen.

Functions associated with soft menu function tiles take precedence over functions associated with push-buttons. Soft menu function tiles include an indication of further menu levels with a two-dot trailer at the end of the word. Soft menu function tiles appear next to the appropriate push-button or in the lower right corner when use of the knob is appropriate (i.e., selection lists and option lists). Selection lists that are too long to be presented in the space available provide an indication of location within the list. Whenever a soft menu is active, an "EXIT" tile appears adjacent to the top right push-button to provide one touch escape from the soft menu system. Whenever a soft menu level is deeper than the first soft menu level, a "BACK" tile appears adjacent to the top left push-button to provide a method of regressing through the soft menu system by one level.

The top-level menu corresponds to the permanent labelling of the push-buttons. Top-level menus are active any time that no soft menu options appear on screen.

Under certain conditions, soft menu tiles automatically appear at the top level to provide the pilot with single-touch access to needed functions. As these menu tiles may be shown for a significant period of time, they have a reduced form factor.

For the EFIS controls and displays, refer to Figure 7-84.

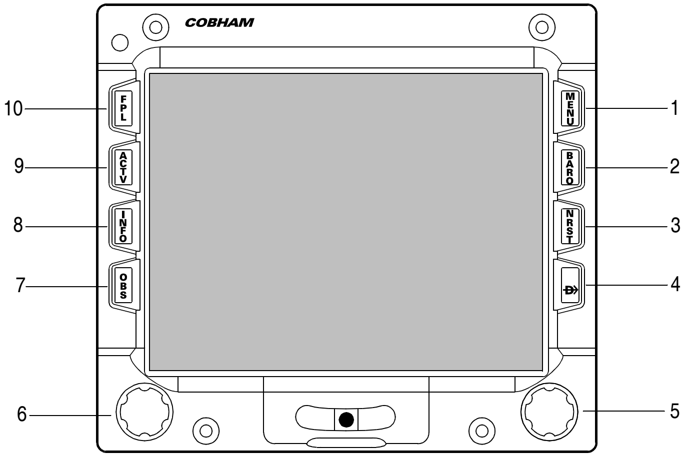

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Figure 7-84 Electronic Flight Instrument System - Controls and Displays

Key to Figure 7-84

Ref.	Control/Display	Function
1	MENU push-button	Activates the first soft menu level associated with the current display screen. The first soft menu level automatically times out after 10 seconds if there are no subsequent pilot actions.
2	BARO push-button	Activates the altimeter menu option.
3	NRST push-button	Activates the nearest menu option.
4	D push-button	Activates the direct menu option.
5	Menu control knob	This knob controls the menu options that appear in the lower right corner of the screen. Its function depends upon screen. Turned, scroll to the desired menu item, letter, or number. Pushed select the item. If there are no menus shown on the MFD screen, turning the control knob sets the scale of the display. Pushed on the MFD brings up a reversionary PFD screen. Pushed again returns to the navigation display.
6	Brightness control knob	Turned clockwise increases the screen brightness, counterclockwise decreases the screen brightness.
7	OBS push-button	Activates the omnibearing selector menu option.
8	INFO push-button	Activates the information menu option.
9	ACTV push-button	Activates the active flight plan menu option.
10	FPL push-button	Activates the flight plan menu option.

CENTRAL WARNING SYSTEM

The central warning system (Figure 7-85) provides the pilots with audio signals to warn that an impending dangerous condition requires an immediate corrective action or a defective condition requires an immediate pilot attention. In the more critical cases the system also indicates that a warning or caution message is shown on the EDU1.

The system comprises an Aural Warning Generator (AWG), the pilot and copilot MASTER CAUTION indicators, the pilot and copilot MASTER WARNING indicators, the MASTER RESET push-button on the pilot and copilot collective stick grips, the AWG switch on the electrical control panel and a horn.

The AWG, installed in the forward avionic compartment, generates tones and synthesized vocal messages that are sent to the flight crew headsets.
For this purpose the AWG is interfaced with several controls and equipment as shown in the schematic diagram of Figure 7-86.

Should two or more messages (maximum five) be activated simultaneously, the messages are played in sequence in the decreasing priority order (Table 7-2).

If more than two messages are activated, the two messages with higher priority are played in alternate mode (once for each message).
If a message is activated when a previous message is being played, the new message is immediately played at the end of the former, if the former has higher priority.
If the new message has a priority higher than the message being played, this message is suspended at the first significant pause and the new and higher priority message is played.

When a message is activated, it is automatically silenced at the end of the play sequence if the related input signal is no longer active. The message can be manually silenced by pushing the MASTER CAUTION indicator or the MASTER RESET push-button on the collective stick grips.

If a failure occurs to the AWG, the AWG FAIL caution message is shown on the EDU1.

The MASTER CAUTION indicators are installed in the left and right upper corners of the instrument panel.

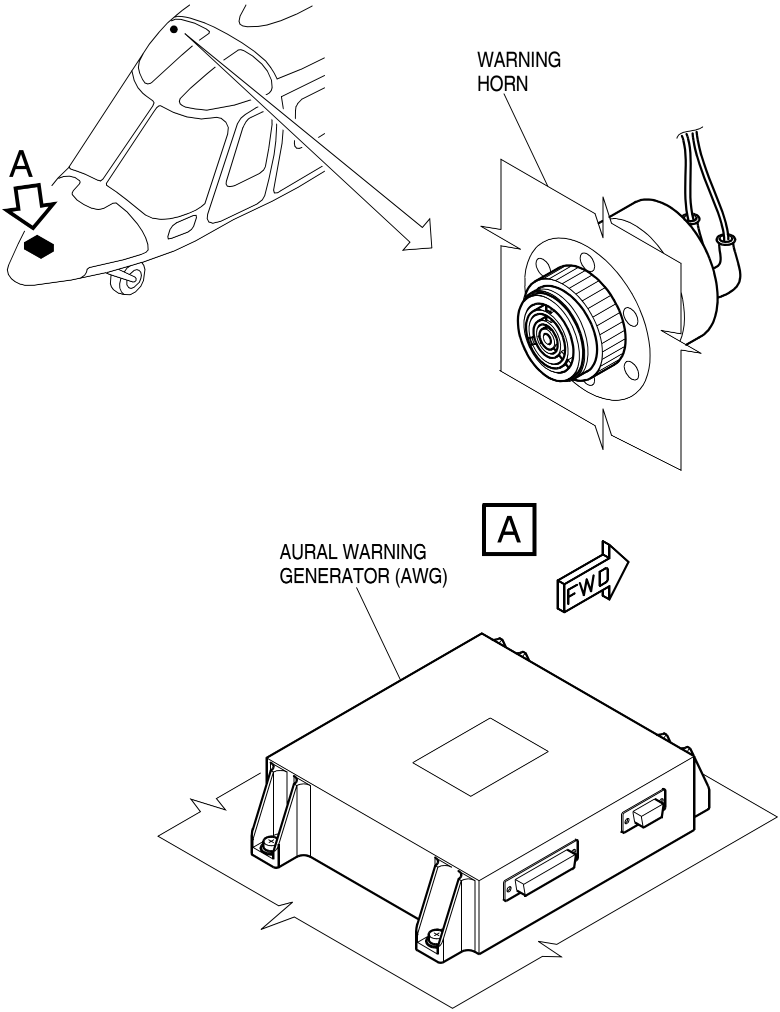

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Figure 7-85 Central Warning System - Components Location

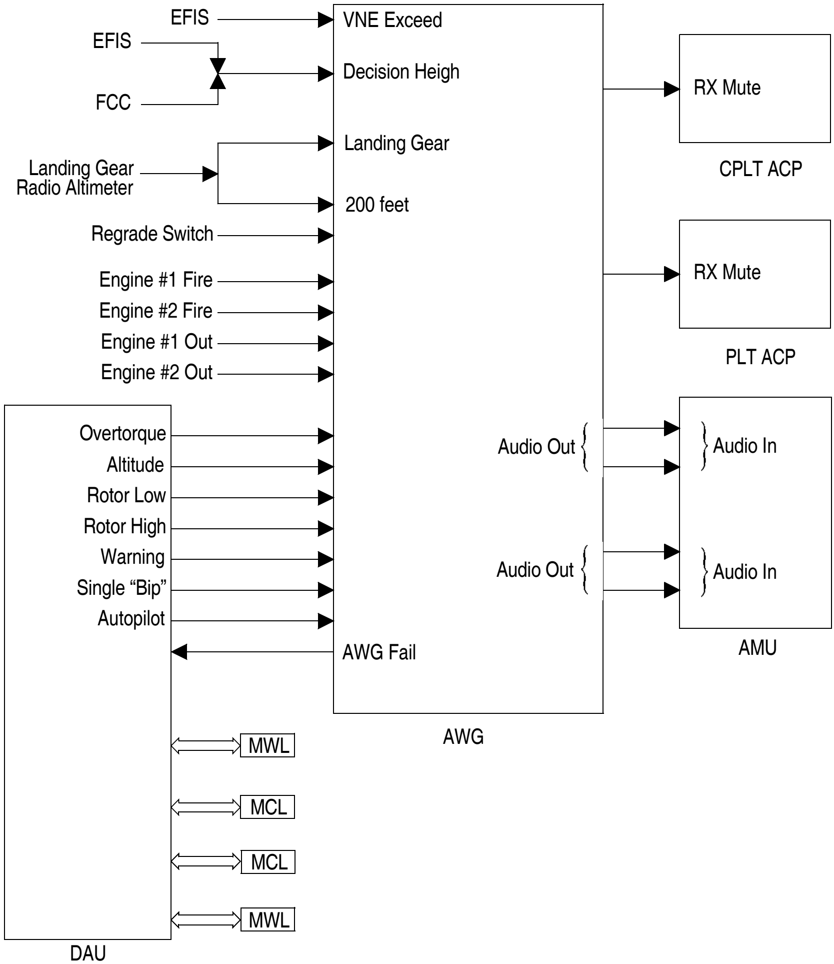

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Figure 7-86 Central Warning System - Schematic Diagram

Table 7-2 Priority Sequence of the Aural Messages

Priority	Aural Message	Activation	Audio Signal	Deactivation
1	ROTOR LOW	Main rotor speed below 96% (Power - ON) or below 95% (Power - OFF)	Precursor tone 1 Precursor tone 1 Precursor tone 1	Manual
2	ENGINE 1 OUT or ENGINE 2 OUT	Gas generator turbine speed (N1) below 50%	Precursor tone 2 Precursor tone 2	Automatic or manual
3	ENGINE 1 FIRE or ENGINE 2 FIRE	Overheat or fire in the related engine compartment	Precursor tone 3 Precursor tone 3	Manual
4	WARNING WARNING	Any warnings except those mentioned in this table	Precursor tone 4 Precursor tone 4	Automatic or manual
4	ROTOR HIGH	Main rotor speed above 105% (Power - ON) or above 110% (Power - OFF)	Precursor tone 4 Precursor tone 4	Automatic or manual
4	AUTOPILOT	Deactivation of one or both the channels of the AFCS	Precursor tone 4	Automatic or manual
5	TRANSMISSION OVERTORQUE	Torque value of the main transmission above 214% for AEO condition and above 161% for OEI condition	Precursor tone 5 Precursor tone 5	Automatic or manual
5	AIRSPEED	Airspeed exceeds the Vne for the current flight condition	Precursor tone 5 Precursor tone 5	Automatic or manual

Table 7-2 Priority Sequence of the Aural Messages (Continued)

Priority	Aural Message	Activation	Audio Signal	Deactivation
6	LANDING GEAR	The helicopter is flying below 200 ft with retracted landing gear	Precursor tone 6	Automatic (The automatic reset is obtained climbing to an altitude above 200 ft)
7	TWO HUNDRED FEET	The helicopter is flying below 200 ft	Precursor tone7	Automatic or manual (The automatic reset is obtained climbing to an altitude above 200 ft or by setting the AWG switch to REGRADE)
7	ALTITUDE	The current value of the "Altitude" datum" differs from the reference value of the ALT operation mode of the AFCS more than 200 ft (above 1000 ft).	Precursor tone 7	Automatic or manual
8	None (only precursor tone)	Activation/ deactivation, degradation or change of the reference value for the AFCS operation modes	Precursor tone 8	Automatic or manual

The indicators comprise a blinking light and a PUSH-TO-RESET push-button. The blinking light indicates, when it is on, that one or more caution messages appeared on the EDU1.
After the blinking light starts to blink and the caution message has been noted, the PUSH-TO-RESET push-button can be pushed to switch off the blinking light.
The caution messages on the EDU1 stay active.

The MASTER WARNING indicators are also installed in the left and right corners of the instrument panel.
The indicators comprise a blinking light and a PUSH-TO-RESET push-button. The blinking light indicates, when it is on, that one or more warning messages appeared on the EDU1.
After the blinking light starts to blink and the warning message has been noted, the PUSH-TO-RESET push-button can be pushed to switch off the blinking light and to silence any audio signal.
The warning messages on the EDU1 stay active.

The MASTER RESET push-buttons on the collective stick grips permit whether to switch off the blinking lights or to silence the audio signals.

The AWG switch, installed on the electrical control panel of the overhead console, has three positions: NORM, REGRADE and TEST.

When the AWG switch is set to NORM, all the audio signals can be activated.

When the AWG switch is set to REGRADE, the TWO HUNDRED FEET aural message can not be activated.
This aural message intends to draw the pilot attention that the helicopter is descending below 200 ft over the terrain.
If this aural message is present, it can be silenced by setting the AWG switch to REGRADE.

When the AWG switch is set to the TEST momentary position, the TEST OK aural message is sent to the pilot headsets.
If the AWG switch is held pushed for more than 6 s, all the aural messages are played in this sequence:

– ROTOR LOW
– ENGINE ONE OUT
– ENGINE TWO OUT
– ENGINE ONE FIRE

- ENGINE TWO FIRE
- ROTOR HIGH
- WARNING
- AUTOPILOT
- TRANSMISSION OVERTORQUE
- AIRSPEED
- LANDING GEAR
- TWO HUNDRED FEET
- ALTITUDE
- Audio tone (transition of an AFCS automatic operation mode).

The horn is installed on the right side of the cockpit, on the bulkhead behind the pilot.
The horn is activated together with the ROTOR LOW aural message.

CENTRAL WARNING SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-87.

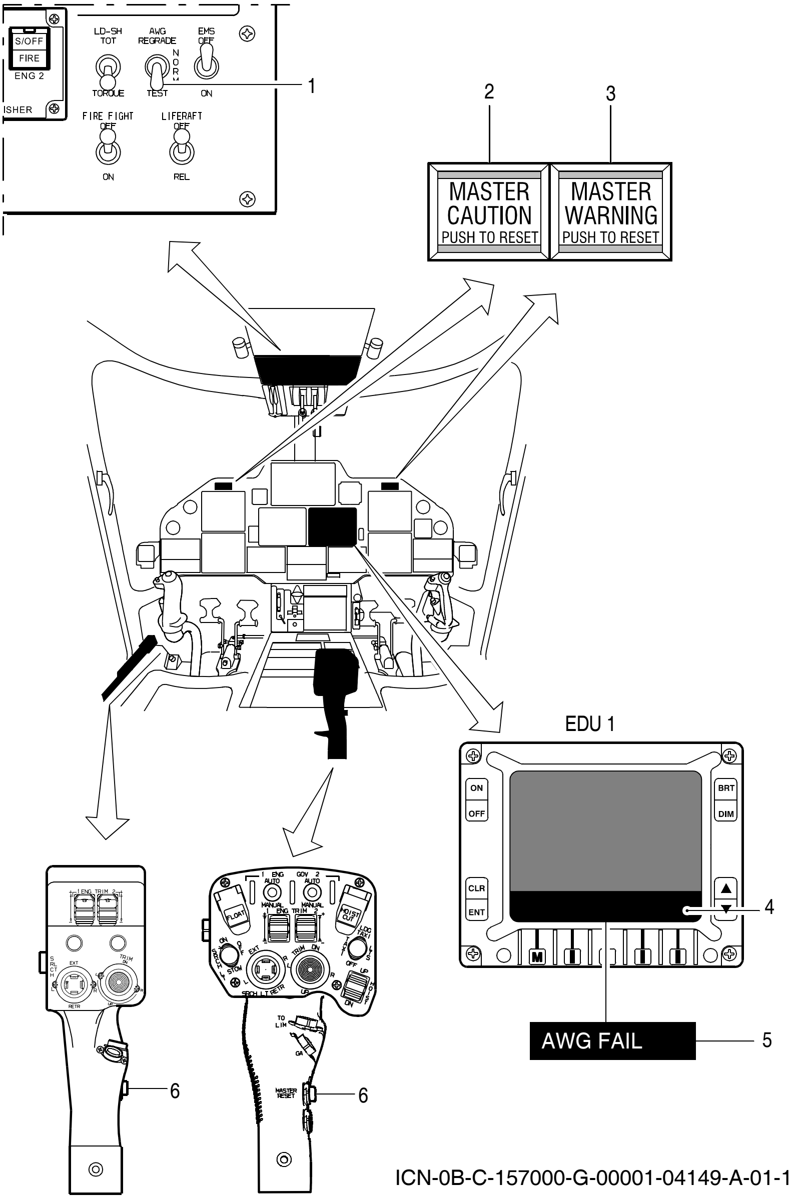

Figure 7-87 Central Warning System - Controls and Display

Key to Figure 7-87

Ref.	Control/Display	Function
1	AWG switch	NORM - all aural messages can be activated. REGRADE - the activation of the “Two hundred feet” message is disabled. TEST (momentary position) - when pressed momentarily the aural message “TEST OK” is heard in the pilots’ headsets. If pressed for more than 6 seconds the AWG is commanded to transmit all the aural messages in sequence.
2	MASTER CAUTION light/push-button switch	Comes on every time a caution event occurs. When on, it blinks and, when pressed, resets the light and any aural condition message, while the caution message remains displayed on the EDU.
3	MASTER WARNING light/push-button switch	Comes on every time a warning event occurs. When on, it blinks and, when pressed, resets the light and any aural condition message, while the warning message remains displayed on the EDU.
4	No 1 EDU	Show all the warning, caution and advisory messages. Refer to Table 7-2for the complete lists of messages.
5	AWG FAIL caution message	Shown in case of failure of the Aural Warning Generator (AWG).
6	MASTER RESET push-button switch	When pressed, performs the reset of both the MASTER WARNING and MASTER CAUTION lights.

COCKPIT MONITOR

The cockpit monitor is a TFT LCD display unit installed at the center of the instrument panel. It can be connected to a variety of systems, including the Digital Map Generator System (optional) and the External Video Camera System (optional).

The system is powered through the circuit breaker that follows:

– DMG DSPL (28 V DC MAIN BUS #2)

COCKPIT MONITOR CONTROLS AND DISPLAYS

The multifunction push-button on the instrument bezel are used to navigate different menus and have different functions according to the configuration and the different systems the cockpit monitor interfaces to. For the basic controls and displays, refer to Figure 7-87A. For detailed information about controls and displays related to the specific systems that can be connected, refer to the related paragraph of the system.

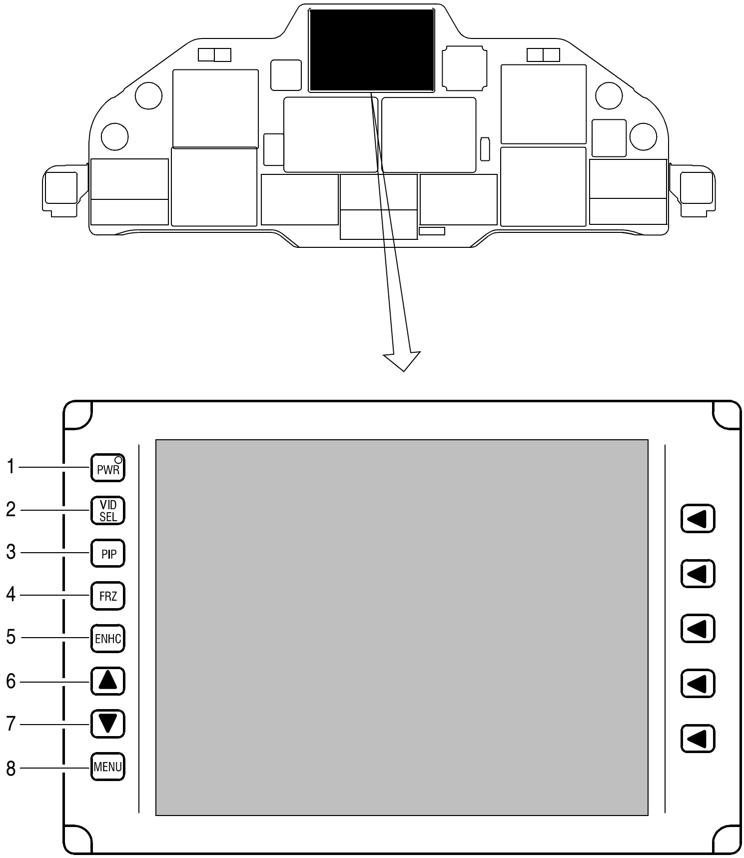

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Figure 7-87A Cockpit monitor - Controls and Displays

Key to Figure 7-87A

Ref.	Control/Display	Function
1	PWR	Pressing this button toggles power to the monitor on and off. A green LED illuminates to indicate when the unit is on.
2	VID SEL	This button selects which video input is displayed. When pressed, this button will cycle through each of the available video inputs.
3	PIP	Pressing this button toggles the picture-in-picture display on and off.
4	FRZ	Pressing this button will freeze the main display. If the picture-in-picture function is on, this image will not be frozen.
5	ENHC	Pressing this button switches on the enhancement function with the small window size selected initially. Further presses will cause the unit to step through other window sizes and finally the enhancement will be turned off.
6	UP arrow	Scrolls the menu up.
7	DOWN arrow	Scrolls the menu down.
8	MENU	Pressing this button will allow the operator to select various parameters for adjustment using the up and down arrow keys. The selected parameter will be shown on the on-screen display. The following may be adjusted: - Brightness - Contrast - Saturation (video mode only) - Backlight intensity. After a period of five seconds without a key-press, the on-screen display will be removed form the display.

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LANDING GEAR

(Chapter 32)

GENERAL

The helicopter is equipped with tricycle landing gear. The landing gear permits take-off, landing, taxiing and towing from prepared and semi-prepared surfaces with a maximum gross weight of 3175 kg.

The landing gear system includes:

- The main landing gear
- The nose landing gear
- The extension and retraction system
- The wheels and brake system
- The nose-wheel centering system
- The position and warning system
- The snow skis installation (optional)
- The slump protection pads installation (optional)
- The fixed tricycle-type landing gear (optional).

LANDING GEAR CONTROL AND DISPLAYS

Refer to Figure 7-88.

MAIN LANDING GEAR

The Main Landing Gear (MLG) consists of a right and a left main landing gear installed symmetrically on the right and left sponson of the fuselage underside.

Each MLG consists of a strut attached to the sponson with a forward bracket and an aft bracket, which permits landing gear retraction and a hydraulic actuator, which extends the landing gear.

The main wheel and brake assembly is attached to a stub axle at the lower end of the oleo strut.

A charging valve on the bottom of the strut permits the strut to be serviced.

Two microswitches (right Weight-On-Wheel and left Weight-On-Wheel (WOW)) are attached to the right and to the left landing-gear strut.

The left WOW microswitch gives indication about the position of the left landing-gear strut to the landing-gear hydraulic control-panel.

The right WOW microswitch gives indication about the position of the right landing-gear strut to the Data Acquisition Unit.

The WOW microswitches are activated by the scissors mechanism and gives the Weight-on-Wheels information to other helicopter systems.

The MLG actuators are connected the helicopter structure at one end and to the MLG strut at the other. When the landing gear is up the actuators are retracted. Each actuator incorporates a mechanical locking device and a microswitch. The locking device operates when the actuator is fully extended and actuates the microswitch connected to the landing gear position and warning system.

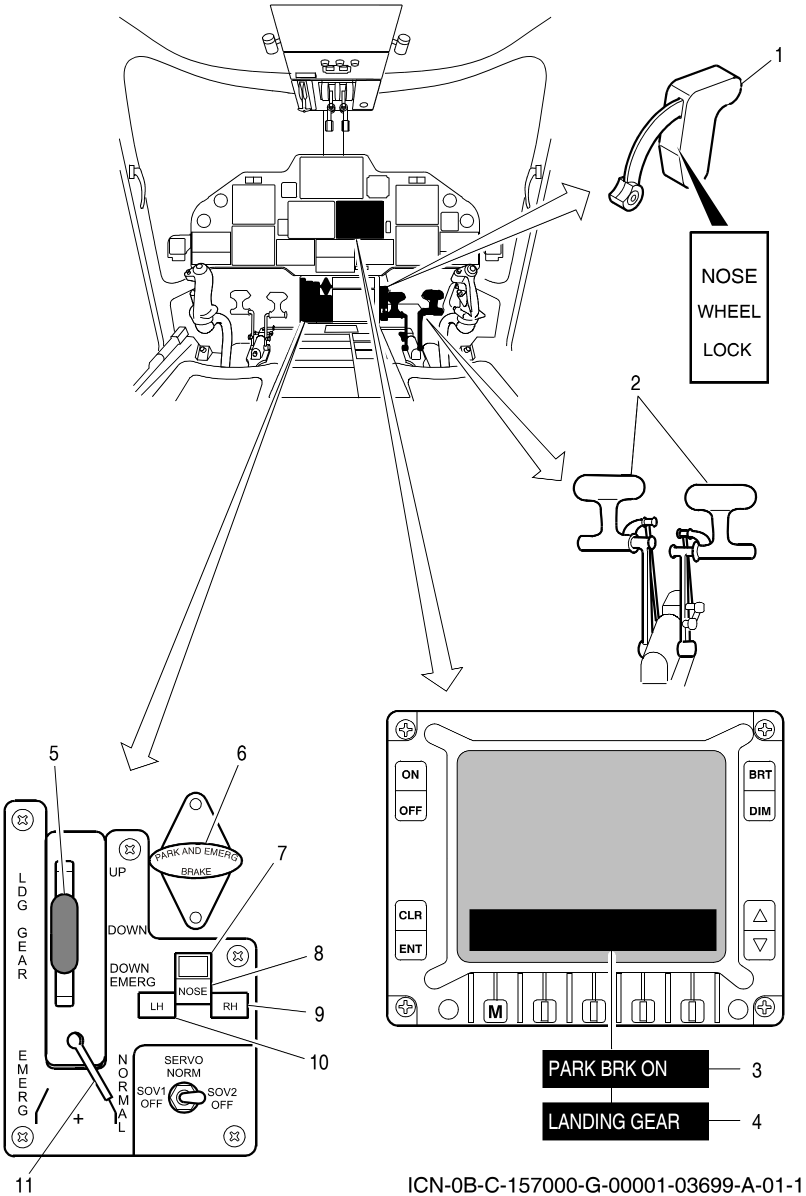

Figure 7-88 Landing Gear System - Controls and Displays

Key to Figure 7-88

Ref.	Control/Display	Function
1	NOSE WHEEL LOCK lever	Up - the nose-wheel centering actuator locks the nose landing gear. Down - the nose-wheel centering actuator unlocks the nose landing gear.
2	Brake pedals	When pressed on the ground they act on the brake system.
3	PARK BRK ON caution message	Comes in view when the parking brake is applied and the helicopter is on the ground.
4	LANDING GEAR caution message	Comes in view when the landing gear is retracted and the helicopter altitude is less than 200 ft. An audio tone and the LANDING GEAR voice message are also sent to the headsets. In this condition, you must put the LDG GEAR lever to DOWN.
5	LDG GEAR lever	UP position - the landing gear retracts and stops automatically in the retracted and locked position. DOWN - the landing gear up-locks open and the landing gear extends and locks automatically in the extended position. DOWN EMERG - the landing gear up-locks open and the landing gear extends with hydraulic power supplied from the emergency extension circuit. To set the LDG GEAR lever in the DOWN EMERG position the EMERG/ NORMAL lever (8) must first be rotated to the EMERG position.
6	PARK AND EMERG BRAKE handle	When pulled and rotated, the parking brakes are applied. When pulled, the brake selector valve is open proportionally of the handle level to obtain a modulate emergency brake action.
7	Red light	Shows that the landing gear is in movement from UP to DOWN or from DOWN to UP. The red light comes on when the landing gear starts to move and goes off when it stops.

Key to Figure 7-88 (Continued)

Ref.	Control/Display	Function
8	NOSE green light	Shows that the nose landing gear is locked in the down position.
9	RH green light	Shows that the right main landing gear is locked in the down position.
10	LH green light	Shows that the left main landing gear is locked in the down position.

NOSE LANDING GEAR

The Nose Landing Gear (NLG) consists of a strut attached to the air-frame. In the retracted position, it is housed in a bay closed by two doors. The doors open during nose landing gear extension and remain open.

The NLG is locked in the up and down positions. An up-lock located in the wheel bay engages a pin fitted to the strut.

The down-lock is incorporated in the nose gear actuator. The NLG is also fitted with a centering lock and an anti-shimmy device.

The lower section of the strut incorporates a wheel fork that can move 40° either side of the centerline to facilitate steering.

Stops are incorporated in the top part of the centering lock collar to prevent excessive movement.

EXTENSION AND RETRACTION SYSTEM

Hydraulic actuators that are powered by the utility hydraulic system extend and retract the landing gear.

The hydraulic circuit of the extension/retraction system consists of the landing gear selector valve, the nose and main landing gear actuators, the nose and main landing gear up-locks and the related hydraulic lines.

The movement of the landing gear is controlled through the LDG GEAR lever installed on the front panel of the landing-gear selector-valve.

The LDG GEAR lever has three positions: UP, DOWN and DOWN EMERG.

– When the LDG GEAR lever is set to UP, the hydraulic pressure is applied only to the retraction port of each actuator. This causes the release of the down-locks and the retraction of the landing gear. When the roller installed on each landing-gear strut touches and then pushes the uplock, the spring-loaded latch locks the roller. In this condition, the landing gear stays in the retracted position.
– When the LDG GEAR lever is set to DOWN, the hydraulic pressure is applied to the normal port of each uplock and to the normal extension port of each actuator.
  This causes the release of the uplocks and the extension of the landing gear. When the landing gear is fully extended, the down-locks, contained in the actuators, lock the landing gear in the down position.
– When the LDG GEAR lever must be set to DOWN EMERG, it is first necessary to move the EMERG/NORMAL lever to the EMERG position.
  The hydraulic pressure is applied to the emergency port of each uplock and to the emergency extension port of each actuators. This causes the release of the uplocks and the extension of the landing gear. When the landing gear is fully extended, the down-locks, contained in the actuators, lock the landing gear in the down position.

When the landing gear reaches the retracted position, it actuates the microswitch attached to each up-lock support. When actuated the microswitches break the circuit to the red landing-gear moving indicator advisory light.

The main components of the extension and retraction system are shown in Figure 7-89.
Refer to Figure 7-90 for the electrical schematic diagram of the system.
Refer to Figure 7-91 for the hydraulic schematic diagram of the system.

The system is powered through the circuit breaker that follows:

– LDG GEAR CONT (EMERG BUS #1).

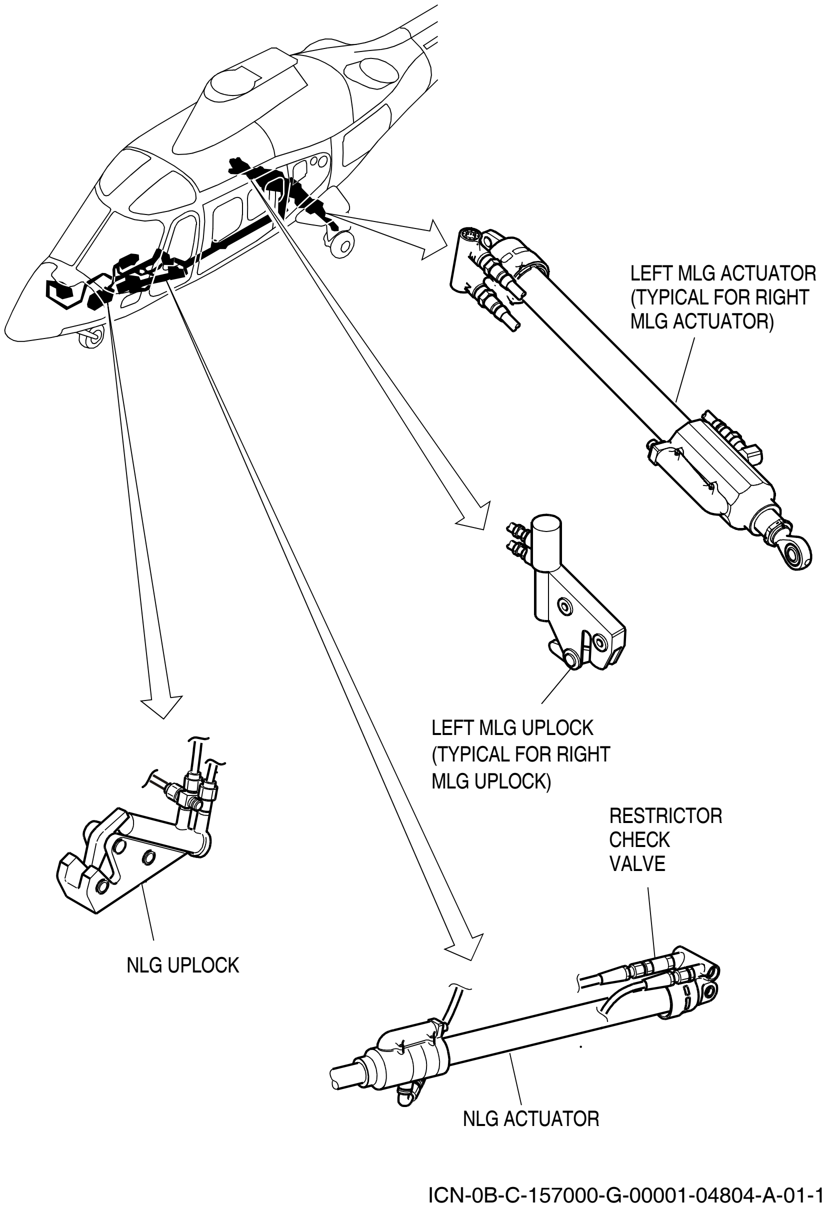

Figure 7-89 Extension and Retraction System - Component Location (Sheet 1 of 2)

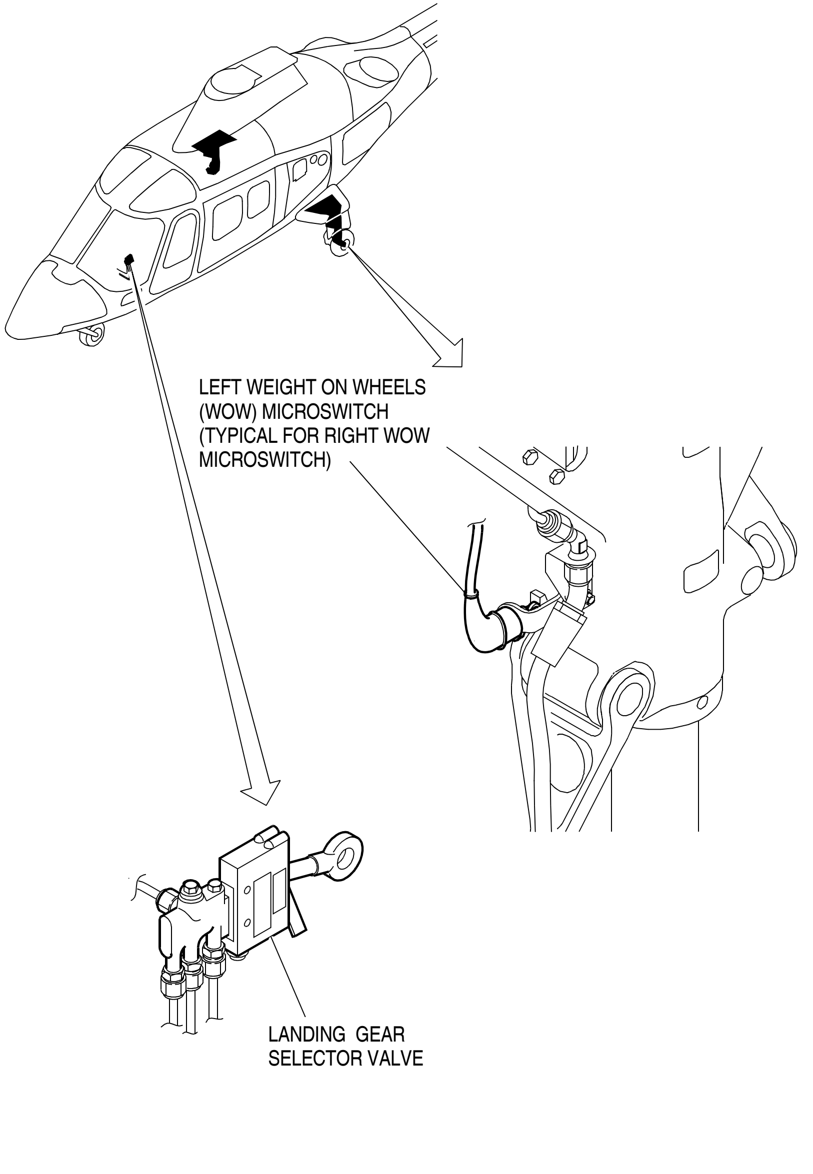

Figure 7-89 Extension and Retraction System - Component Location (Sheet 2 of 2)

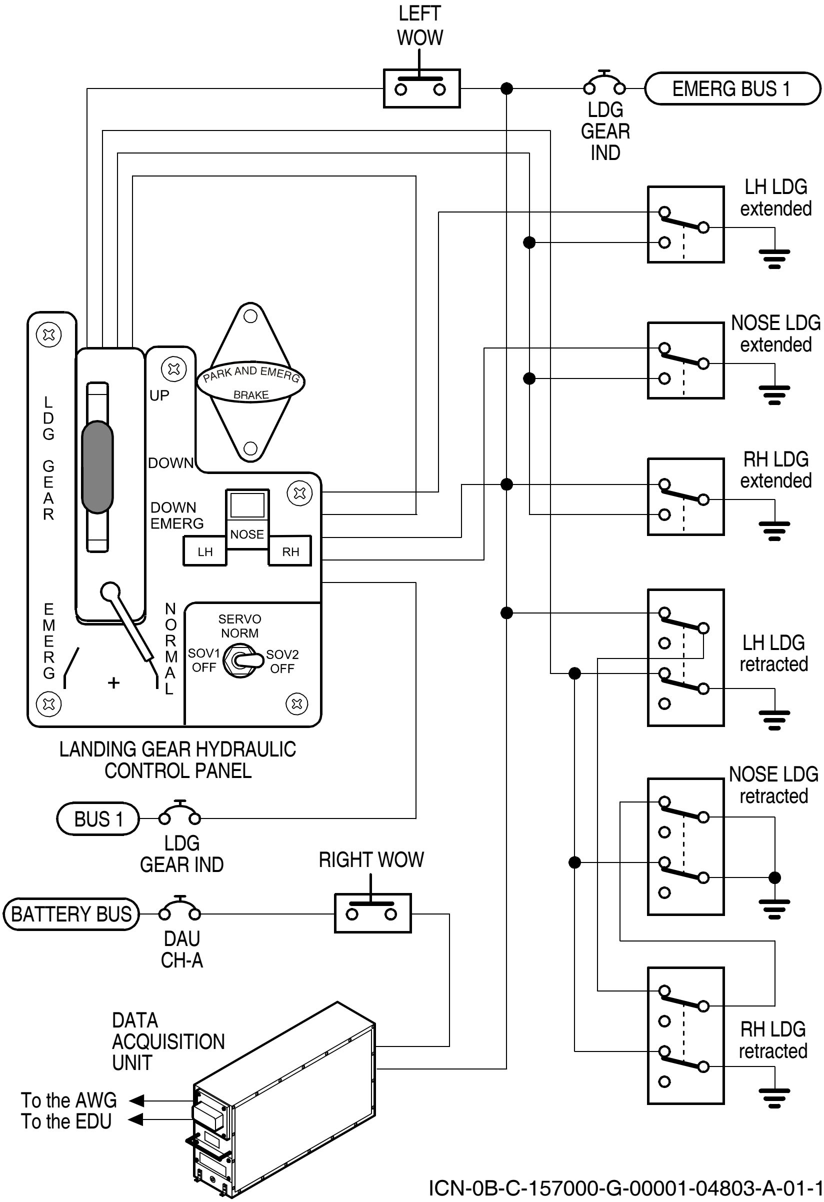

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Figure 7-90 Extension and Retraction System - Electrical Schematic Diagram

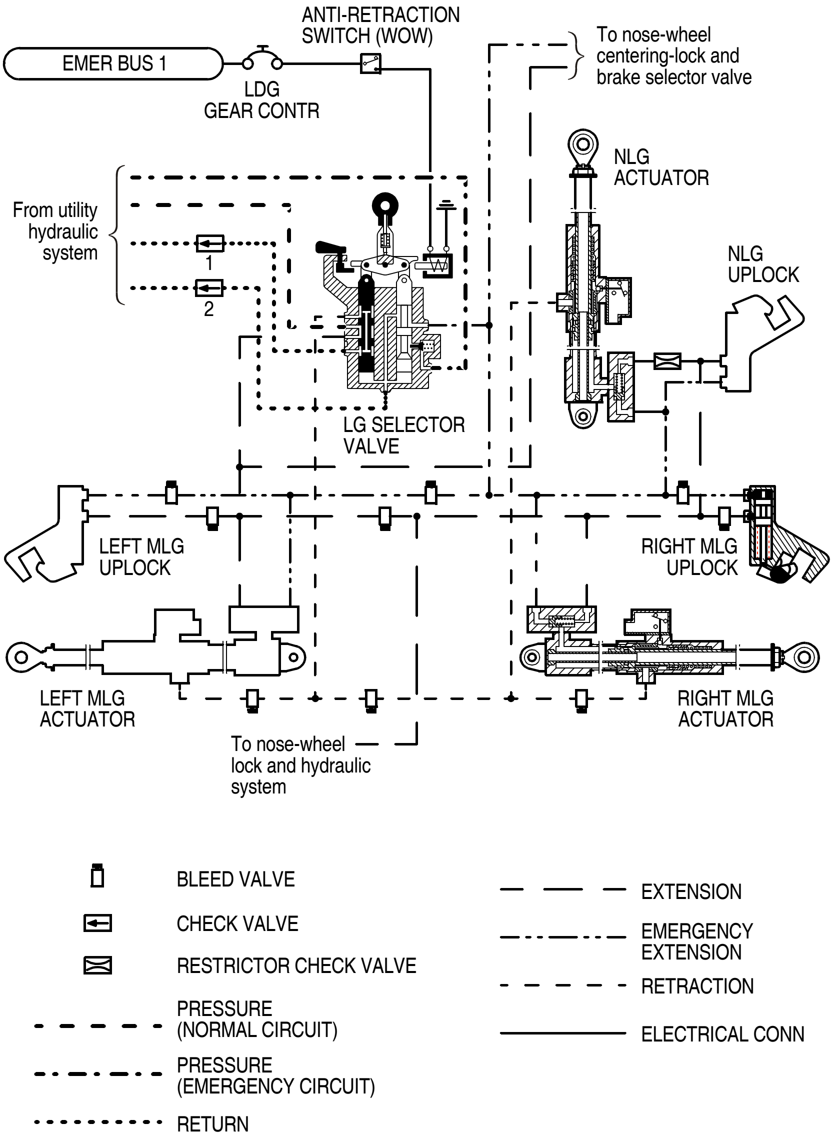

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Figure 7-91 Extension and Retraction System - Hydraulic Schematic Diagram

WHEELS AND BRAKE SYSTEM

The main components of the wheels and brake system are shown in Figure 7-92.

WHEELS

Each main wheel consists of two magnesium-alloy wheel-halves attached with five bolts.

Each main wheel has a tubeless tire. The outer wheel-half has the air inflation valve, while the inner wheel-half is provided with the keyway liners for the brake disc tooth.
The nose wheel consists of two magnesium-alloy wheel-halves with five bolts. The nose wheel has a tubeless tire. One wheel-half is provided with the air inflation valve.

BRAKE SYSTEM

The two MLG wheels have disc-type brakes which are hydraulically operated by the control valve which is mechanically connected to the brake pedals that are fitted to the pilot's control pedals.
The hydraulic power to operate the control valve is supplied by the normal circuit of the utility hydraulic system.

If no pressure is available in the normal circuit, the brakes can be operated with the pressure of the emergency circuit. In this case the brakes are not applied by means of the pedals but by means of the PARK AND EMERG BRAKE handle, through the nose gear centering lock and brake selector valve located in the nose compartment.
The brake control valve pressurizes the wheel brake circuit, located below the pilot's seat.

The valve is controlled by rods and levers connected to the brake pedals installed on the pilot yaw control pedals. Two shuttle valves are installed in the hydraulic circuit of the wheel brakes. The shuttle valves connect the wheel brakes to either the normal or the emergency circuit.

Refer to Figure 7-93 for the hydraulic schematic diagram of the wheels and brakes system.

NOSE WHEEL CENTERING LOCK SYSTEM

The nose-wheel-centering-lock system consists of the nose centering lock actuator, the nose-wheel centering-lock control lever, the nose-wheel centering-lock and brake selector valve and the related hydraulic lines (Figure 7-91).

The nose-wheel centering-lock actuator piston is hydraulically controlled by the nose wheel-centering-lock and brake-selector valve, and it is mechanically connected to the nose-wheel centering-lock lever located on the right side of the console.

POSITION AND WARNING SYSTEM

The helicopter is equipped with a landing gear position indicating system consisting of a LDG GEAR POS indicator panel located on the front panel, and a set of switches installed as follows:

– One in each landing gear actuator
– One in each landing gear up-lock
– Three green landing gear down-locked lights
– One red landing gear moving light
– One test push-button switch.

When the landing gear is down and locked the three green lights are on. The red light goes off as soon as the landing gear is up-locked. With the landing gear up-locked all the four lights (green and red) must be off.
As soon as the landing gear starts to extend, the red light comes on. The red light goes off and the three lights come on as soon as the landing gear is down and locked.
The signals of the system are transmitted to the DAU. From the DAU the signals are transmitted to the AWG and EDU to provide visual and aural messages.

Refer to Figure 7-90 for the electrical schematic diagram of the system.

The system is powered through the circuit breaker that follows:

– LDG GEAR IND (EMERG BUS #1).

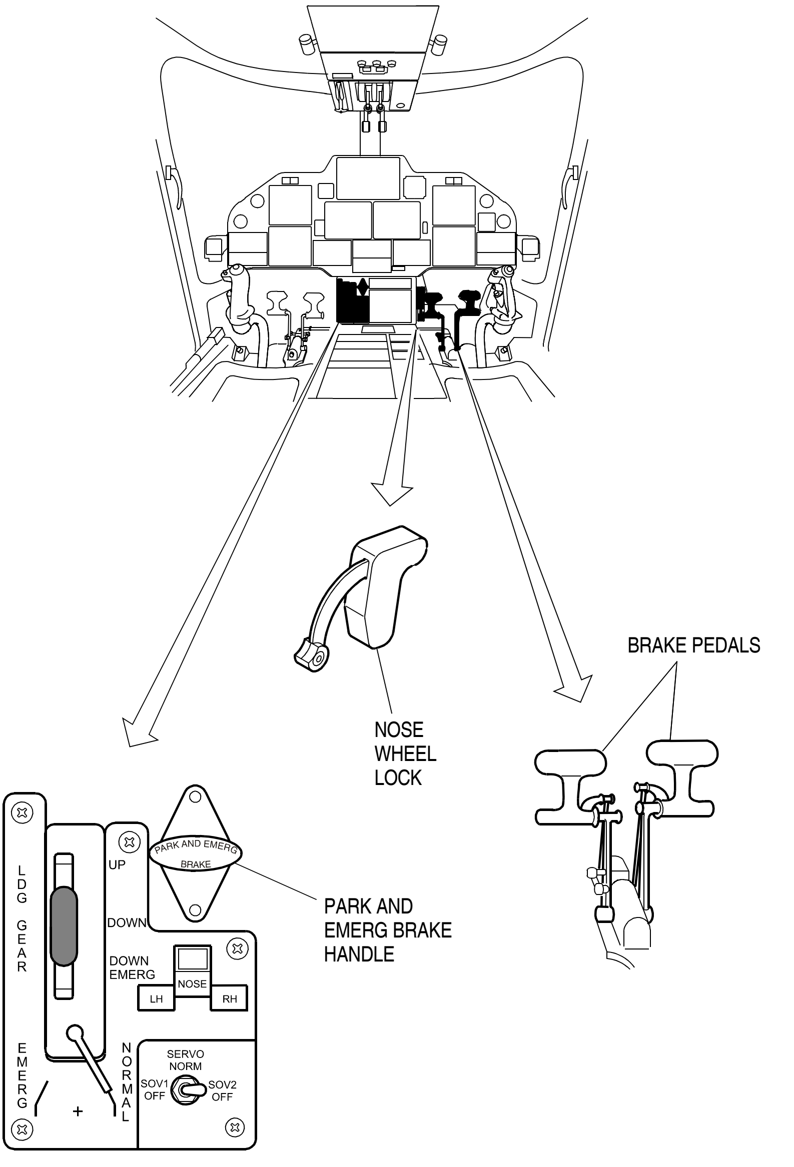

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Figure 7-92 Wheels and Brake System - Component Location (Sheet 1 of 2)

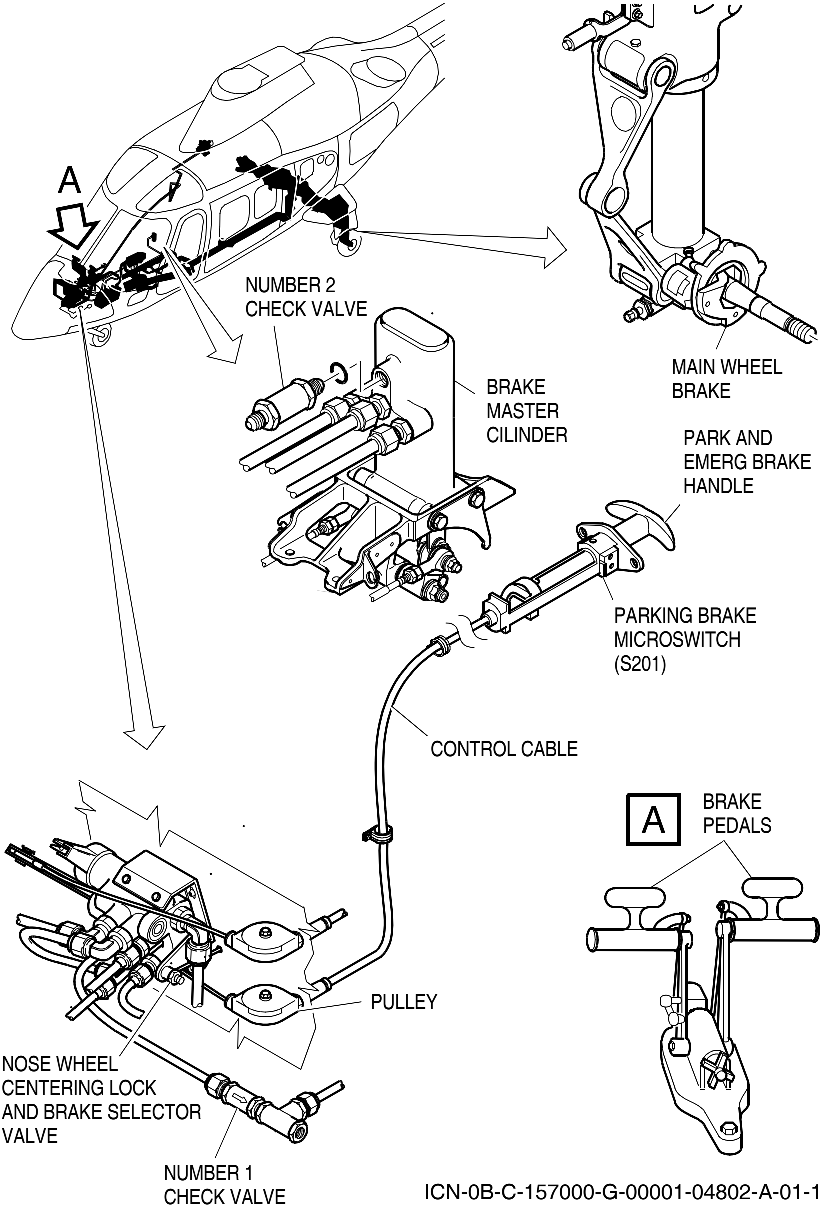

Figure 7-92 Wheels and Brake System - Component Location (Sheet 2 of 2)

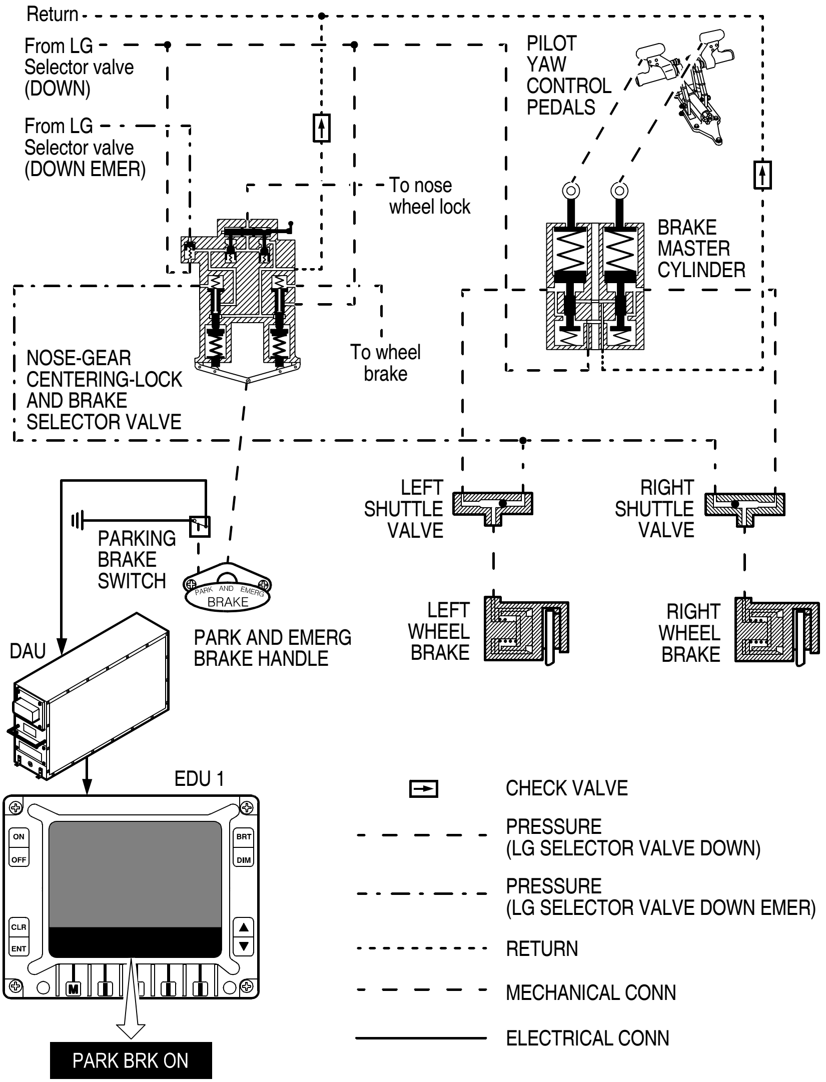

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Figure 7-93 Wheels and Brake System - Hydraulic Schematic Diagram (Sheet 1 of 2)

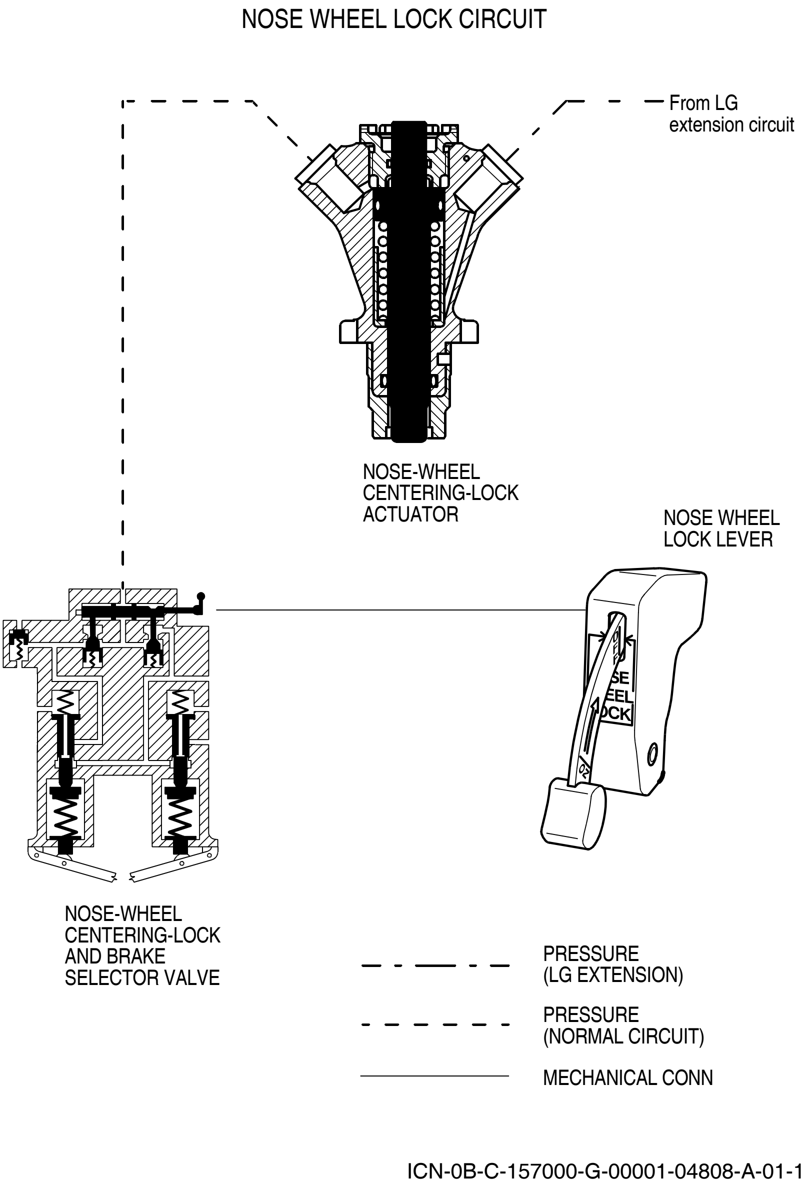

Figure 7-93 Wheels and Brake System - Hydraulic Schematic Diagram (Sheet 2 of 2)

SNOW SKIS INSTALLATION

The snow skis installation (in both versions “Fixed restraint cables” P/N 109-0811-99 and “Adjustable restraint cables” P/N 109G3271F01) is composed of three honeycomb panel-type skis attached to each wheel axle of the nose and main landing gears.

Each ski is held in position by retention blocks, bolts and nuts. Cables and bungee cords allow each ski to tilt during landing to adapt at ground surface.

The installation also includes a locking device to prevent operation of the landing gear control lever and a guard located over the nose wheel centering lock lever to prevent inadvertent operation.

The main components of the snow skis installation are shown in Figure 7-94.

SLUMP PROTECTION PADS INSTALLATION

The slump protection pads installation (in both versions “Fixed restraint cables” P/N 109-0811-73 and “Adjustable restraint cables” P/N 109G3272F01) is composed of three slump protection padss attached to each wheel axle of the nose and main landing gears. Each slump protection pads is made of aluminum alloy.

Each slump protection pad is held in position by retention blocks, bolts and nuts. Cables and bungee cords allow each pad to tilt during landing to adapt at ground surface.

The installation also includes a locking device to prevent operation of the landing gear control lever and a guard located over the nose wheel centering lock lever to prevent inadvertent operation.

The main components of the slump protection pads installation are shown in Figure 7-94A .

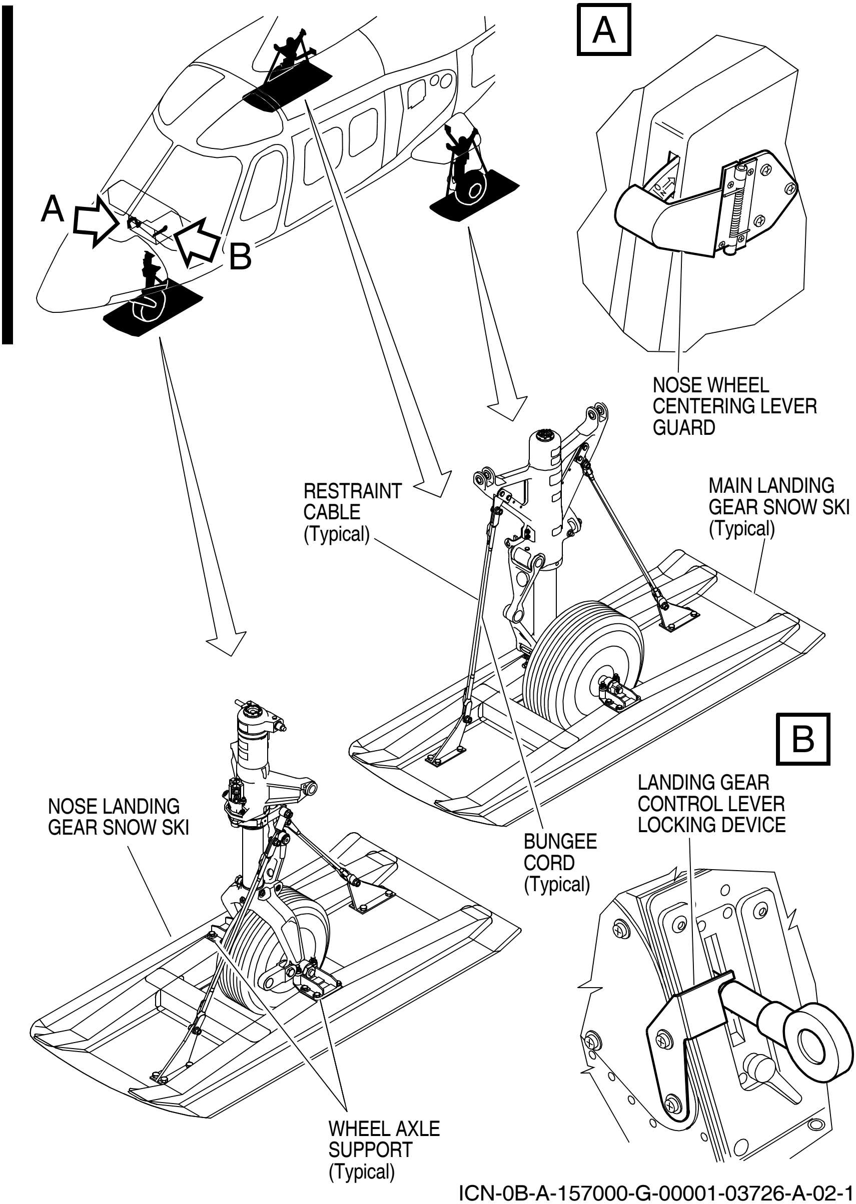

Figure 7-94 Snow Skis Installation - Component Location

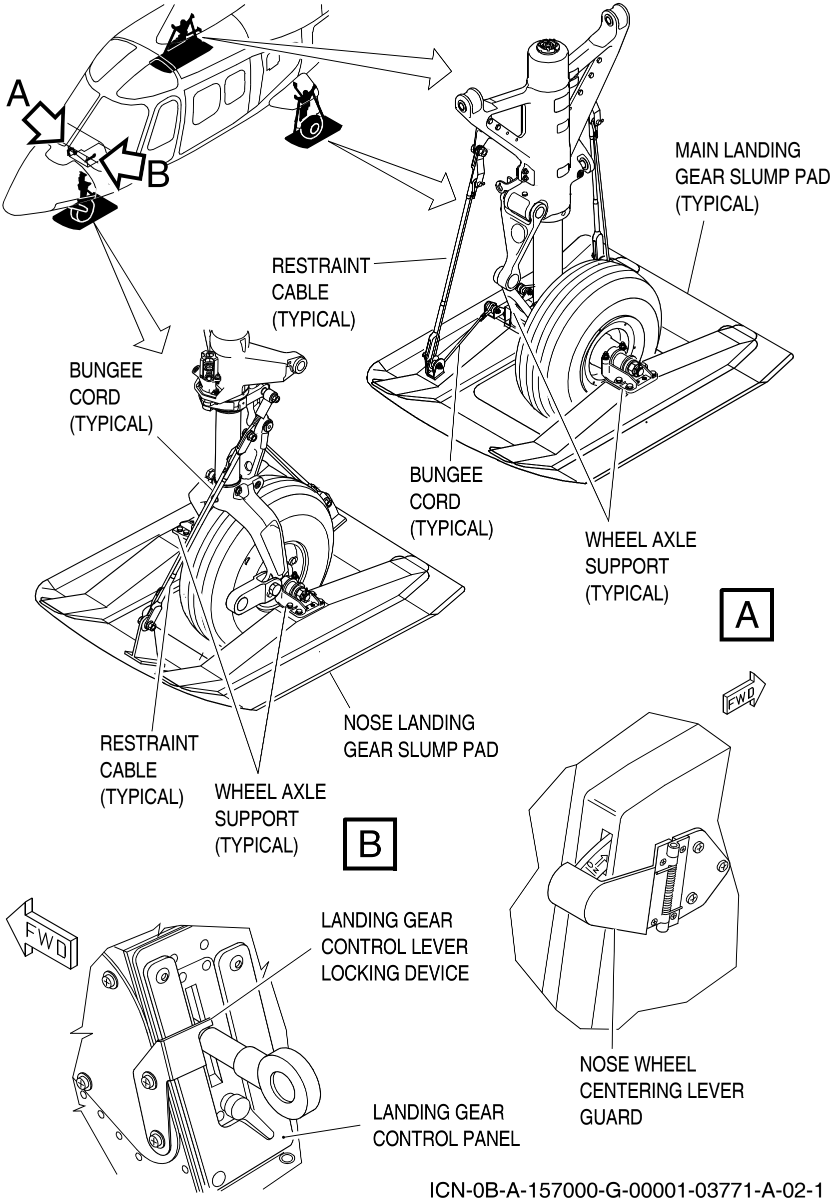

Figure 7-94A Slump Protection Pads Installation - Component Location

FIXED TRICYCLE-TYPE LANDING GEAR

The fixed tricycle-type landing gear includes a fixed main landing gear and a fixed nose landing gear.

When the fixed trycicle-type landing gear is installed, the extension and retraction system is not installed. Thus the landing gear control panel (see Ref. 5 and 7 thru 11 of Figure 7-88) is not installed and the LANDING GEAR caution message is not shown on the EDU1, Also the LANDING GEAR voice message is not transmitted to the headsets (see Ref. 4 of Figure 7-88),

FIXED MAIN LANDING GEAR

The fixed Main Landing Gear (MLG) consists of a right and a left main landing gear installed symmetrically on the right and left sponson of the fuselage underside.

Each MLG consists of a strut attached to the sponson with a forward bracket and an aft bracket. A push-rod holds the MLG in the extended position.

The main wheel and brake assembly is attached to a stub axle at the lower end of the oleo strut. A charging valve on the bottom of the strut permits the strut to be serviced.

Two microswitches (right Weight-On-Wheel and left Weight-On-Wheel (WOW)) are attached to the right and to the left landing-gear strut. The WOW microswitches are activated by the scissors mechanism and give the Weight-on-Wheels information to other helicopter systems.

FIXED NOSE LANDING GEAR

The fixed Nose Landing Gear (NLG) consists of a strut attached to the airframe. A push-rod holds the NLG in the extended position.

The NLG is fitted with a centering lock and an anti-shimmy device.

The lower section of the strut incorporates a wheel fork that can move 40° either side of the centerline to facilitate steering.

Stops are incorporated in the top part of the centering lock collar to prevent excessive movement.

LIGHTS

(Chapter 33)

GENERAL

The light system includes:

- The internal light system
- The external light system.

INTERNAL LIGHT SYSTEM

The internal lights system includes:

- The instrument lights
- The cockpit lights
- The cabin lights
- The baggage compartment lights.
- The map light system (optional).

INSTRUMENT LIGHTS

The instrument lights system includes three circuits. An appropriate knob controls each circuit, as follows:

- PED (collective grips and interseat console)
- INST PNL (main instrument panel)
- OVHD & CSL (overhead consoles).

The instrument lights are powered through the circuit breakers that follow:

- LT SYS OVHD CSL (28 V DC BUS #2)
- LT SYS INST PNL (28 V DC BUS #2)
- LT SYS PED (28 V DC BUS #1).

COCKPIT LIGHTS

The cockpit lights are installed on the left and right side of the overhead panel. Each spotlight includes a light unit, an offset mounting and a coiled cable.

A knob controlling a rheostat and a push-button are installed at the rear of the light unit. Rotation of the knob permits the brigthness of the spotlight to be adjusted, while the push-button causes the spotlight to go on only while it is kept pushed.
The offset mounting permits horizontal and vertical movement of the spotlight. The light unit can also be removed from its mounting and used as a hand-held flashlight.
The system is powered through the circuit breakers that follow:

– LT SYS CKPT CPLT (28 V DC BUS #1)
– LT SYS CKPT PLT (28 V DC EMER BUS #2).

CABIN LIGHTS

The cabin lights includes four lights. Two lights are installed on the cabin floor (one on each side) and two lights are installed on the cabin roof (one on each side).
Each cabin light has a push-button on its side to switch it on and off.

The cabin lights are powered through the toggle-type circuit breakers that follow:

– CABIN LT (28 V DC BUS #1)
– PASS WRN (28 V DC BUS #1).

ELECTRO-AVIONIC EQUIPMENT COMPARTMENT LIGHTS

The electro-avionic equipment compartment lights include two lights installed on the compartment roof.
A microswitch, operated by the compartment door, switches the lights on (when the compartment door is open) and off (when the door is closed). The same microswitch activates a BAG DOOR OPEN caution message on the EDU1 when the compartment door is open.

The lights are powered through the circuit breaker that follows:

– BAG LT (28 V DC AUX BUS #1).

INTERNAL LIGHT SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-95.

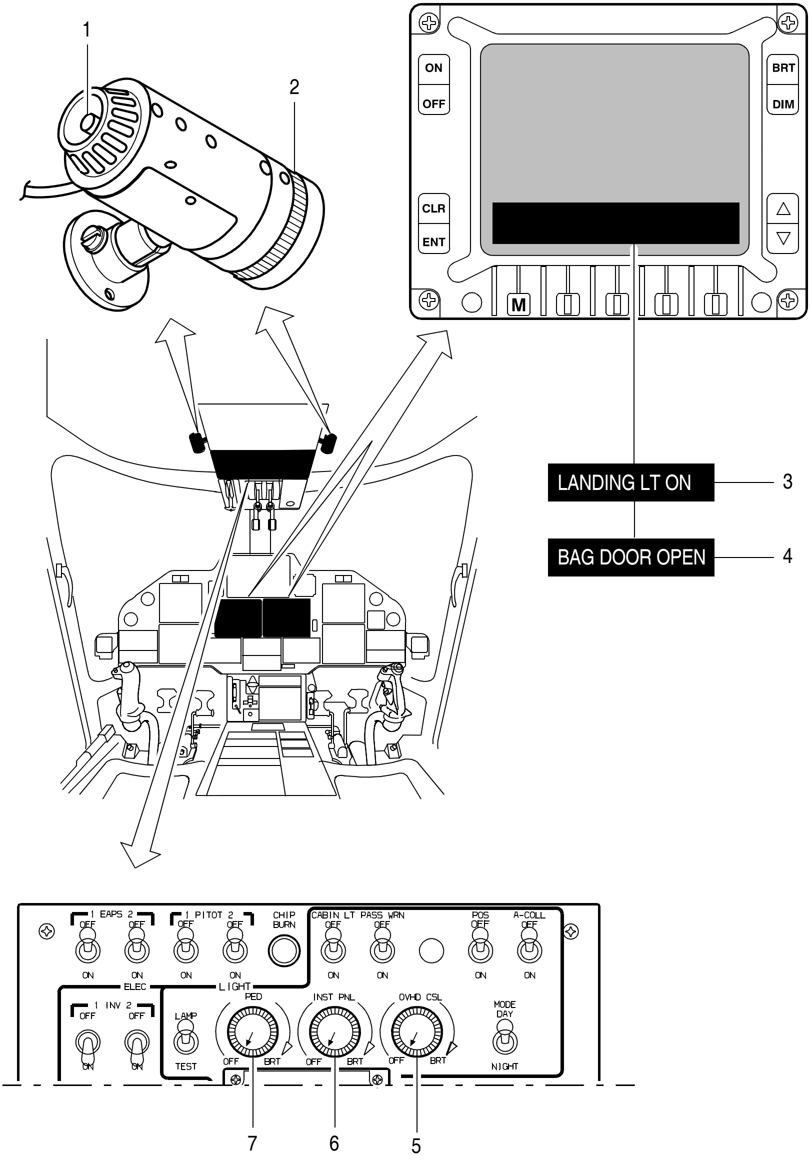

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Figure 7-95 Lights System - Controls and Displays (Sheet 1 of 2)

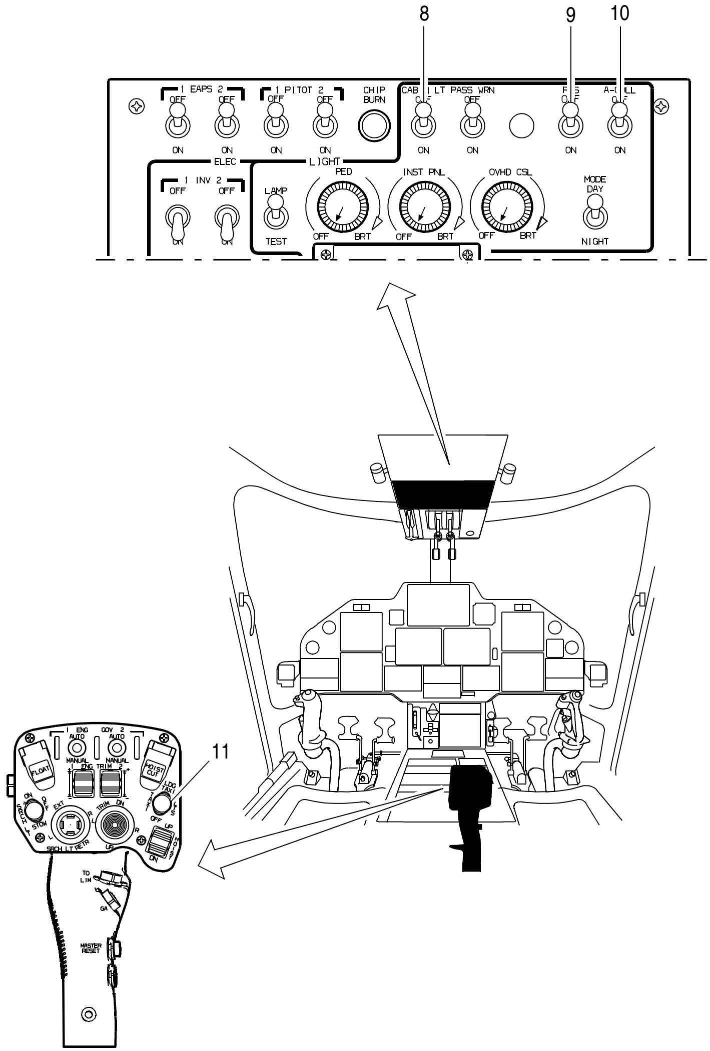

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Figure 7-95 Lights System - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-95

Ref.	Control/Display	Function
1	Spotlight momentarily illumination push-button	Pushed - the spotlight is ON only as long as the push-button is held.
2	Spotlight knob	Turned clockwise -switches on the light, and adjusts the brightness
3	LANDING LT ON advisory message	Comes in view on the two EDU when the LDG TAXI SWITCH at ON position.
4	BAG DOOR advisory message	Comes in view on the two EDU when the door of the electro-avionic compartment is open.
5	OVHD & CSL knob	Rotated clockwise - turns on and adjusts the brightness of the lights.
6	INST PNL knob	Rotated clockwise - turns on and adjusts the brightness of the main instruments.
7	PED knob	Rotated clockwise - turns on and adjusts the brightness of the collective grips and interseat console lights.
8	CABIN LT switch	ON - the light in the passenger cabin is on. OFF - the light in the passenger cabin is off.
9	POS switch	ON - the position lights are on. OFF - the position lights are off.
10	A-COLL switch	ON - the anticollision lights are on. OFF - the anticollision lights are off.
11	LDG TAXI switch	LDG TAXI - the landing and taxi lights are on. OFF - the landing and taxi lights are off.

MAP LIGHT SYSTEM

The map light system is installed on the left and right side of the overhead panel. The system includes a light unit and a potentiometer.
The knob that controls the potentiometer is placed near of the light unit. Turn the knob to adjust the intensity of the light and to turn it on and off. Refer to Figure 7-95A.

The following circuit breaker energize the system:

– LT SYS OVHD CSL (28 V DC BUS #2)

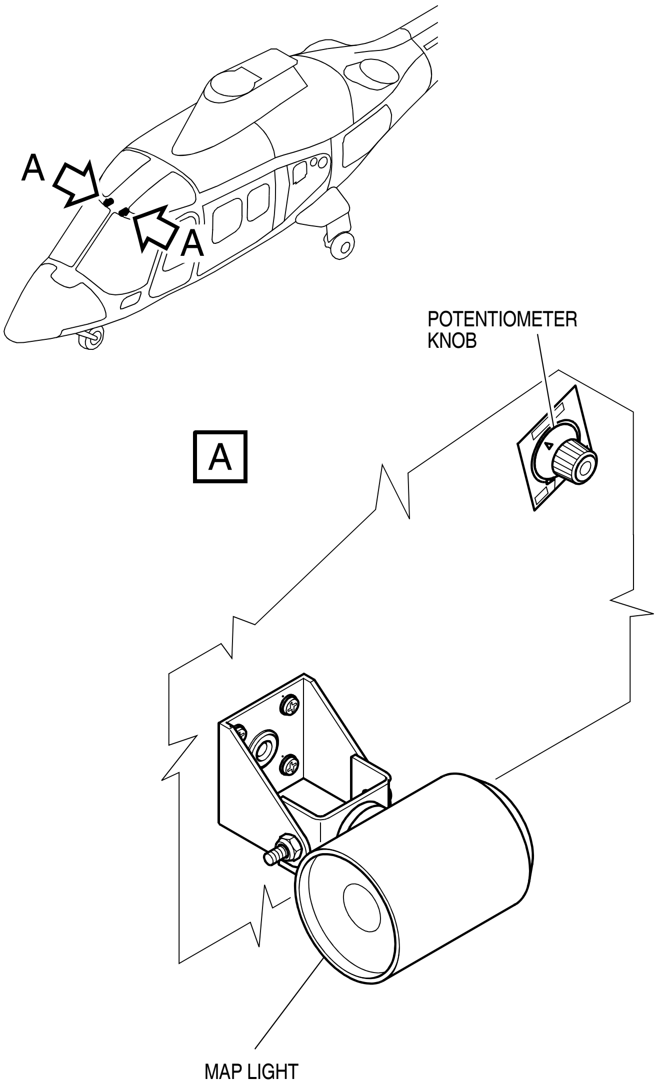

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Figure 7-95A Position of the Map Light System

EXTERNAL LIGHT SYSTEM

The external light system includes:

– The position lights
– The anti-collision lights
– The landing and taxi lights
– The searchlights system (optional).
– The strobe light system (optional).

Refer to Figure 7-96 for location of the external lights.

POSITION LIGHTS

The position lights are used to let the position of the helicopter in the air, its direction of flight or its position on the ground be seen from short distance, at night or in conditions of poor visibility.

The position lights are located as follows:

– one red lights on the left winglet of the horizontal stabilizer
– one green on the right winglet of the horizontal stabilizer
– one white on the tail (pointing aft).

The position lights are powered and controlled through the toggle-type circuit breaker that follows:

– LT SYS POS (28 V DC EMER BUS #1).

ANTI-COLLISION LIGHTS

The anticollision lights system includes two anticollision lights. One anticollision light is installed on the top of the fuselage and one is installed on the lower side of the tail boom.
The anticollision lights permit the helicopter position to be identified by other aircraft at a distance longer than that from which the position lights can be seen, thus preventing risks of collision.
The visible lights flash with aviation red light at a frequency of approximately 50 flashes per minute and their coverage is 30° above and below the horizontal axis of the helicopter.

The anti-collision lights are powered through the toggle-type circuit breaker that follows:

– LT SYS A-COLL (28 V DC BUS #2).

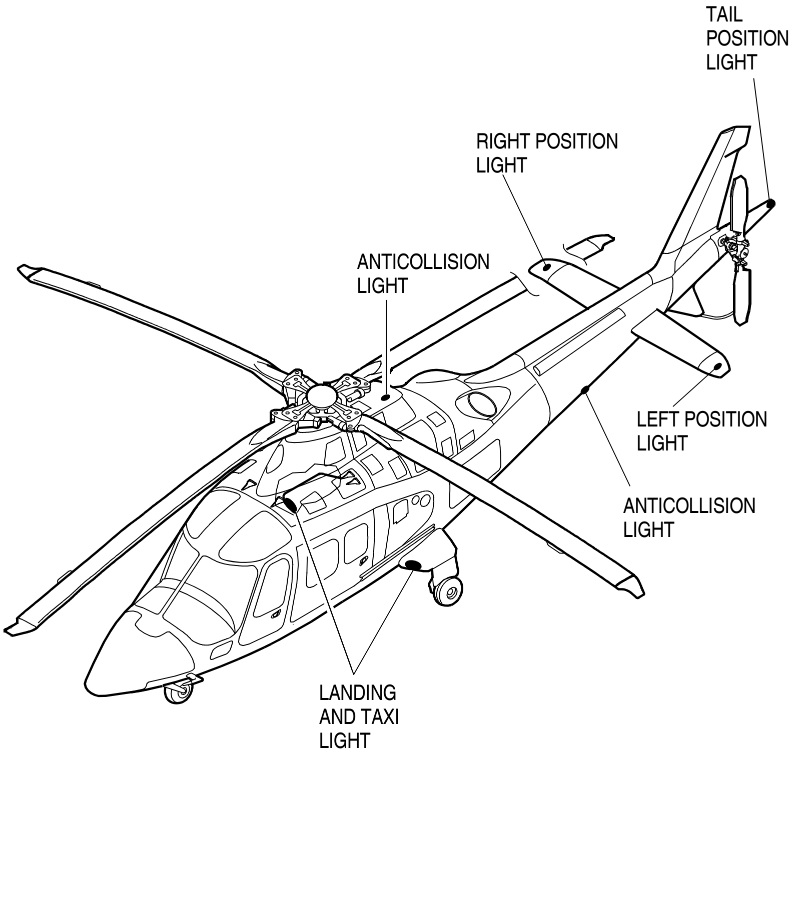

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Figure 7-96 Position of the External Lights

LANDING AND TAXI LIGHTS

The landing light and taxi consists of two landing lights beaming downward and two taxi lights beaming (parallel to the ground). The lights are installed inside each landing gear sponson protected by a fairing.
The landing and taxi lights are controlled through the LDG TAXI switch on the pilot's collective-lever switch-box. The LANDING LT ON advisory message comes on the EDU1 and 2 when the landing and taxi lights are on.
The landing and taxi lights are powered through the circuit breakers that follow:

– TAXI LT (28 V DC BUS #2)
– LDG/TAXI LT CTRL (28 V DC EMER BUS #1)
– LDG LT (28 V DC EMER BUS #1).

SEARCH LIGHT SYSTEM

The search light system is a controllable search light installed in the lower side of the helicopter nose. Four screws and four washers attach the search light to the panel. The light can be extended up to 120° from the fully retracted position and can be rotated towards left or right. Refer to Figure 7-97.
The pilot and the copilot control the movement of the light into the four directions through the directional light switch installed on each collective grip. The priority of operation is given to the pilot.
The search light system-circuit is energized when the search light control switch SRCH ON/OFF/STOW on the collective grip of the pilot is at ON. The copilot’s EDU shows the PRI SRCH LT advisory message.

The searchlight system is powered thru the circuit breakers that follow:

SRCH CTL (28 V DC EMER BUS #2)

– SRCH PWR (28 V DC EMER BUS #2).

SECOND SEARCH LIGHT SYSTEM

The search light system includes a search light and a control panel. The search light is installed in the right lower part of the fuselage, into a protective fairing. The control panel is installed on the interseat console. Refer to Figure 7-98.

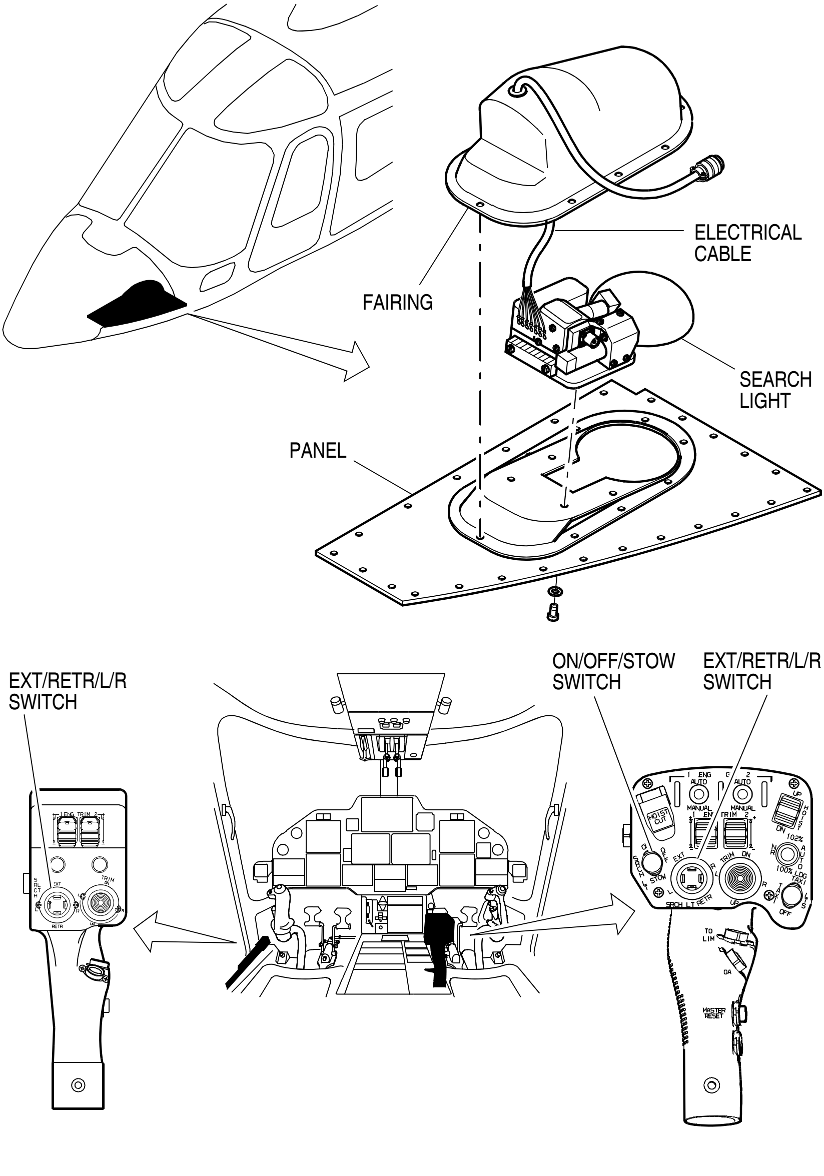

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Figure 7-97 Position of the Search Light Sytem

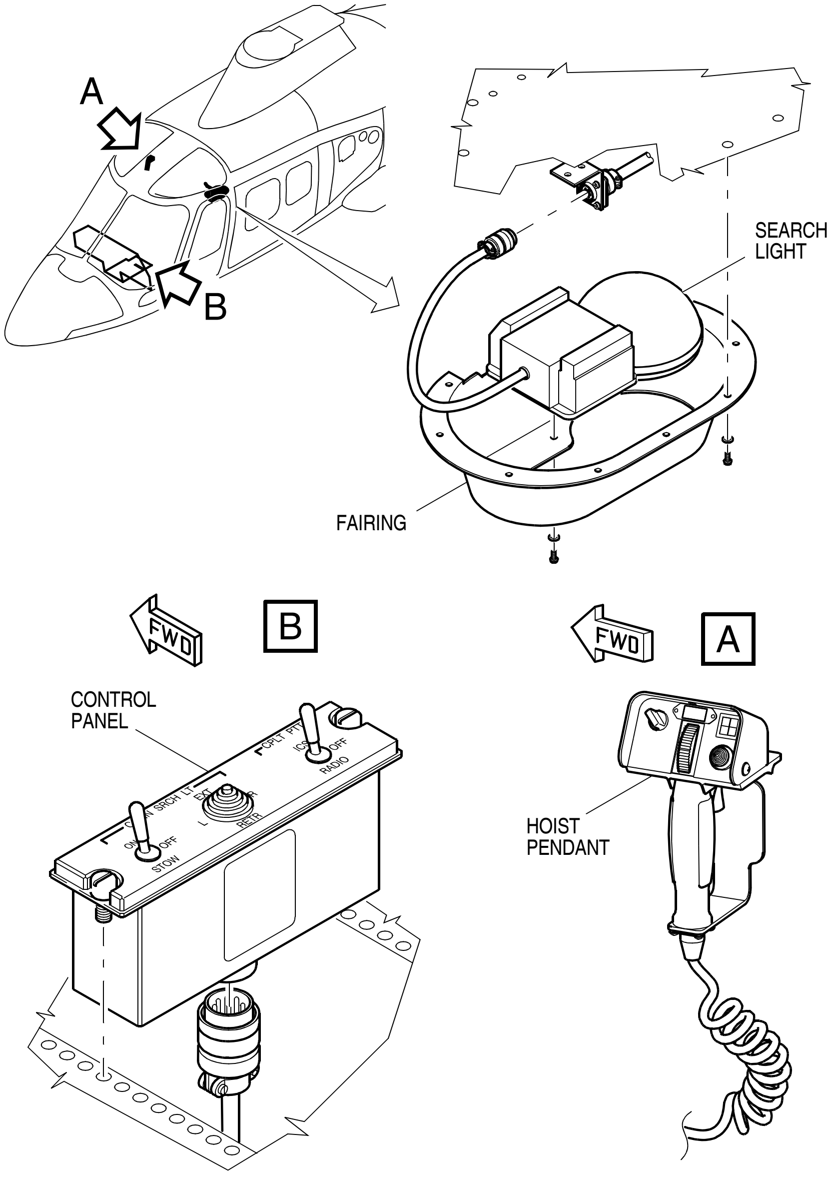

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Figure 7-98 Position of the Second Search Light Sytem

The search beacon can move up and down and turn in azimuth. The down travel is from 0° (stowed position) to 120°. The search light can turn 360° in azimuth, but in the rear arc of 180° it stays on only if its extension is 60° minimum. The search light can be extended up to a helicopter speed of 135 kts. When the search light is extended, the maximum speed at which it can stay extended is 180 kts.

The control panel has a three position (ON/OFF/STOW) switch and a four position (EXT/RETR/L/R) selector switch. The crew can use the control panel to set the light on or off, or to move and turn the search beacon. The search light can be controlled also by the hoist pendant. But the hoist pendant has only the controls to turn the search light in azimuth and for its extension/retraction.

The search light system is used for search operations at night, but can be used also during landing and taxiing at night. The SRCH LT selector switches move the search light in the four directions. If the two selector switches are operated at the same time, the control-panel selector-switch overrides the control switch of the hoist pendant.

When the search light is fully retracted and the SRCH LT switch is at ON, the search light comes on. At the same time the advisory message LANDING LT ON comes in view on the Number 2 Electronic Display Unit (EDU2). When the SRCH LT switch is set to STOW and the search light is in an extended position, the search light goes off. Then the search light turns until it aligns with the helicopter longitudinal axis and moves up until it is fully stowed. The LANDING LT ON advisory message shown on the EDU2 goes out of view.

The searchlight system is powered thru the circuit breakers that follow:

– CAB SRCH (MAIN BUS #1)
– CAB SRCH CTRL (MAIN BUS #1).

STROBE LIGHT SYSTEM

The strobe light system consists of two lights (right light and left light) and a power supply. The lights are installed on the outboard sides of the horizontal stabilizer.
An electrical cable supplies the necessary electrical power. The strobe light has a transparent lens.

The power supply is installed in the upper part of the baggage compartment. Four screws and a four washers attach it to the structure.

The power supply sends the electrical power to the two strobe lights at the intervals necessary to make the lighs flash at the approved rate. Refer to Figure 7-98A.

The strobe light system gives light signals that make the helicopter be seen at large distance. This will decrease the risk of collisions when the helicopter is in flight. A STROBE ON/OFF switch, positioned on the overhead panel, switches the lights on.

The strobe light system is supplied with electrical power through the circuit-breaker that follows:

– STROBE (28 V DC BUS #2).

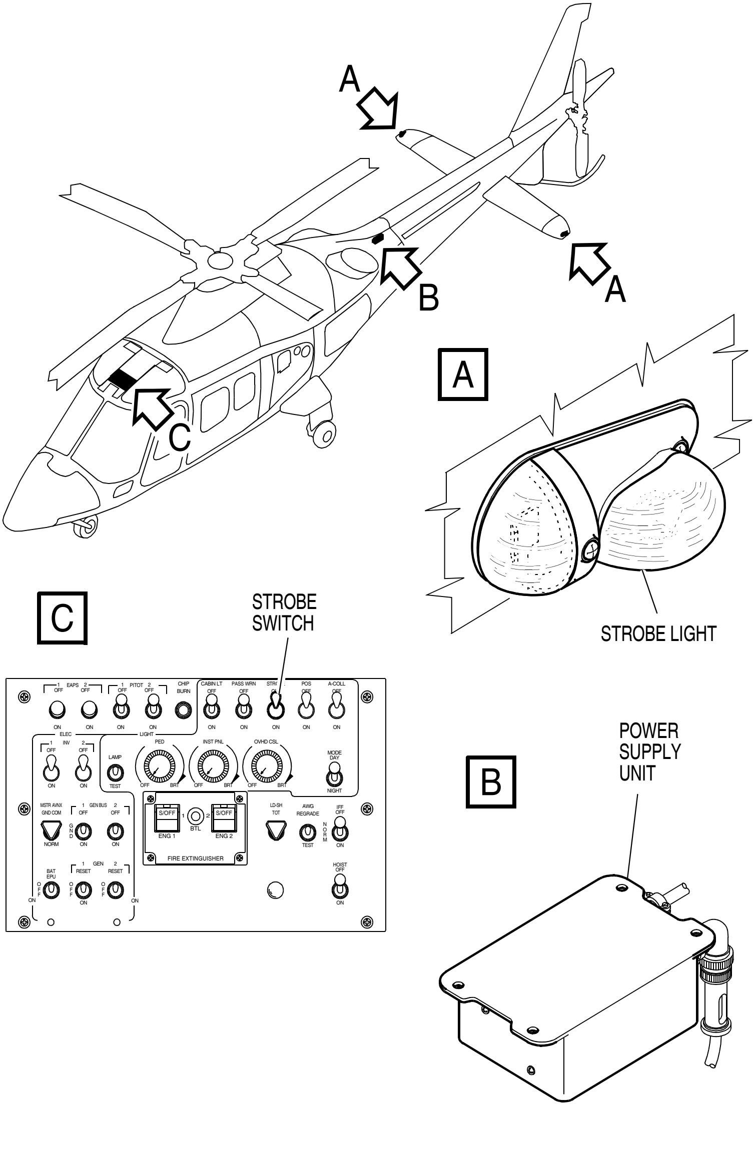

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Figure 7-98A Position of the Strobe Light System

SX-16 SEARCHLIGHT INSTALLATION

The SX-16 searchlight installation (Figure 7-98B) is a high intensity external light source installed on the right side of the helicopter fuselage. It provides further external lighting for reconnaissance operations during night time. The installation consists of the searchlight, the gimbal support, the support assembly, the control panel, the pendant control box and the necessary hardware and wires.

The searchlight is installed on a gimbal support that allows movements of the searchlight both in azimuth and elevation. When the searchlight is on, a cooling fan decreases the searchlight temperature.

The support assembly, composed of three support rods, holds the gimbal support with the search light. Theree quick release pins attach the three support rods to the fuselage structure.

The control box, on the interseat console, has the controls to switch on or off the search light.

The pendant control box is installed on its support in the aft part of the cabin. The operator can remove the control box from its support to control the searchlight from the cabin. With the control box, the operator can:

– Set the searchlight on or off
– Adjust the width of the light beam
– Point the light beam to the required direction.

The SX-16 searchlight installation is powered through the toggle-type circuit breaker that follows:

– SUN CONTR.

SX-16 SEARCHLIGHT SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-98C.

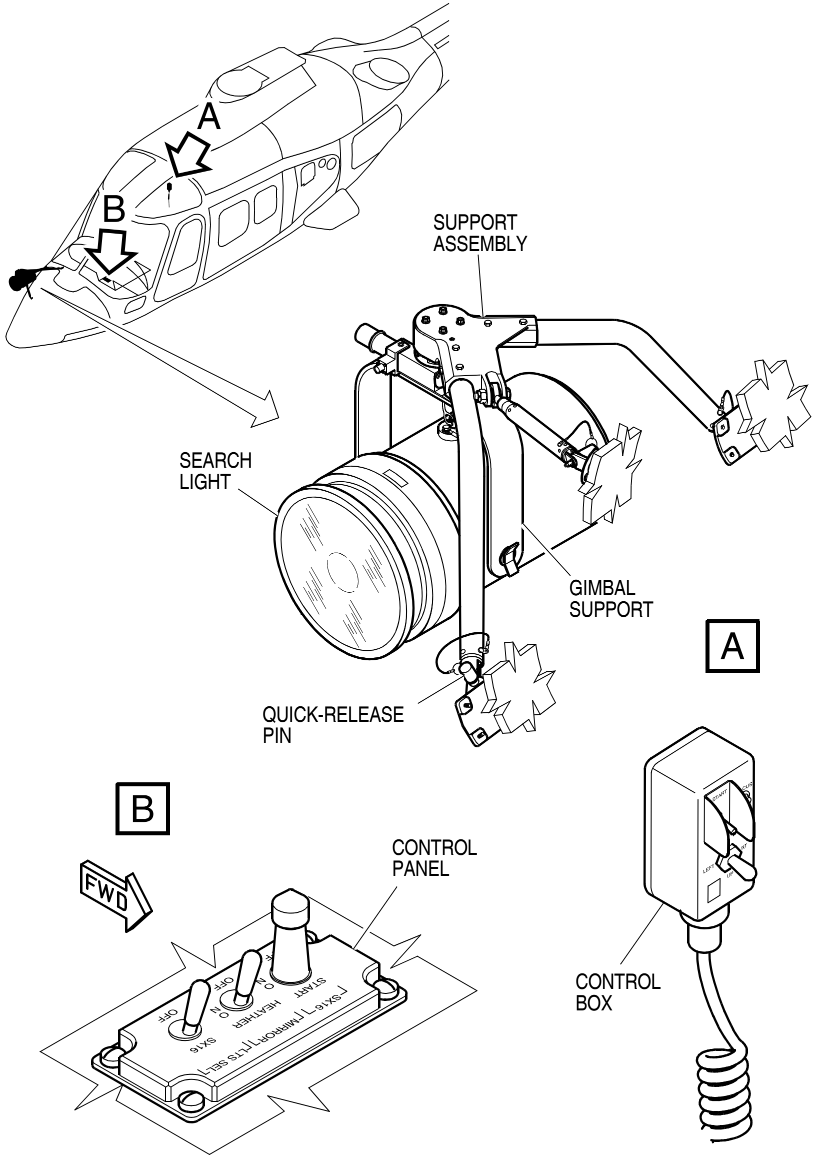

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Figure 7-98B SX-16 Searchlight Installation - Component Location

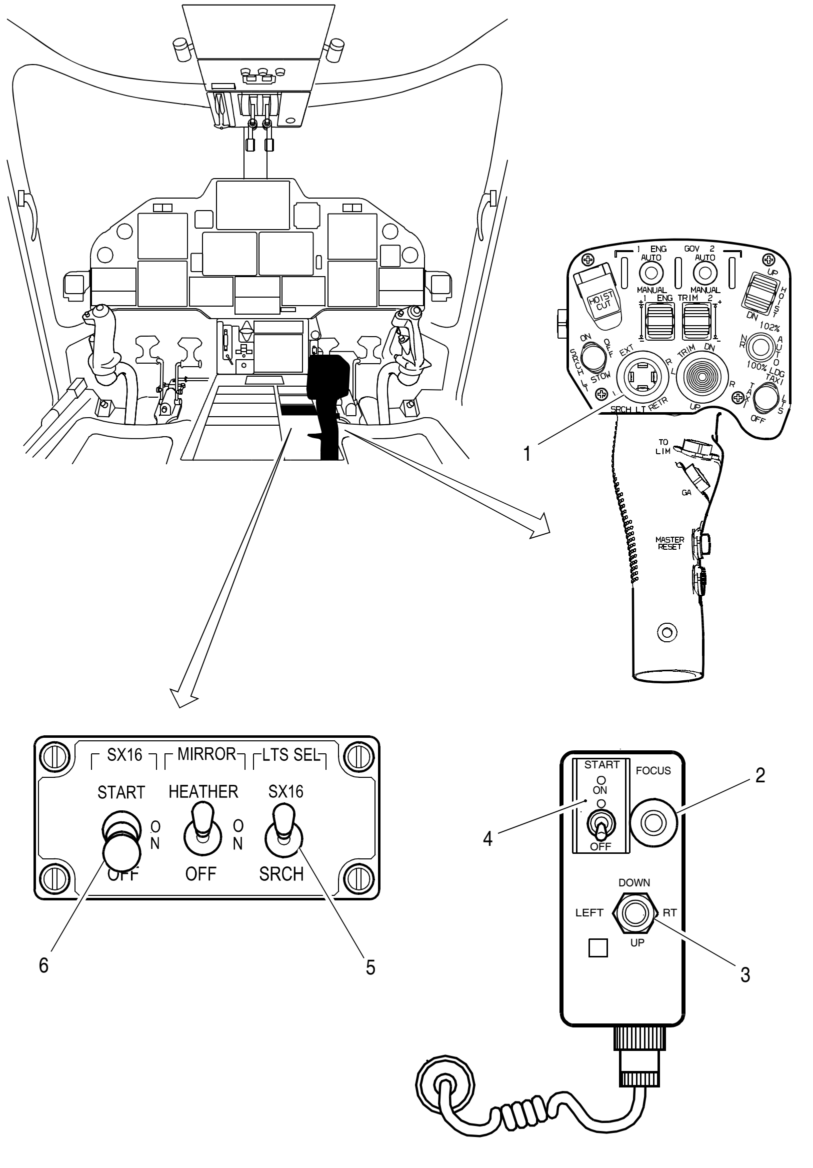

Figure 7-98C SX-16 Searchlight Installation - Controls and Displays

Key to Figure 7-98C

Ref.	Control/Display	Function
1	EXT/RETR/L/R slew switch	The 8-way directional slew switch moves the searchlight up, down, right or left. It operates only when the SX16 SEARCH switch is set to SWITCH.
2	FOCUS switch (two-position momentary contact toggle type)	Pushed - Operates the mechanism that continuously changes the width of the light beam from small diameter, high intensity to a larger diameter of less intensity. The light beam can be adjusted at any width between wide and narrow.
3	DOWN-UP-LEFT-RIGHT slew switch	The 8-way directional slew switch moves the searchlight up, down, right or left.
4	START-ON-OFF switch	START momentary position - Energizes the lamp starting circuit. Once the lamp starts, the switch must be released at the ON position. ON position - Energizes the lamp and the cooling fan for continuous operation. OFF position - De-energizes the system and extinguishes the light.
5	LTS SEL SX16/SRCH switch	SX16 - The EXT/RETR/L/R slew switch on the collective grip is enabled to control the SX-16 search light.
6	SX16 START/ON/OFF switch	START momentary position - Energizes the lamp starting circuit. Once the lamp starts, the switch must be released at the ON position. ON position - Energizes the lamp and the cooling fan for continuous operation. OFF position - De-energizes the system and extinguishes the light.

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NAVIGATION

(Chapter 34)

GENERAL

This system shows the pilots the information necessary for the navigation of the helicopter.

The navigation system includes:

– The flight environment data
– The attitude and direction system
– The landing aid system
– The independent position determining system
– The dependent position determining system.

FLIGHT ENVIRONMENT DATA

The flight environment data of the helicopter includes the Pitot-static pressure system and the air data system.

PITOT-STATIC PRESSURE SYSTEM

The Pitot-static pressure system (Figure 7-99 and Figure 7-100) provides the total and static pressures required for the operation of the Number 1 Air Data Unit (ADU1), the Number 2 Air Data Unit (ADU2) and the standby electronic indicator.

The system consists of two Pitot tubes, a static pressure selector valve, connecting pipes and nine drain plugs.

The two Pitot tubes are installed on the two sides of the helicopter, in the forward part, near the nose door.

Each Pitot tube is provided with a static port, an integral dynamic and static port and a heating element to prevent the formation of ice.

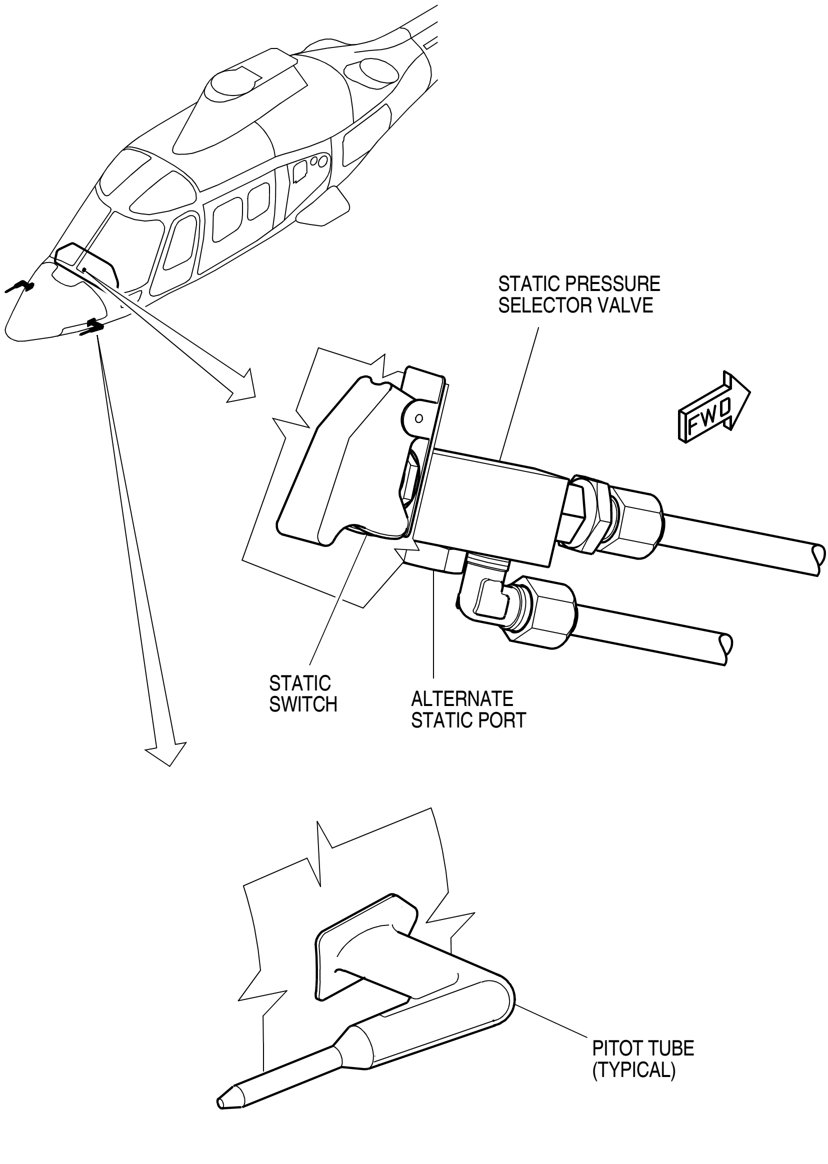

Figure 7-99 Pitot-Static Pressure System - Component Location

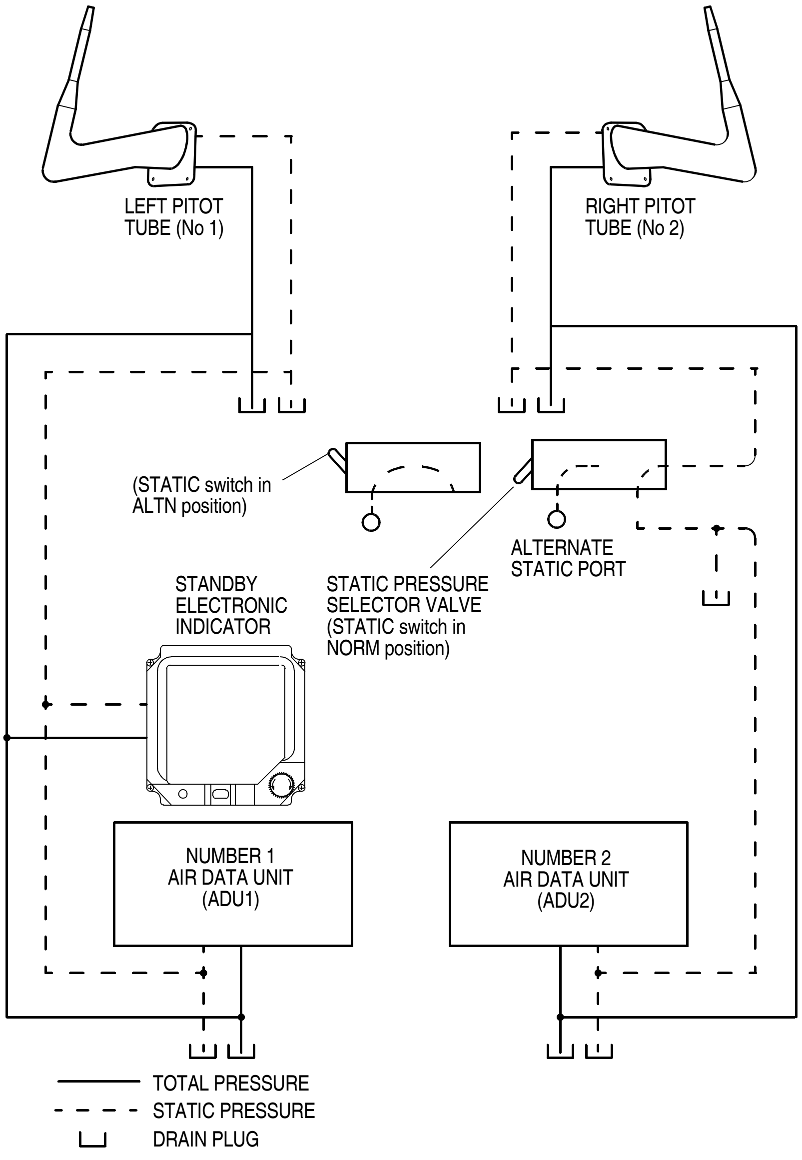

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Figure 7-100 Pitot-Static Pressure System - Schematic Diagram

The left Pitot tube (Number 1) supplies the total and static pressures to the ADU1 and the standby electronic indicator. The right Pitot tube (Number 2) supplies the total and static pressures to the ADU2.

The static pressure selector valve is installed at the center of the right side of the instrument panel.

The bottom part of the valve has a static port, alternate to the static pressure line of the right Pitot tube.

The front side of the valve shows the STATIC switch, provided with a guard.

In case of necessity, it is possible to select the alternate static port by moving the STATIC switch from NORM to ALTN.

PITOT-STATIC PRESSURE SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-101.

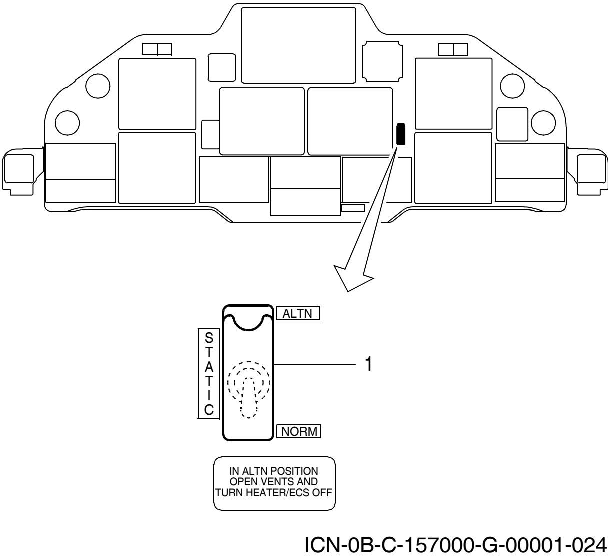

Figure 7-101 Pitot-Static Pressure System - Controls and Displays

Key to Figure 7-101

Ref.	Control/Display	Function
1	STATIC switch	NORM - The right Pitot tube is the source of static pressure for the ADU2. ALTN - The alternate static port in the cockpit is the source of static pressure for the ADU2.

AIR DATA SYSTEM

The air data system (Figure 7-102) calculates the air pressure and temperature values and provides these data, and those correlated to them, to the AFCS, the AHRS, the TAS and the EFIS.

The survey of the air pressure is made by the Pitot-static pressure system, while the survey of the air temperature is made by two Outside Air Temperature (OAT) sensors.

The system includes two OAT sensors, the Number 1 Air Data Unit (ADU1), the Number 2 Air Data Unit (ADU2) and the ADU selector on the RCP.

The two OAT sensors are installed on the top part of the helicopter. The Number 1 OAT sensor sends the temperature data to the ADU1, while the Number 2 OAT sensor sends the temperature data to the ADU2 and to the DAU.
The outside temperature value measured by the Number 2 OAT sensor is shown on the EDU2 in the MAIN format.

The ADU1 and the ADU 2 are installed in the aft avionic compartment.

When the ADU selector on the RCP, installed on the interseat console, is set to NORM, each ADU sends the altitude and airspeed data to the connected systems. When the ADU selector is set to 1 or 2, the selected ADU is the source of the air data.

Also the standby electronic indicator is an alternate and independent source of the air data, that are shown autonomously on its display.
The available data (corrected barometric altitude, indicated airspeed, vertical speed) are independent from the position of the ADU selector on the RCP and are always referred to the pressure value measured by the left Pitot tube.

The ADU1 is supplied with electrical system from the 28 V dc Number 1 auxiliary bus through the ADU 1 circuit breaker. The ADU2 is supplied with electrical system from the 28 V dc Number 2 essential bus through the ADU2 circuit breaker.

When an ADU is not energized or is failed, the ADC 1 or ADC 2 caution message is shown on the EDU1.

AIR DATA SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-103.

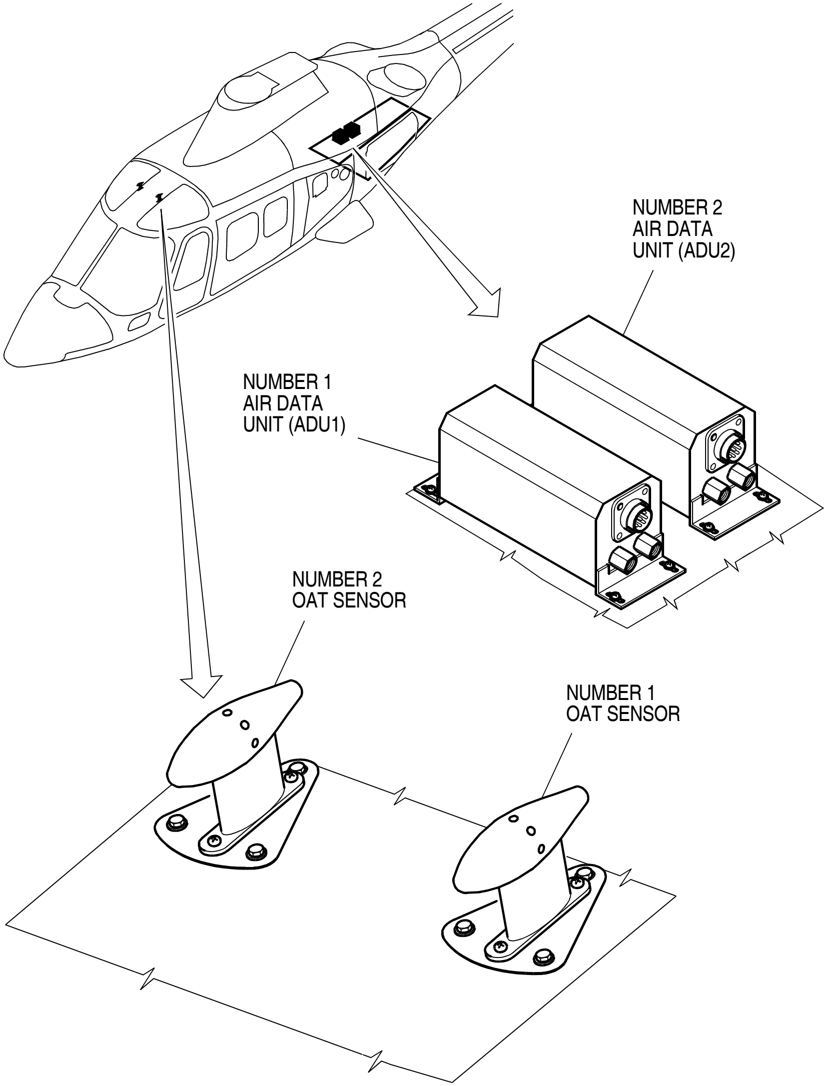

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Figure 7-102 Air Data System - Component Location

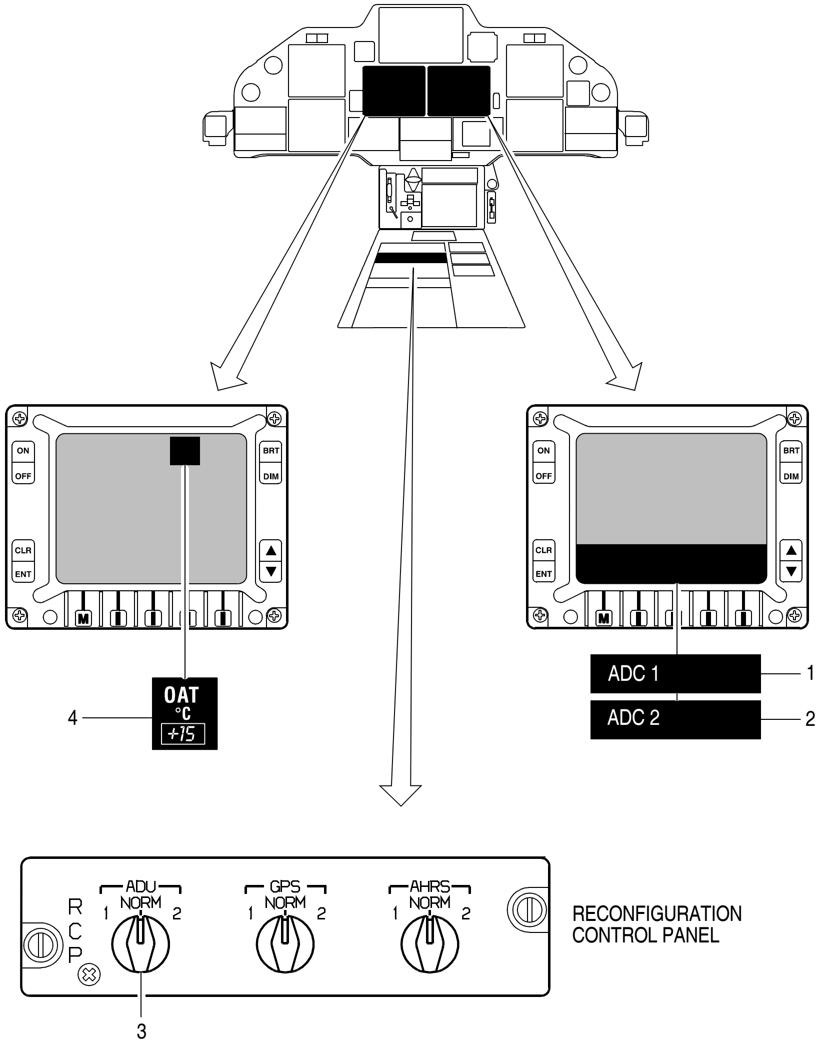

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Figure 7-103 Air Data System - Controls and Displays (Sheet 1 of 2)

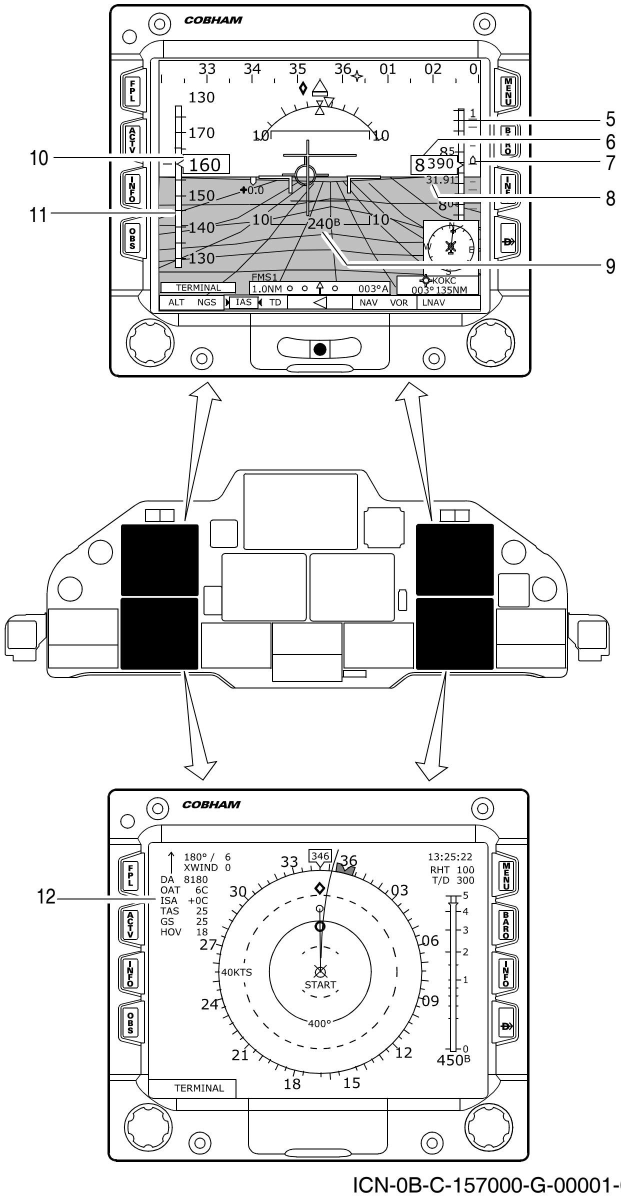

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Figure 7-103 Air Data System - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-103

Ref	Control/Display	Function
1	ADC 1 caution message	Shown when the ADU1 is not energized or is failed.
2	ADC 2 caution message	Shown when the ADU2 is not energized or is failed.
3	ADU selector	NORM - Each ADU provides the information to the related instruments: – The ADU1 to the copilot PFD and MFD. – The ADU2 to the pilot PFD and MFD. 1 - The ADU1 provides the air data to pilot and copilot PFDs and MFDs, while the ADU2 is excluded. 2 - The ADU2 provides the air data to pilot and copilot PFDs and MFDs, while the ADU1 is excluded.
4	OAT window	Shows the outside air temperature in °C.
5	Altitude scale	Shows the barometric altitude in feet MSL (Mean Sea Level). The part of the altitude tape that is at and below the ground level is of brown color. The tape has graduations in hundred-foot, but only the 500 ft increments have a label. A minimum of two labels can be seen at all times
6	Altitude readout	Shows digitally the barometric altitude in feet MSL (Mean Sea Level). The altitude is shown in ten-foot increments.
7	Vertical speed indicator	Shows the vertical speed of the helicopter. The vertical speed indicator points up when the helicopter climbs. The vertical speed indicator points down when the helicopter descents.
8	Setting value of barometric pressure	Is shown the setting value of the barometric pressure that can be set with the right control knob.

Key to Figure 7-103 (Continued)

Ref	Control/Display	Function
9	AGL altitude readout	Shows the AGL altitude value of the helicopter. The letter B after the altitude value shows that the value is a barometric altitude less the database ground elevation. It is used when the radio height and the GPS geodetic height are not correct.
10	Indicated airspeed scale	Shows the indicated airspeed value.
11	Indicated airspeed readout	Shows digitally the indicated airspeed in knots or miles per hour.
12	Wind, density altitude, air temperature and airspeed data	The air data system supplies inputs to the EFIS to let the calculation of the data shown on the top left corner of the MFDs..

ATTITUDE AND DIRECTION SYSTEM

The system, which gives information about the attitude of the helicopter and data for the magnetic navigation, includes:

– The Number 1 Attitude Heading Reference System (AHRS1)

– The Number 2 Attitude Heading Reference System (AHRS2)

– The standby electronic indicator.

ATTITUDE AND HEADING REFERENCE SYSTEMS (AHRS1 AND AHRS2)

The helicopter installs two Attitude Heading Reference Systems: AHRS1 and AHRS2 (Figure 7-104 and Figure 7-105).

Each system provides information about the magnetic heading and the attitude of the helicopter (pitch and roll angles).

Each AHRS comprises an Attitude and Heading Reference Unit (AHRU), an AHRU tray, a Magnetic Sensor Unit (MSU) and a calibration unit (CalPROM). The AHRU tray has a cooling fan, powered directly from the AHRU installed on the tray.

The system comprises also a control panel, which controls both AHRUs, and the AHRS selectors on the RCP, installed on the interseat console.

The AHRUs are installed in the aft avionic compartment. Each AHRU mainly includes three fiber optic rate gyros, three accelerometers, a digital computer, a power supply and the interface connectors.
The digital computer mathematically integrates the angular rates to obtain heading, pitch and roll data.

The AHRU trays are aligned to the helicopter axes when the trays are installed on the structure.
The misalignment can never exceed 3°.

The MSUs are installed on the bottom of the tail boom.
Each MSU detects the horizontal component of the earth's magnetic field and transmits it to the related AHRU, which uses the datum as a long term heading reference.

The calibration units (CalPROM) are installed on each AHRU.
Each calibration unit mainly consists of a EEPROM card which stores the helicopter specific information necessary for the compensation of the installation errors of the MSU, the possible misalignments of the AHRU tray and the distance on the three axes of the GPS antenna from the AHRU.
Thus the CalPROM relevant to AHRS1, identified with code 9, contains data different from those contained in the CalPROM relevant to the AHRS2, identified with code 10.
The CalPROMs are connected to the AHRU so that their position cannot be inverted if an AHRU is replaced for failure.

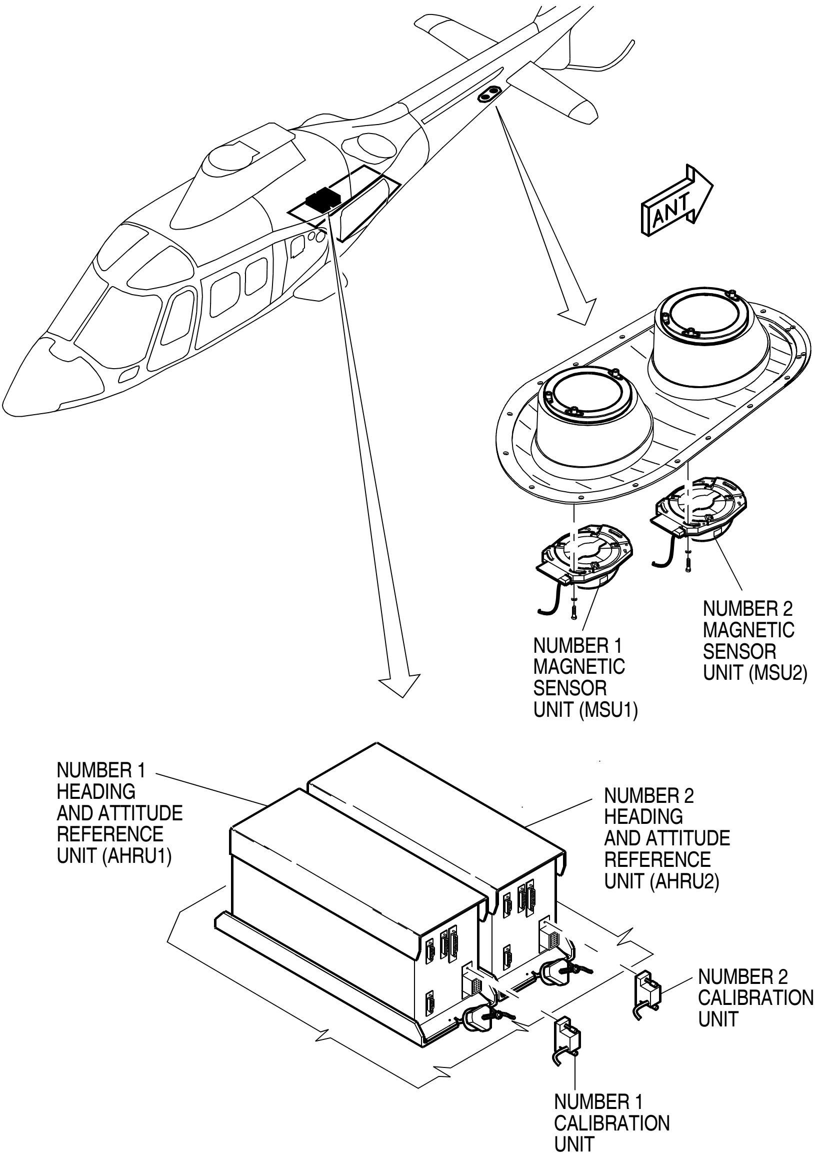

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Figure 7-104 AHRS System - Component Location

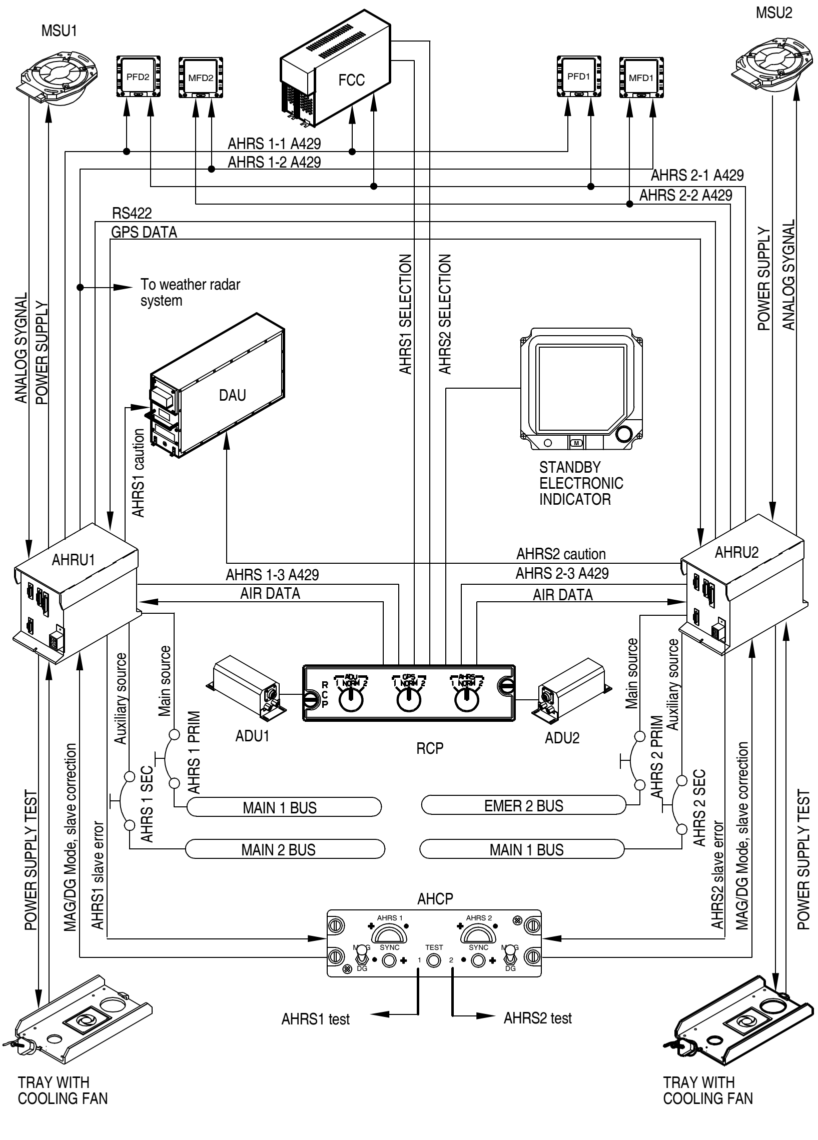

Figure 7-105 AHRS System - Schematic Diagram

The heading and attitude information are shown respectively on the PFD and MFD of the pilot and copilot.
The information validity and the possible differences between the data of the two systems are managed with appropriate messages that are shown on the PFD and MFD displays when necessary.

Each AHRS is interfaced with the air data system, from which it receives the airspeed and barometric pressure data, and with the FMS, from which it receives the position and the ground speed through the GPS receiver.
Each AHRU uses these data to increase the accuracy of the computed data and provides the results to the requiring systems as the EFIS (to show the attitude, heading and ground speed data), the AFCS, the FMS and the weather radar system.
The air data are also required by the AHRU to compute the vertical speed of the helicopter and to show it on the EADI display.

In the event of failure to an AHRS, the DAU causes the EDU1 to show the related AHRS 1 or AHRS 2 caution message.

The AHRS1 is supplied with electrical power from the 28 V dc Number 1 main bus through the AHRS 1 PRIM circuit breaker and from the 28 V dc Number 2 main bus through the AHRS 1 SEC circuit breaker.
The AHRS2 is supplied with electrical power from the 28 V dc Number 2 emergency bus through the AHRS 2 PRIM circuit breaker and from the 28 V dc Number 1 main bus through the AHRS 2 SEC circuit breaker.

Operation of the AHRS

Start Up Mode

As the AHRS is energized, the AHRU automatically initiates the start up mode and performs the actions that follows:

– Activates a start up advisory signal
– Activates a signal of No Computed Data (NDC) in attitude, heading, acceleration and rate data

– Performs the Power On Self Test to check the proper functionality of the AHRS
– Initializes and starts the inertial sensors.

If the ambient temperature is outside of the range of +15 °C to +35 °C, the gyro temperature stabilization phase is implemented by up to 25 seconds.

If the start up process identifies a power interruption with a duration less than 500 ms and if the helicopter motion does not exceed certain limits, a transition to the fast alignment phase takes place instantaneously.
On the contrary the system performs a normal alignment.

If the self-test has not detected the setting of one of the maintenance mode input discretes or any flight critical failures, the AHRU activates the gyro alignment phase.

Alignment Mode

The alignment phase includes the heading and attitude alignment.

For the heading alignment the AHRU uses the data from the MSU. If the helicopter is in flight, the heading alignment is performed during a straight and level flight.

For the attitude alignment (pitch and roll angles), the AHRU uses the data from the internal accelerometers.
If the helicopter is in motion, the AHRU uses the valid TAS to have an augmented accuracy. If TAS is not available, the attitude alignment is achieved during a straight and level flight only.
If the helicopter is not in motion, the attitude alignment is always performed assuming a reference TAS of 0 kt.

During a static alignment, the earth rate and the gyro drift are estimated in order to improve the system accuracy.
The estimation is performed independent of whether the AHRU is in DG or MAG mode.
The estimated value of the vertical earth rate is stored in a non-volatile memory and is used for the following power-on initialization of the estimation procedure.

Operational Mode

When the gyro alignment is completed, the AHRS continuously updates the angular rates, the accelerations, the heading and the attitude.

With regard to the heading information, the system can operate in the MAG or DG mode.
The system operates in the MAG mode when the MAG/DG switch on the AHRS control panel is set to MAG.
In this operational mode the heading is determined by the AHRU using the data from the MSU.
The information on the MFD is the magnetic heading referred to the local magnetic North.

If the MAG/DG switch is moved to DG, the system operates in the DG mode.
The information on the MFD corresponds to the bearing of the directional gyro. In this operational mode the desired heading can be selected by moving the alignment switch to left or right.
When the alignment switch is moved to right, the compass rose turns clockwise; when moved to left, the compass rose turns anti-clockwise.

With regard to the attitude information, the system can operate in the Normal mode or in the Basic mode.
The Normal mode is activated when the helicopter is in flight and a valid TAS from an ADU is available or when the helicopter is on ground and not in motion.
The TAS is used to improve the attitude information accuracy.

The Basic mode is activated when the helicopter is in flight and no valid TAS is available or when the helicopter is on ground and in motion.
In this operational mode the helicopter attitude is determined autonomously.

When the system operates in the MAG or Normal mode, the estimated earth rate and gyro drift are used for compensation.
The estimation is continuously determined to increase the system accuracy.
Besides, on the basis of the reception of valid data from the GPS receiver of the FMS, the system is able to provide the ground speed value which is shown on the MFDs.

The AHRS is able to meet any temporary loss of the GPS signal until the computed ground speed value is considered sufficiently accurate. However once this threshold is exceeded or anyway in case of prolonged or total loss of the GPS signal, the ground speed value is shown as not valid and not more used by the systems interfaced with the AHRS, which, when acceptable, uses the ground speed value provided by the FMS as backup.

Self-test Mode

The self-test of the AHRS1 or the AHRS2 can be activated by setting the TEST switch on the AHRS control panel respectively to 1 or 2.
The self-test is automatically disabled during the start up or the shut down and when the helicopter is in flight.
The self-test data, generated by the AHRU during the self-test, supersede the output data of the interfaced systems during the AHRU normal operation.

The AHRU operation is not influenced by the self-test data.

During the self-test the PFDs and the MFDs show these data:

– heading: ....15°
– pitch: ....5° down
– roll: ....45° to left
– vertical speed: ....600 ft/min.

At the end of the self-test the AHRS returns to normal operation after a delay of 1 s.

Shut Down Mode

The shut down of the AHRS is initiated automatically when a low voltage condition or a power interruption occurs.
For short interruptions of less than 5 ms, an internal power supply takes place and holds the system functionality unchanged.

If a power interruption causes the system shut down, the AHRU transits into an idle state until the power supply remains below a specified value.

If the input power increases above this value, the AHRU returns to operate again.

If the shut down is initiated by a failure, the AHRU automatically shuts down the power supply.
To restart the AHRS, the power supply must be switched off and on again.

AHRS CONTROLS AND DISPLAYS

Refer to Figure 7-106.

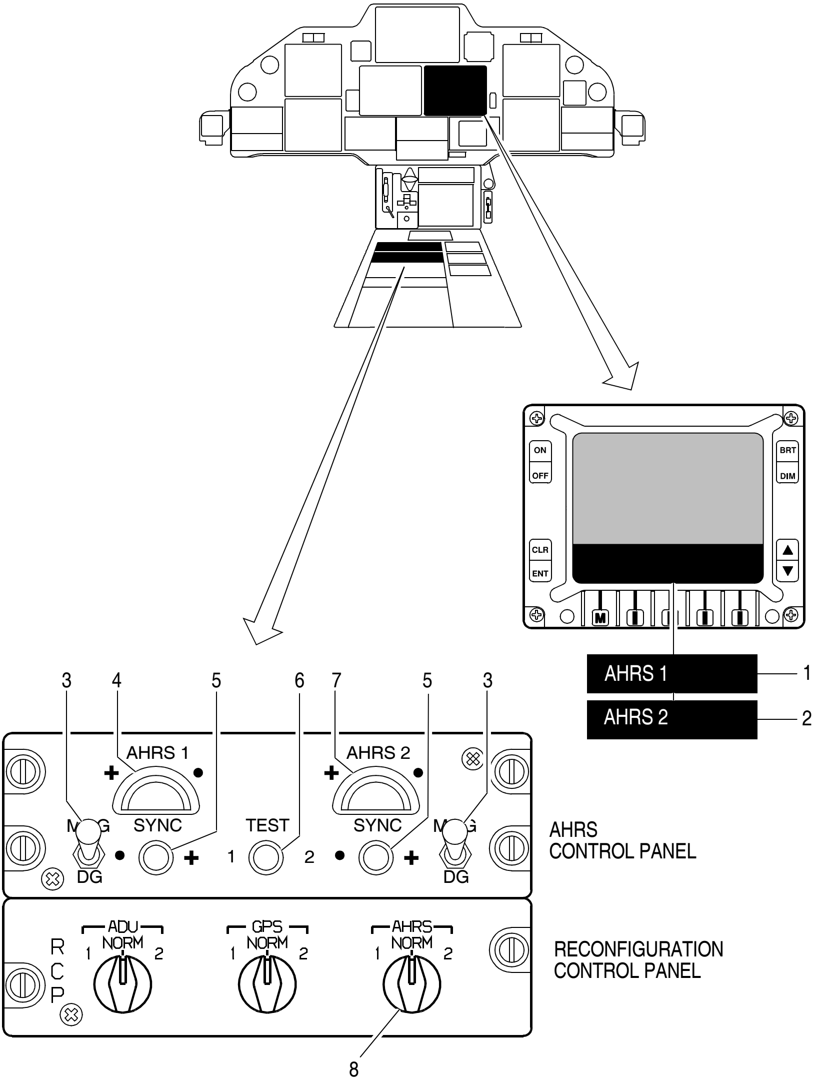

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Figure 7-106 AHRS System - Controls and Displays (Sheet 1 of 2)

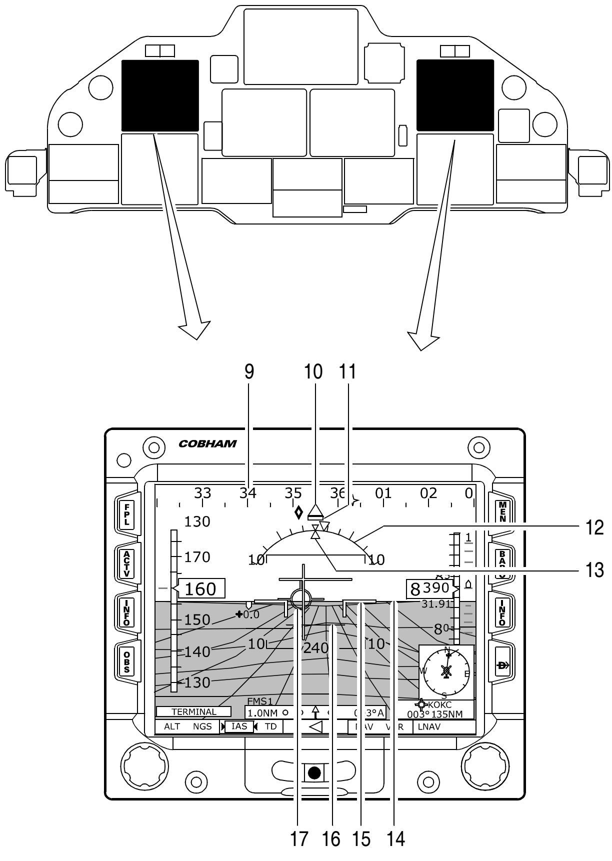

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Figure 7-106 AHRS System - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-106

Ref	Control/Display	Function
1	AHRS 1 caution message	Shown when a failure of rate and acceleration sensing components in the AHRU1 occurs.
2	AHRS 2 caution message	Shown when a failure of rate and acceleration sensing components in the AHRU2 occurs.
3	MAG/DG switch	MAG - The directional gyro of the AHRU is slaved to the related MSU. The compass rose of the EFIS instruments indicates the magnetic heading. DG - The directional gyro of the AHRU is not slaved to the MSU, but operates as a free gyro. The compass rose of the EFIS instruments indicates the bearing of the directional gyro.
4	AHRS 1 alignment indicator	In the MAG mode, when the needle is aligned with the fixed reference, the heading information provided by the compass rose of the EFIS instruments is consisting with the real direction of the helicopter. When the needle moves to (o) or (+) marks, the alignment indicator indicates a misalignment between the directional gyro output of the AHRU1 and the related MSU signal. The misalignment can be corrected by using the underlying alignment switch.
5	Alignment switch	In the MAG mode permits to align the directional gyro of the AHRU to the related MSU as indicated by the alignment indicator. In the DG mode permits to align the directional gyro of the AHRU to a desired reference. The alignment operation causes the compass rose to turn clockwise (-) or anti-clockwise (o).
6	TEST switch	1 or 2 - Starts the test procedures of the related AHRS.

Key to Figure 7-106 (Continued)

Ref	Control/Display	Function
7	AHRS 2 alignment indicator	In the MAG mode, when the needle is aligned with the fixed reference, the heading information provided by the compass rose of the EFIS instruments is consisting with the real direction of the helicopter. When the needle moves to (o) or (+) marks, the alignment indicator indicates a misalignment between the directional gyro output of the AHRU2 and the related MSU signal. The misalignment can be corrected by using the underlying alignment switch.
8	AHRS selector	NORM - Both AHRUs are connected to the PFDs, the MFDs and the FCC. The pilot PFD and MFD show the AHRU2 information, the copilot PFD and MFD show the AHRU1 information, the FCC takes account of both AHRU. Besides: – The standby electronic indicator is connected to the AHRS1 –The weather radar is connected to AHRS1. 1 - Both AHRUs are connected to the PFDs, the MFDs and the FCC. The PFDs and MFDs of pilot and copilot show the information of the AHRS1. The FCC, the weather radar and the standby electronic indicator uses the data from the AHRS1. 2 - Both AHRUs are connected to the PFDs, the MFDs and the FCC. The PFDs and MFDs of pilot and copilot show the information of the AHRS2. The FCC, the weather radar and the standby electronic indicator uses the data from the AHRS2.

Key to Figure 7-106 (Continued)

Ref	Control/Display	Function
9	Directional scale	Shows the heading value of the helicopter. In standard mode, the directional scale is 70° wide with graduations at 5° increments and figures at 10° increments. In zoom mode, the directional scale is 35° wide with graduations at 5° increments.
10	Heading indicator	Shows the current heading of the helicopter on the directional scale.
11	Slip indicator	Moves left or right to show the lateral acceleration that the AHRS senses.
12	Bank angle scale	Lets the roll pointer show the bank angle of the helicopter.
13	Roll pointer	Shows the bank angle of the helicopter on the bank angle scale.
14	Horizon line	Divides the background of the attitude display area in two sectors: - the area above the horizon line is blue and represents the sky (positive pitch) - the area below the horizon line is brown and represents the ground (negative pitch).
15	Waterline	Shows where the helicopter is pointed.
16	Pitch scale	Shows the pitch angle of the helicopter. The pitch scale turns together with the horizon line. The pitch scale has increments of 5°. At 10° increments the line is larger with the scale label.
17	Flight path marker	Shows the actual flight path of the helicopter on the outside world. The flight path marker is shown laterally from the waterline to show the difference between the aircraft track and the heading. The flight path marker is shown vertically from the waterline to show the aircraft climb or descent angle.

STANDBY ELECTRONIC INDICATOR

The standby electronic indicator, installed on the top central part of the instrument panel, permits to show all the primary flight and navigation information in the event of partial or total loss of the instrument panel displays.

The instrument provides indications of:

– attitude
– heading
– coordinated turn
– altitude
– airspeed
– navigation (VOR, ILS, Flight Plan)
– failure.

The attitude indications are generated by the same instrument by means of a self-contained three-axis inertial system.

The heading indications are provided by the AHRS1 through the RCP when the AHRS selector is set to 1 or NORM or by the AHRS2 when the AHRS selector is set to 2.

For the altitude and airspeed indications the rear part of the instrument is connected to the left Pitot tube.
The navigation information are provided by the VOR1 and VOR2 receivers and the FMS in the EFIS instruments.

The standby electronic indicator is provided in its rear part with a Detachable Configuration Module (DCM).
The DCM contains a non-volatile memory which permits to store the configuration data of the instrument as the codes of the used hardware and software.
In case of instrument replacement the DCM can remain on the helicopter and be connected to the new instrument.

The standby electronic instrument is supplied with electrical power from the 28 V dc battery bus through the STBY IND circuit breaker.

Operation of the Standby Electronic Indicator

When the standby electronic indicator is supplied with electrical power, the self-tests of the instrument automatically start before the

normal operation.
During the self-tests, for the first 15 seconds, the display shows no information and its background is blank.

When the self-tests are completed, the display shows an identification screen containing the information that follows (Figure 7-107):

– The aircraft type for which the instrument has been configured
– The total number of operating hours
– The installed software identification number.

If the self-tests detect a failure, the identification screen also contains the failure message or code. If the failure affects the whole instrument, the power on sequence interrupts; if the failure affects one only or more functions, the power-on sequence will be completed, but the instrument will operate without the failed functions.
The information about the invalid functions are shown on the display.

If no failures are detected, the display shows the ATT FAIL message followed by the ALIGNING message for a duration of 3 minutes.
During this phase any external abnormal condition such as the helicopter motion will make the alignment time double.

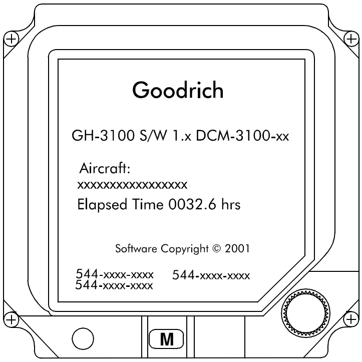
If the alignment phase should fail, the message will change to ALIGNMENT FAIL and the instrument will not start the normal operational mode.

During the normal operational mode the standby electronic indicator continues to perform the diagnostic self-tests to make sure that the information is accurate.

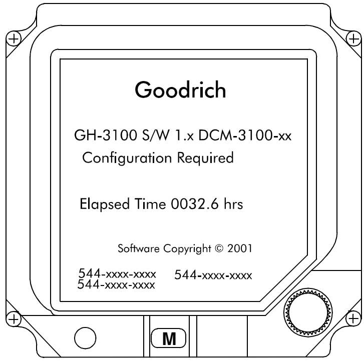
If there is a permanent loss of a valid signal to the instrument, the related indications are removed and the IAS, ALT or HDG messages are shown.
If the loss of signal is temporary, also the loss of indication is temporary.

In case of loss of attitude information, the ATT FAIL message is shown at the center of the display and the attitude indication disappears.

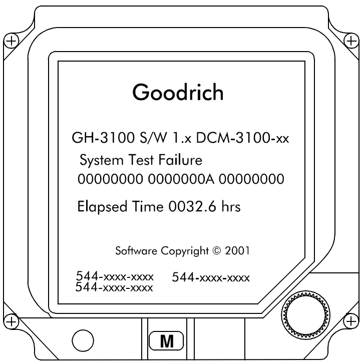
If the diagnostic self-tests detect an internal not rectifiable condition, the "WARNING - Power Reset Needed" message is automatically shown on the display.
The reset operation of the power supply must be performed only in straight and level flight and without accelerations.



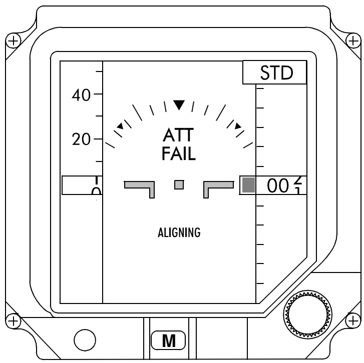
DATA VISUALIZATION ON
COMPLETION OF SELF-TESTS
WITH NO DETECTED FAILURES



DATA VISUALIZATION ON
COMPLETION OF SELF-TESTS
WITH FAILURE MESSAGE



DATA VISUALIZATION ON
COMPLETION OF SELF-TESTS
WITH NO FAILURE CODE



DATA VISUALIZATION IN
THE ALIGNMENT PHASE

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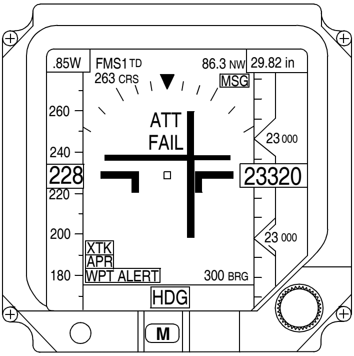
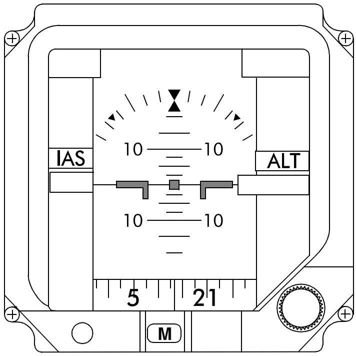
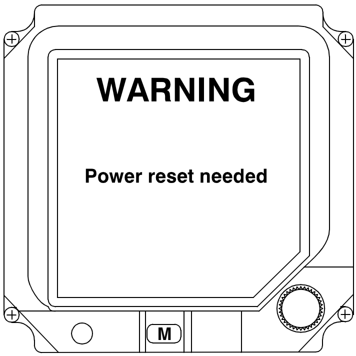
Figure 7-107 Operation of the Standby Electronic Indicator (Sheet 1 of 2)



ATTITUDE AND HEADING
INDICATIONS INVALID



AIRSPEED AND ALTITUDE
INDICATIONS INVALID



NEEDED POWER RESET
INDICATIONS

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Figure 7-107 Operation of the Standby Electronic Indicator (Sheet 2 of 2)

Data presentation

When the alignment phase is completed, the standby electronic indicator provides the flight and navigation information according to the selected presentation mode.
This selection can be done by pushing the M push-button on the bottom of the instrument bezel.

The bottom of the display shows a maximum of four items of the presentation mode list. By turning the knob in the bottom left corner of the instrument, it is possible to scroll the menu item list.
The " < " symbol at the beginning of the menu item indicates the active item, highlighted also by a greater brightness.
Some menu items are distinguished by two conditions (ON and OFF), of which the not active condition is highlighted.
Other menu items are followed by **...** to indicate the presence of a submenu, which is shown in the bottom of the display when the adjustment knob is pushed.

The presentation mode list is the following:

– FAST ERECT	Push the adjustment knob to start the fast gyro alignment.
– SET BRIGHTNESS OFFSET	Push the adjustment knob to select the submenu, turn the knob to adjust the brightness, then push the knob to exit the submenu.
– FAST ALIGN	Push the adjustment knob to cause: - the fast alignment of the gyro vertical axis to the current vertical axis of the helicopter - a 90-second sensor alignment.
– SET HEADING	Push the adjustment knob to select the submenu, turn the knob to select the heading, push the knob to exit the submenu.
– NAV (ON or OFF)	Push the adjustment knob to set the opposite of current condition.
– NAV MODE	Push the adjustment knob to select the submenu, turn the knob to select the navigation mode, push the knob to exit the submenu.
– SET CRS	Push the adjustment knob to select the submenu, turn the knob to select the course, push the knob to exit the submenu.

–	ILS (BC/NORMAL)	Push the adjustment knob to set the opposite of current condition.
–	CRS AUTO CENTER	Push the adjustment knob to start the operation.
–	NAV DISPLAYS	Push the adjustment knob to select the submenu, turn the knob to select, push the knob to exit the submenu.
–	BAROMETRIC TYPE	Push the adjustment knob to select the submenu, turn the knob to select the measurement unit (inHg, hPa, mbar), push the knob to exit the submenu.
–	IAS TAPE DIRECTION (UP or DOWN)	Push the adjustment knob to set the opposite of current condition.

By pushing the M push-button again or after 15 to 20 seconds are elapsed without performing any operation on the menu, the menu items disappear.

STANDBY ELECTRONIC INDICATOR CONTROLS AND DISPLAYS

Refer to Figure 7-108.

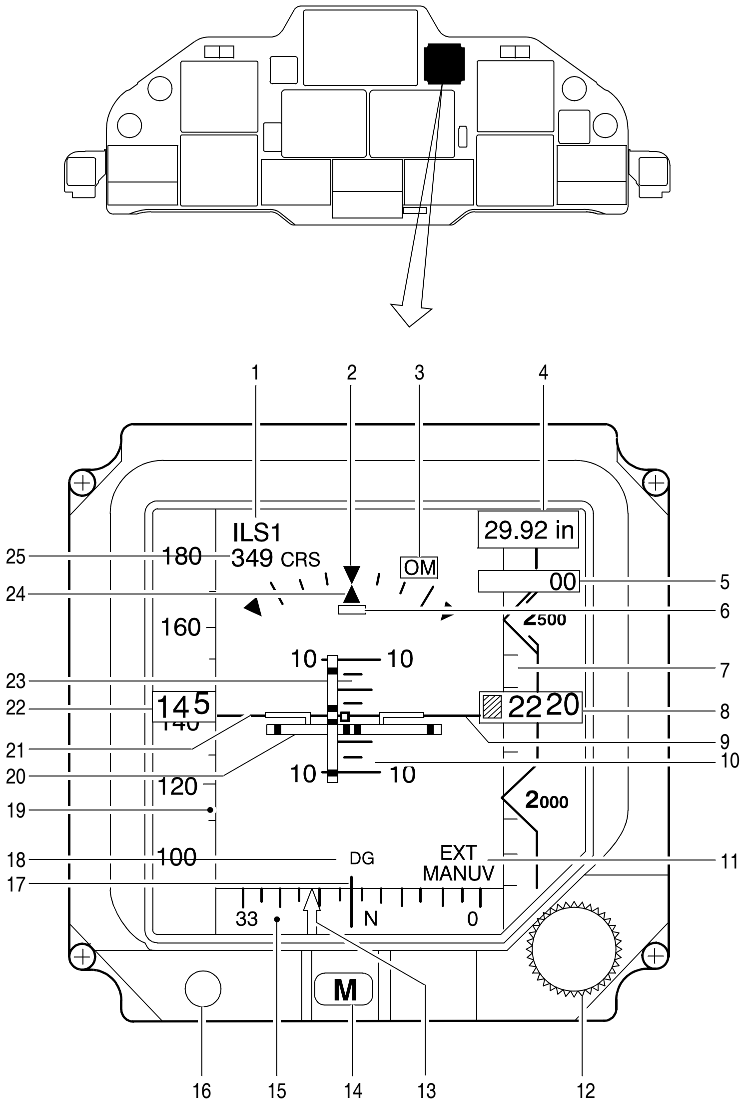

Figure 7-108 Standby Electronic Indicator - Controls and Displays

Key to Figure 7-108

Ref	Control/Display	Function
1	Navigation mode indicator	Shows the presentation mode of navigation data on the display.
2	Roll scale	Leis the roll pointer show the roll angle of the helicopter.
3	Marker beacon	Shows when the helicopter flights over a marker beacon station. The station is shown with the letters: - OM for the outer beacon (Outer Marker) - MM for the middle beacon (Middle Marker) - IM for the inner beacon (Inner Marker).
4	Barometric correction readout	Shows the barometric correction value set with the adjustment knob.
5	Vertical speed digital readout	Shows the vertical speed of the helicopter.
6	Coordinated turn indicator	Shows when the turn maneuver of the helicopter is correctly coordinated. The maneuver is coordinated when the indicator follows the roll pointer. In case of slip or skid, the indicator moves laterally from the roll pointer
7	Altitude scale	Is of the tape type which scrolls vertically. It is capable of showing altitudes from -1,000 to +55,000 ft. The scale has white marks every 100 foot increments except at the 500 foot marks where the chevrons are used and the altitude values are shown. If the altitude signal is invalid, a red box with the ALT legend is shown instead of the scale.
8	Altitude digital readout	At the center of the altitude scale, shows the altitude value that the helicopter has reached. Up to five large size digits are used.

Key to Figure 7-108 (Continued)

Ref	Control/Display	Function
9	Horizon line	Divides the background of the attitude display area in two parts: – the area above the horizon line is blue and represents the sky (positive pitch), – the area below the horizon line is brown and represents the ground (negative pitch).
10	Pitch scale	Lets the aircraft symbol to show the pitch angle of the helicopter. The longest scale lines represent 10° increments in pitch. Each of these lines has a white number at both ends. The scale lines midway between the 10° lines are shorter than the 10° lines and represent the 5° increments.
11	EXT MANUV message	Shown when one of the following conditions occurs: – The helicopter is flying with a roll angle greater than 7° for a duration exceeding 6 min. – The helicopter heading is not inside ±8° compared with the determined magnetic heading.

Key to Figure 7-108 (Continued)

Ref	Control/Display	Function
12	Adjustment knob	Turned when the menu mode is on - Permits to move the highlighting of an item in the menu list. When turned clockwise the highlighting jumps up, while when turned counter-clockwise the highlighting jumps down. When the highlighted item is at the top of the list, if the knob is turned again clockwise, the bottom menu item is highlighted. When the highlighting item is at the bottom of the list, if the knob is turned again counter-clockwise, the top menu item is highlighted. Turned when the menu mode is off - Permits to adjust the barometric correction value. Pushed - Permits to access the submenu of the set menu item or enter the changed parameter in the submenu or the barometric correction value.
13	Set course arrow	Shown on the heading scale in correspondence of the selected course. The arrow is upwards when the helicopter is flying in the direction of the selected course. The arrow is downwards when the helicopter is flying in the direction opposite the selected course.
14	M push-button	Pushed the first time - The push-button light comes on and the bottom part of the display shows four items of the menu list. The selected menu item is highlighted with white letters and the " > " symbol at the beginning of the line. Pushed a second time - The menu disappears.

Key to Figure 7-108 (Continued)

Ref	Control/Display	Function
15	Heading scale	Is of the tape type which slides horizontally to left or right. The display shows a scale span of 65°. The scale has white marks every 5°. The marks every 10° are twice as long as those at 5° except where the heading numbers exists. The cardinal headings are indicated with the letters N, E, S and W instead of the heading numbers. If the altitude signal is invalid, a red box with the HDG legend is shown instead of the scale.
16	Light sensor	Senses the ambient light and adjusts the brightness of the display.
17	Index line	Permits the reading of the magnetic heading of the helicopter.
18	DG message	Shown when the external heading signal is invalid.
19	Airspeed scale	Is of the tape type which scrolls vertically. It is capable of showing airspeeds between 0 and 450 kt. The display shows a scale span of 10 kt. The scale has white marks at 10 kt increments, while the airspeed values are every 20 kt. If the airspeed signal is invalid, a red box with the IAS legend is shown instead of the scale.
20	Up / down deviation indicator	Moves up or down to show the up or down direction from the navigation source. In ILS mode the indicator shows the up or down deviation from the glideslope.
21	Aircraft symbol	Provides the indication of the helicopter position with respect to the horizon line.

Key to Figure 7-108 (Continued)

Ref	Control/Display	Function
22	Airspeed digital readout	Shown at the center of the airspeed scale, indicates the IAS that the helicopter has reached. Up to three large size digits are used.
23	Left / right deviation indicator	Moves left or right to show the left or right deviation from the navigation source. In ILS mode the indicator shows the left or right deviation from the localizer.
24	Roll pointer	Shows the roll angle of the helicopter on the roll scale.
25	Set course digital readout	Shows the set course in the VOR or ILS modes.

LANDING AID SYSTEM

The system provides guidance during approach and landing of the helicopter.

The landing aid system includes the radio altimeter system.

RADIO-ALTIMETER SYSTEM

The radio-altimeter system consists of a transceiver Rockwell Collins type ALT-4000, installed in the aft avionic compartment, two antennas, one for the transmission and the other for the reception, installed below the tail boom at the centerline and the relay K119.

The system provides instantaneous indication of the height of the helicopter above the terrain between 0 and 2500 ft.

The transceiver is interfaced with the EFIS and the FCC through two ARINC 429 busses for the transmission of the measured height and the possible malfunctions.
In the event of malfunctions the proper messages are shown on the EFIS instruments instead of the indication of radio-altimeter height.

The transceiver is also interfaced with the central warning and caution system and the IDS, to which it sends a signal when the measured height is lower than 200 ft.

The system is supplied with electrical power from the 28 V dc Number 2 emergency bus through the RAD ALT circuit breaker.

Operation of the radio-altimeter system

The radio-altimeter system produces, radiates and detects microwaves at an operational frequency of 4300 ±15 MHz with a frequency modulation of 50 Hz.

The transceiver produces an output signal that is applied to the transmit antenna.
The transmit antenna beams the signal to the terrain that reflects the signal back to the receive antenna.
The reflected signal is then applied to the radio-altimeter transceiver.
Because the elapsed time between the transmission and the reception is proportional to the helicopter height above the terrain, the transceiver measures the time and converts the time to a height signal that is sent to the EFIS instruments and the FCC.

Being the transmitted signal continuous, this permits a constant check of the helicopter height.

The helicopter height is shown on the PFDs at the center of the display, immediately below the aircraft mock-up in the 0000 R format. The helicopter height is also shown on the MFDs only in the Hover page through a green scale on the right side of the display. The scale is from 0 to 500 ft when the helicopter is below 500 ft. The scale is from 0 to 2500 ft when the helicopter is above 500 ft. Below the radio height scale is there is the radio height readout. The helicopter height is not shown above 2500 ft or below 20 ft or when the signal is invalid.

When the measured height is lower than 200 ft, the transceiver sends a signal to the AWG and the IDS.

The AWG generates the TWO HUNDRED FEET aural message and, if the landing gear is retracted, the LANDING GEAR aural message, that are sent to the pilot headsets.

The IDS causes the EDU1 to show the LANDING GEAR caution message.

When the landing gear is extended, the K119 relay causes the exclusion of the LANDING GEAR aural message and the analogous caution message on the EDU1.

The TWO HUNDRED FEET aural message can be excluded by setting the AWG switch on the overhead console to REGRADE.

RADIO-ALTIMETER SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-109.

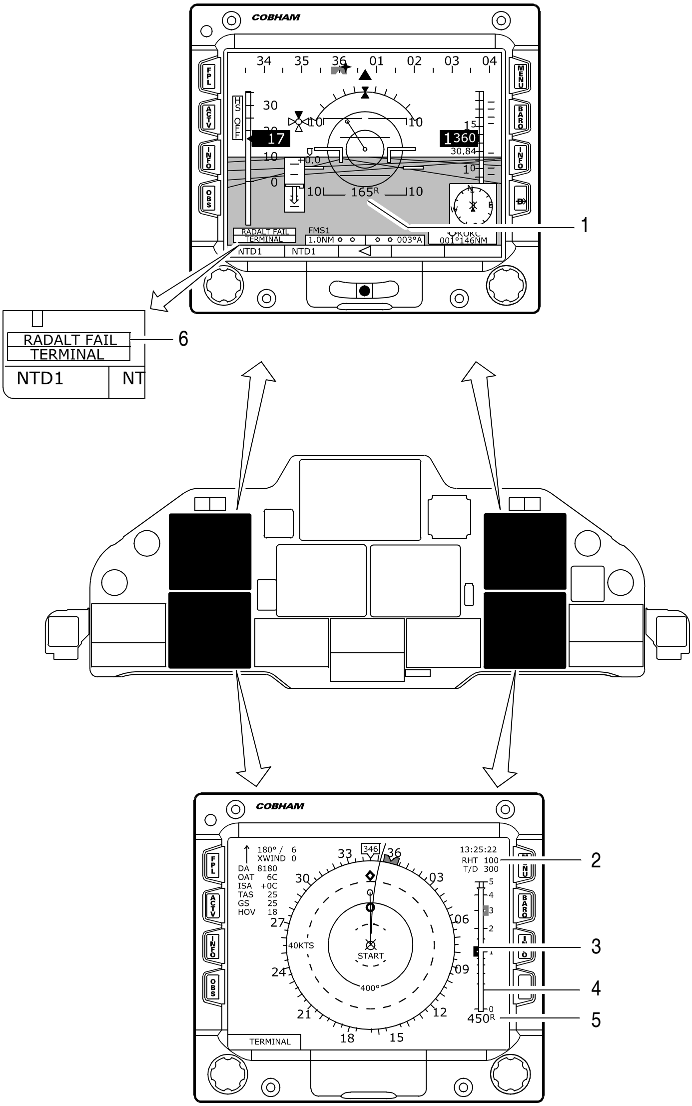

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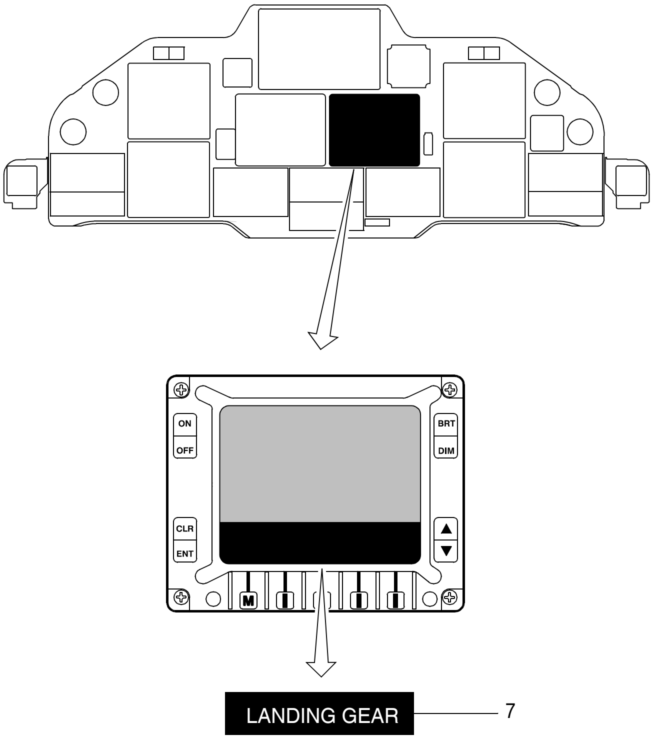
Figure 7-109 Radio Altimeter System - Controls and Displays (Sheet 1 of 2)



ICN-0B-C-157000-G-00001-02489-A-01-1

Figure 7-109 Radio Altimeter System - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-109

Ref	Control/Display	Function
1	Radio height indicator	Shows the AGL altitude of the helicopter on the PFD. The indicator is of four-digit type. The letter R after the height indicator shows that it is the radio altimeter which moves the altitude indication. The indicator is not shown when the measured height is more than 2500 ft.
2	Reference radio-height readout	Shows the value of the reference height set for the hovering of the helicopter.
3	Reference radio-height bug	Shows the value of the height reference on the height scale
4	Radio height scale	Shows the AGL altitude of the helicopter on the MFD. The altitude is shown in green. The top of the scale is 500 ft AGL. The scale is logarithmic to show smaller resolutions at lower altitude.
5	Radio height readout	Shows the AGL altitude value of the helicopter that is shown on the scale. The letter R after the altitude value shows that is the radio altimeter which moves the altitude indication.
6	RADALT FAIL caution message	Shown when the helicopter is in flight below 2500 ft and the radio altimeter indication is invalid.
7	LANDING GEAR caution message	Shown when the helicopter is at an altitude of less than 200 ft and the landing gear is not extended.

INDEPENDENT POSITION DETERMINING SYSTEM

The system provides information to determine the position of the helicopter independently from ground installations or orbital satellites.

The independent position determining system includes the traffic advisory system.

TRAFFIC ADVISORY SYSTEM

The traffic advisory system Avidyne type TAS620 is used to identify the altitude, range and bearing of nearby transponder-equipped aircraft. Aircraft equipped with non-Mode C transponders can provide only range and bearing information.
The system consists of a processor and a mounting tray, installed in the aft avionic compartment, a top antenna, installed above the cockpit, and a bottom antenna, installed below the cockpit.
The processor can track up to 50 intruding aircraft within a horizontal radius of 21 NM and a vertical height of +/-9900 ft, but show only up to 9 targets simultaneously.
The air traffic information are displayed on the EFIS instruments.
The system is interfaced with the ADU1 and ADU2.
A suppression signal is sent to the DME and the ATC transponder when the traffic advisory system operates.
The system is supplied with electrical power from the 28 V dc Number 2 main bus through the TAS circuit breaker.

Operation of the traffic advisory system

The operation of the traffic advisory system is based on the communications through the directional antennae between the processor and the transponders installed on aircraft in the airspace around the helicopter. The processor generates and transmits interrogations to the aircraft and computes the replies to obtain altitude, range, bearing and approach speed of the detected aircraft.
Relative altitude information is derived from decoding the altitude replies from nearby aircraft, and comparing the data with the encoded altitude information from the helicopter.
Range information is calculated using time of arrival techniques.
Bearing information is calculated using the directional antennae located on the top and bottom of the helicopter.
The altitude data from the intruding aircraft is referenced to the same

standard pressure altitude of the onboard encoder, thus separation is not depending on the altitude setting.
The positions and the altitude of tracked aircraft are shown on the PFD and the MFD with specific symbology. The PFD shows an additional thumbnail on the lower right corner with traffic information.
The MFD has a dedicated page called TRAFFIC page which the flight crew can access to have a more complete awareness of the traffic going on all the around the helicopter.
The system provides both visual traffic advisory and an aural message when it predicts that an aircraft may present a collision threat.

These different advisory levels can be displayed to the flight crew:

– Traffic Advisories (TA) that are also audibly announced
– Proximates Advisories (PA) that are related to the traffic within parameters of the display defined by the flight crew
– Other Traffic (OT) that are related to intruders that are not TAs or PAs.

The aural messages are sent directly from the EFIS to the ICS. The aural message is a tone followed by the TRAFFIC, TRAFFIC voice message. The TRAFFIC caution message is shown on the PFD and the MFD together with the TRAFFIC, TRAFFIC voice message.

The Traffic Advisory (TA) is generated when the system detects that the current track of an intruding aircraft could result in a near miss or collision. The system uses tau (the time to closest approach) and the time to reach the same altitude to calculate TAs. An intruding aircraft that is very close but not closing can also generate a TA. Only TAs generates audible warnings.
A TA is generated and an initial TA announcement is issued when the tau value of an intruding aircraft is less than the “TA threshold”. A TA is also generated when the range and the altitude separation are both less than the “TA threshold”. A TA remains in effect until the range between the helicopter and the intruding aircraft begins to diverge or is no longer detected for 8 s, whichever is longer.

Table TA Threshold

INTRUDER TYPE	HELICOPTER TO INTRUDER		
	TAU (SECONDS)	RANGE (NM)	ALTITUDE SEPARATION
Altitude reporting intruders	< 30	< 0.55	< 800
Non-altitude reporting intruders	< 25	< 0.20	N/A

The intruding aircraft are always shown, but the TRAFFIC caution message and the aural message are not given when:

– The AGL altitude of the helicopter is less than 400 ft
– The intruding aircraft is below 200 ft AGL.

Distant aircraft at low altitudes may be partially obscured by terrain.
When an ADU that supplies the altitude data to the processor has a failure, the related indication is shown on the EFIS instruments.
The AURAL WARN / TAS switch on the miscellaneous control panel permits the flight crew to inhibit the aural annunciations and the relative flags, but the detected traffic continues to be displayed either on the PFD and the MFD. When the AURAL WARN / TAS switch is set to OFF, the TAS INHBT advisory message is shown on the MFD.

TRAFFIC ADVISORY SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-110.

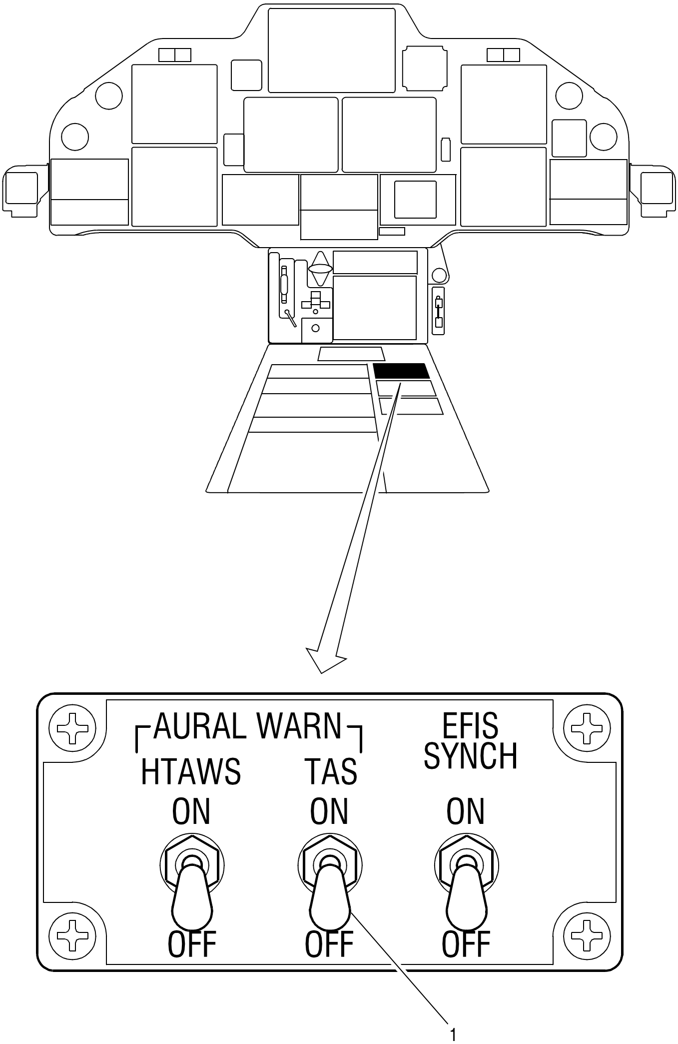


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Figure 7-110 Traffic Advisory System - Controls and Displays (Sheet 1 of 2)

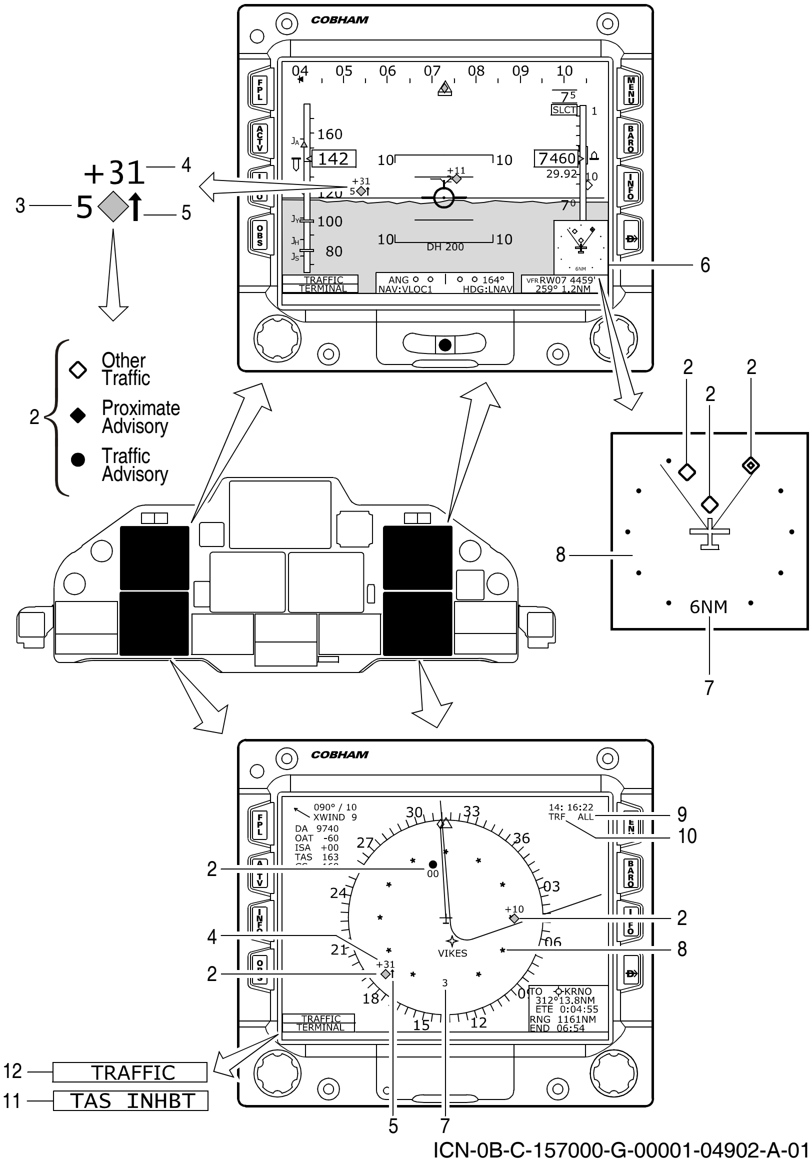

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Figure 7-110 Traffic Advisory System - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-110

Ref	Control/Display	Function
1	AURAL WARN / TAS switch	OFF - Prevents that the aural message of TAS is heard and the related caution messages are shown on the EFIS instruments. The TAS INHBT advisory message is shown on the MFD. ON - The aural annunciation of the traffic and the related warning flags are active.
2	Intruding aircraft symbol	Shows the position of the intruding aircraft respect to the helicopter. – A empty cyan diamond shows an intruding aircraft which does not meet the criteria for a Traffic Advisory (TA). The aircraft is detected beyond 6 NM and 1200 ft. – A full cyan diamond shows an intruding aircraft with a Proximate Advisory (PA) condition. The aircraft is detected within 6 NM and 1200 ft. – A full yellow circle shows an intruding aircraft within immediate vicinity based on flight parameters. When a full yellow circle is shown the display, the TAS generates an aural message consisting of a tone followed by the TRAFFIC, TRAFFIC voice message. The TRAFFIC caution message is shown together with the TRAFFIC, TRAFFIC voice message (see ref. 12).
3	Range indication	Shows in nautical miles the distance between the helicopter and the intruding aircraft. It is shown at the left of the intruding aircraft symbol.

Key to Figure 7-110 (Continued)

Ref	Control/Display	Function
4	Relative altitude indication	Shows the altitude of the intruding aircraft respect to the helicopter. It is shown in hundreds of feet below (-) the intruding aircraft symbol if the intruding aircraft is below the helicopter. It is shown above (+) the intruding aircraft symbol if the intruding aircraft is above the helicopter.
5	Vertical trend arrows	Is shown on the right side of the intruding aircraft symbol when the detected aircraft is climbing or descending at a rate greater than 500 ft/min. The arrow is upwards when the intruding aircraft is climbing. The arrow is downwards when the intruding aircraft is descending. The arrow is not shown when the intruding aircraft has a vertical speed lower than 500 ft/min or does not report the altitude.
6	Traffic thumbnail	Is automatically shown in the bottom right corner when there is a TA. The traffic thumbnail has clock face markings and a scale usually set at 6 NM. The scale is automatically adjusted in multiples of 2 NM (2 NM, 4 NM or 6 NM) to optimally display the traffic. The flight crew can declatter the traffic thumbnail. The traffic thumbnail disappears in the Unusual Attitude Mode. Display of the minimap and the traffic thumbnail are mutually exclusive with the traffic thumbnail taking precedence during a TA.
7	Reference ring scale	Shows the distance of the position reference ring from the helicopter. The scale is shown at the six o’clock position.
8	Position reference ring	Is used as reference to estimate the distance of the threat aircraft. Asterisks are shown around the helicopter symbol in the same positions of the clock.

Key to Figure 7-110 (Continued)

Ref	Control/Display	Function
9	Altitude filter indication	Shows the traffic altitude filter that was selected between AUTO, ABOVE, BELOW, NORMAL and ALL. For the selection of the traffic altitude filter, do the procedure that follows: 1. Push the MENU push-button on the MFD. 2. Push the FORMAT push-button. 3. Turn the right control knob to highlight ALT FILTER legend. 4. Push the right control knob. 5. Turn again the right control knob to scroll through the traffic altitude filter options, then push to set the selected option.
10	Traffic mode annunciator	Shown only when the Moving Map page is set on the MFD. When the TFR legend is shown, the Moving Map page supplies the traffic data. When an X is through the TFR legend, the traffic data are not correct.
11	TAS INHBT advisory message	Shown on the MFD when the AURAL WARN / TAS is set to OFF.
12	TRAFFIC caution message	Shown when a TA condition occurs. The caution message is not shown when the AGL altitude of the helicopter is less than 400 ft or when the intruding aircraft is below 200 ft AGL.

WEATHER RADAR SYSTEM

The weather radar system Honeywell type RDR 2000 consists of an antenna/receiver/transmitter group Honeywell type ART 2000, a configuration module Honeywell type CM 2000 and a radar control panel Honeywell type CP 466A.

The antenna/receiver/transmitter group is installed in the forward avionic compartment.
The group includes an antenna, a radar receiver and a radar transmitter constructed as a single unit mounted on a single support base.

The antenna array is attached to a microwave generator which are moved together as the radar scans.
The support base contains the receiver and the processing and control circuits.

The configuration module is installed on the same bulkhead which the antenna /receiver/transmitter group is installed on.
The system configuration and calibration data are memorized in the configuration module.

The radar control panel is installed in the forward interseat console.

The information processed by the system are shown on the MFD in proper formats or overlaid to the navigation information.
To interface the weather radar system with the multifunction display system, it is necessary that two modules are installed on the rear of the MFDs for the data conversion: the Weather Radar Module (WRM) and the ARINC 429 Expansion Module (AEM).

The weather radar system is supplied with electrical power from the Number 2 main bus through the WXR PWR and WXR CONTR circuit breakers.

Operation of the weather radar system

The weather radar system lets the visualization of the data about the weather conditions and the accurate representation of terrain maps.

The weather radar radiates pulses of radio waves in the X-band (on the frequency of 9.375 ± 30 MHz).
These pulses are reflected by sufficiently dense obstacles such as precipitations (raindrops, wet snow and hail), ground and solid structures.

The reflected signals (echoes) are processed by the receiver so that an image of the weather condition or the ground, that the pilot can easily interpret, can be shown on the selected MFD.
For this reason the different intensities of the reflected signals are coupled to the following colors according the growing order of intensity: black, green, yellow, red and magenta.

The operation of the weather radar is controlled by the function selector and the push-buttons on the radar control panel.

The function selector has four positions: OFF, STBY, TST and ON.

When the function selector is moved from OFF to STBY, the system warm-up is activated, the antenna is upwards in the parking position and does not perform any transmission of radar pulses.

When the function selector is set to TST, the self-test of the system is activated. A test pattern with five colors is shown on the MFD display.

When the function selector is set to ON, the Wx mode of operation is automatically selected.

The direction of the radar pulses can be changed by moving properly the antenna both in the vertical direction of ±15° with respect to the horizon through the TILT adjustment knob on the radar control panel and in the horizontal direction up to 50° each side with respect to the helicopter centerline through the left and right TRK push-buttons on the radar control panel. When the VP (Vertical Page) function is selected, the antenna performs a vertical scan of ±30°.

The movements of the antenna are stabilized in roll and pitch by the related signals from the AHRS1.

When the helicopter is on ground, the microswitch on the left leg of the main landing gear sets the system to the standby condition so that the radar cannot transmit, even if the function selector is in the ON position.

The push-buttons on the radar control panel permit to select the operation modes of the weather radar system (Wx, WxA and GND MAP), the VP (Vertical Profile) function and the azimuth position where the radar has to perform the vertical scanning.

Wx Mode (Weather Mapping)

The Wx mode (Weather Mapping) is automatically selected when the function selector is set to ON.
When a different mode is selected, the Wx mode is selected by pushing the Wx push-button on the radar control panel.
This mode is the normal operation mode of the radar for the analysis of the weather conditions.

WxA Mode (Weather Alert)

The WxA mode (Weather Alert) is selected by pushing the WxA push-button on the radar control panel. In this operation mode the magenta areas of the weather image flash with the purpose of advising the pilot about the presence of severe thunderstorms.

GND MAP Mode (Ground Mapping)

The GND MAP mode (Ground Mapping) is selected by pushing the GND MAP push-button on the radar control panel.
In the GND MAP mode the antenna must be turned downwards until the image on the MFD shows the desired portion of the ground.

The tilt angle that the antenna must have depends on altitude of the helicopter and the selected distance.

The gain value is preselected to provide the desired visualization of the ground mapping, but however it is possible to change this value to obtain the better visualization by turning the GAIN adjustment knob.

With the experience, the pilot will develop the ability to interpret the color display patterns that indicate water regions, coastal lines, hilly or mountainous regions, cities or even large structures.
It is possible that the pilot becomes familiar with the correct use of the GND MAP mode comparing the radar image on the MFD display with the ground during flights in clear visibility.

VP (Vertical Profile) Function

The VP (Vertical Profile) function is selected by pushing the VP push-button on the radar control panel.
This function permits to perform vertical scans of the antenna, above and below the helicopter horizontal plane, to process the information related to the scanned vertical portion and to show these information

on the display of the not-in-command MFD in a proper format ("Vertical Page").

The possibility of performing vertical scans permits the pilot to analyze the characteristics of particular interest of a thunderstorm such as the vertical development and the area of most concentrated precipitation.

After accessing the "Vertical Page", it is necessary to move the azimuth reference, in correspondence of which the vertical scan will be performed, to the desired position with the left and right TRK push-buttons. After the "Vertical Page" is shown, the antenna initiates the vertical scan of ±30° in the selected position.

Presentation of the radar image on the MFDs

The visualization of the radar image of the weather condition or the ground on the MFDs is obtained by pushing first the MENU push-button on the right side of each instrument, then selecting the FUNCTION label by pushing the NRST push-button, selecting the WX RDR page by turning the right knob and pushing it to confirm.
The radar image ("Horizontal Page") is shown on the display overlaid to the navigation information

Besides the radar image, the format shows the data related to the selected operation mode (Wx, WxA or GND MAP), the antenna inclination (TLT) and the insertion of antenna stabilization system (STAB OFF or STAB LMT).

When the VP function is selected on the radar control panel, the MFD display shows the image of the vertical scan ("Vertical Page") with the data of the selected operation mode (Wx or GND MAP), the selected position in azimuth (TRK) and the reference scale.

When a failure to the system is detected, the radar image disappears, while the warning messages are shown.

WEATHER RADAR SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-110A.

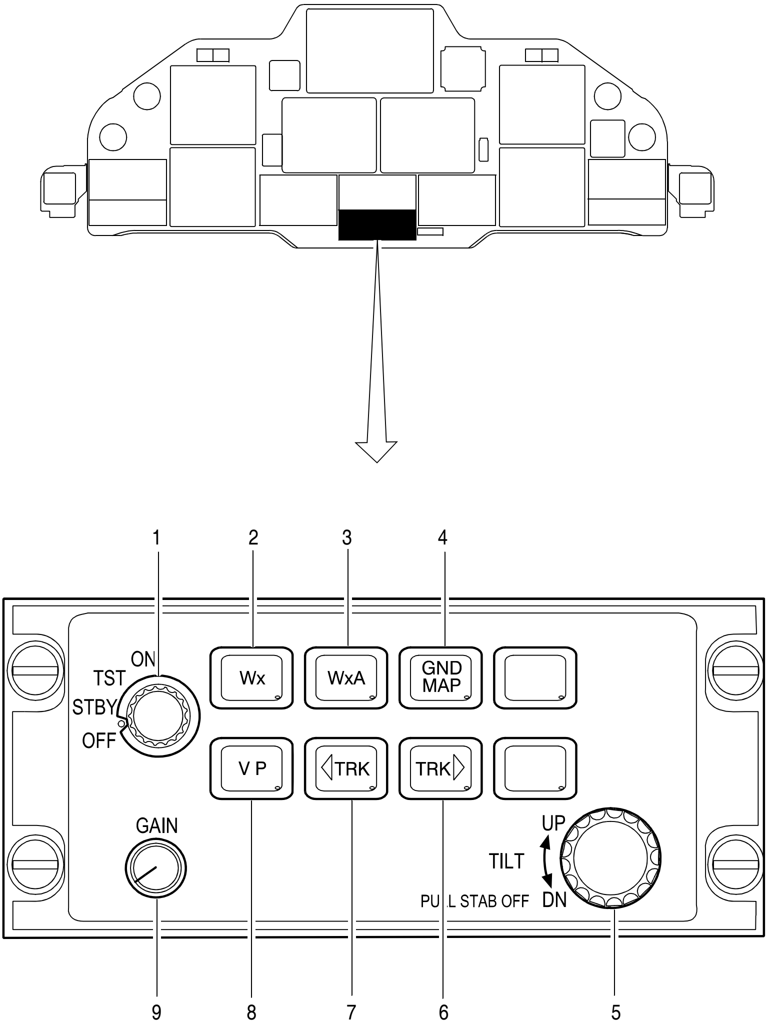

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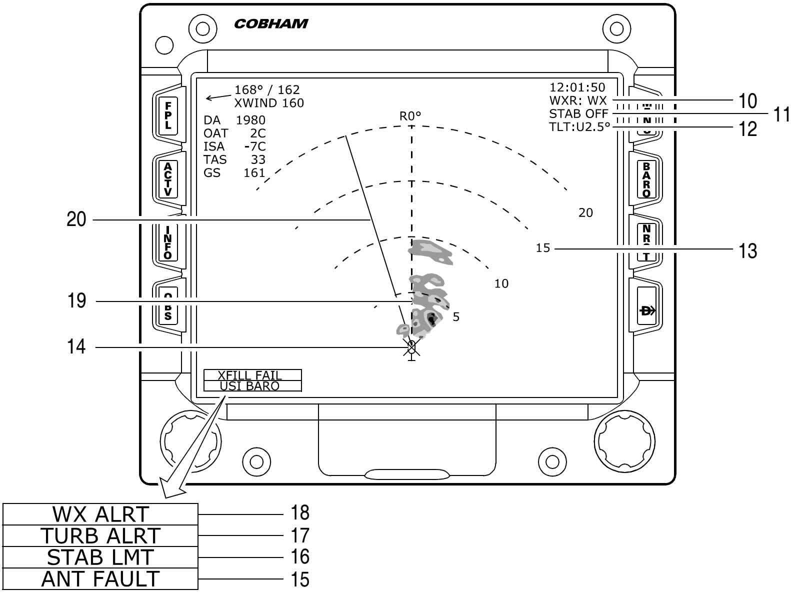
Figure 7-110A Weather Radar System - Controls and Displays (Sheet 1 of 2)

WXR - "HORIZONTAL PAGE" FORMAT



WXR - "VERTICAL PAGE" FORMAT

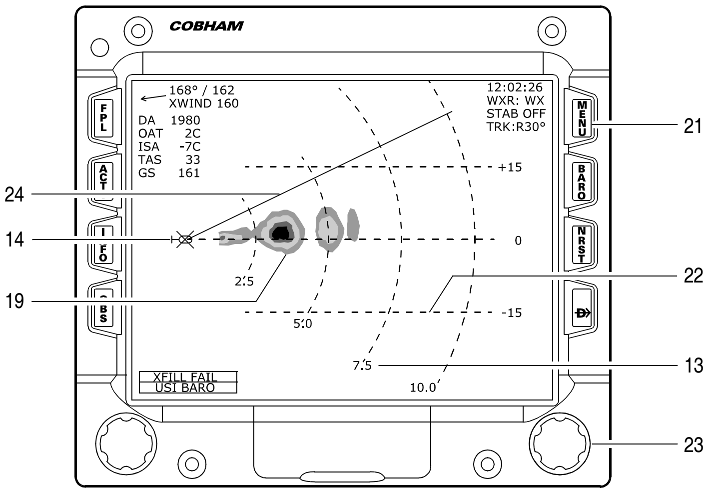


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Figure 7-110A Weather Radar System - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-110A

Ref.	Control/Display	Function
1	Function selector	OFF - The system is de-energized.
		STBY - Operates the system warm-up period. The antenna is upwards in the parking position and does no transmission of the radar pulses.
		TST - Operates the system self-test. On the set MFD a test pattern of five colors and the TEST legend are automatically shown.
		ON - The Wx mode of operation is automatically set.
2	Wx push-button	Sets the Wx mode (Weather Mapping).
		The WX legend is shown on the set MFD (see ref. 10).
3	WxA push-button	Sets the WxA mode (Weather Alert).
		The WXA legend is shown on the set MFD (see ref. 10).
		The magenta areas of the image flash.
4	GND MAP push-button	Sets the GND MAP mode (Ground Mapping).
		The GMAP legend is shown on the set MFD (see ref. 10).
		The GN indicator is shown in the upper right corner (see ref. 25).
		The radar image does not use the red color.
5	TILT adjustment knob	Turned towards DN or UP - Adjusts the antenna tilt and points the radar signal beam from -15° to +15° with respect to the horizon.
		The antenna tilt angle is shown on the set MFD (see ref. 12).
		The horizon position is indicated as 0° on the knob.
		Pulled - The antenna stabilization system does not operates more and this causes the STAB OFF legend to be shown on the set MFD (see ref. 11).

Key to Figure 7-110A (Continued)

Ref.	Control/Display	Function
6	Right TRK push-button	Pushed - Moves the radar antenna up to 50° right with increments of 1° at a rate of about 15° per second. The azimuth position of antenna is shown on the set MFD (see ref. 21).
7	Left TRK push-button	Pushed - Moves the radar antenna up to 50° left with increments of 1° at a rate of about 15° per second. The azimuth position of antenna is shown on the set MFD (see ref. 21).
8	VP push-button	Pushed the first time - Sets the VP function. When the VP function is set, the display of the not-in-command MFD shows in a proper visualization page ("Vertical Page") the image of the vertical scan of ±30° with respect to the horizon in the set azimuth position. The selection of the VP function does not change the set function or mode (TEST, Wx or GND MAP), but it can be changed after the VP function is set. Pushed the second time - The display of the not-in-command MFD shows again the image of the page shown before the VP function was set.
9	GAIN adjustment knob	Turned - Changes the gain of the radar receiver in the GND MAP mode. In the VP function and in the Wx, WxA and TEST modes the gain is preset.
10	WXR indicator	Shows the operation mode of the weather radar system: OFF (red crossed) - Shows that the system is de-energized. STBY - Shows that the system is in the warm-up period (function selector in STBY position).

Key to Figure 7-110A (Continued)

Ref.	Control/Display	Function
		TEST - Shows that the system self-test is in progress (function selector in the TST position). The MFDs show the test pattern. WX - Shows that the Wx mode operates. WXA - Shows that the WxA mode operates. GMAP - Shows that the GND MAP mode operates.
11	STAB OFF legend	Shown when the antenna stabilization system does not operates (see ref. 5).
12	TLT indicator	Shows the tilt angle of the antenna. The angle is shown with the U or D letter to which the angle value in degrees from -16° to +15,75° follows. The value is set with the TILT adjustment knob (see ref. 5).
13	Distance reference scale	Lets the distance of a thunderstorm from the helicopter be calculated.
14	Helicopter symbol	The helicopter nose indicates the start point of the reference scale of the radar scan and the turning point of the radar scan.
15	WX ALRT caution message	Shown when there is a weather alert condition.
16	TURB ALRT caution message	Shown when there is a turbulence alert condition.
17	STAB LMT caution message	Shown when the change of the helicopter attitude has exceeded the limits of the antenna stabilization system.
18	ANT FAULT caution message	Shown when the turbulence causes the weather radar antenna be temporarily disconnected.
19	Image of the weather condition and ground	Shown as process result of echoes that the system receives. The image is made with different colors in the intensity order that follows: black, green, yellow, red and magenta.

Key to Figure 7-110A (Continued)

Ref.	Control/Display	Function
20	Track line	Shows the azimuth angle where the antenna does the horizontal scan.
21	MENU push-button	When pushed, lets the FUNCTION option be set through the NRST push-button. When the FUNTION option is set, It is possible to set the WX RDR format on the MFDs with the right knob (turn and push).
21	TRK indicator	Shows the azimuth angle value where the antenna is put to do the vertical scan. The angle is shown with the L or R letter to which the azimuth angle value (from 50° left to 50° right) follows. The value is set with the TKR push-buttons of the radar control panel (see ref. 6 and 7). The azimuth angle value set to do the scan is visible only on the "Vertical Page".
22	Vertical altitude indication	Shown with two lines above and below the helicopter center line. The two lines show +/- 15 ft respect to the center line with a distance reference scale of 10 NM; ±30 ft for 20 NM; ±60 ft for 40 NM; ±120 ft for 80NM; ±240 ft for 160 NM; ±360 ft for 240 NM.
23	Right knob	Turned clockwise - Increases the reference distance in the following sequence (values in NM): 10 - 20 - 40 -80 - 160 - 240. Turned counter-clockwise - Decreases the reference distance in the following sequence (values in NM): 240 - 160 - 80 - 40 - 20 - 10.
24	Elevation line	Shows the elevation angle where the antenna performs the vertical scan.
25	GN indicator	Shows the receiver gain value in the GND MAP. The indicator can show the indication that fol-low: CAL, -xxDB, +0DB, MAX.

DEPENDENT POSITION DETERMINING SYSTEM

The system provides navigation data to determine the position of the helicopter with information acquired from ground installations and orbital satellites.

The dependent position determining system includes:

– The VOR1/ADF system
– The VOR2/ILS system
– The DME system
– The ATC transponder system.
– The digital map generation system.

VOR1/ADF SYSTEM

The VOR1/ADF system consists of a receiver Rockwell Collins type NAV-4000, installed in the bay below the aft avionic compartment, three antenna couplers, installed in the aft avionic compartment, a VOR/LOC antenna, composed by two half antennas installed respectively on the left side and on the right side of the tail boom, a glideslope antenna, installed on the bottom of the helicopter nose, the marker beacon antenna, installed longitudinally on the fuselage bottom between the two main landing gears and an ADF antenna Rockwell Collins type ANT-462A, installed on the fuselage bottom behind the marker beacon antenna.

The receiver type NAV-4000 contains the VOR/ILS/MKR and ADF receivers in the same unit.

The VOR/ILS/MKR receiver receives and processes the VOR signals in the frequency range between 190 and 1799 / 2088 and 2094 / 2179 and 2195 kHz, the localizer signals in the frequency range between 108.10 and 111.95 MHz, the glideslope signals in the frequency range between 329.15 and 335.00 MHz and the marker beacon signals at 75.0 MHz.

The signals are used to provide the bearing of VOR station, the deviation from the selected course, the To/From indications, the localizer and glideslope deviations and the indications of the passage of the helicopter over the Marker Beacon stations. All these data are shown on the EFIS instruments and on the standby electronic indicator.

The ADF receiver provides the bearing of NDB stations or radio stations which broadcast in the AM frequency ranges between 190 and 1799, 2088 and 2094, 2179 and 2185 kHz. The continuous bearing of the selected transmitting station is shown on the MFDs.

The receiver is interfaced with the RMS through ARINC 429 busses, with the ICS for the reception in the headsets of the VOR and marker beacon signals, with the DME, the AFCS and the FMS included in the EFIS instrument.

The VOR1/ADF system is supplied with electrical power from the 28 V dc Number 1 main bus through the VOR 1 ADF circuit breaker.

Operation of the VOR1/ADF System

The VOR1 system is controlled through the RTU panel of the RMS, on which the TLP1 page must be selected to access the related NAV1 control display.
By pushing twice the multi-function push-button on the left side of the panel and adjacent to the NAV1 control display, it is possible to access the MDP page and then by pushing the PRESET PAGE push-button the preset page.

The display sequence of the VOR1 control pages is shown in Figure 7-111.

The VOR1 system allows to receive the VOR/ILS signals using the active frequency shown on the left side of the NAV1 control display. The active frequency can be replaced by the standby frequency, shown on the right side of the display.
The standby frequency can be set with the adjacent multi-function push-button and the tuning knobs of the RTU panel or selected between the 20 frequencies memorized in as many channels.
All the channels can be programmed by selecting the preset page.

The VOR1 system allows also to receive the signals of the passage of the helicopter over the marker beacon stations.
A flashing letter shown on the bottom part of the PFD displays advises about the passage.

The letters are as follow:

– O letter of white color in a box of cyan color for the outer beacon (Outer Marker) (9 km from the runway)

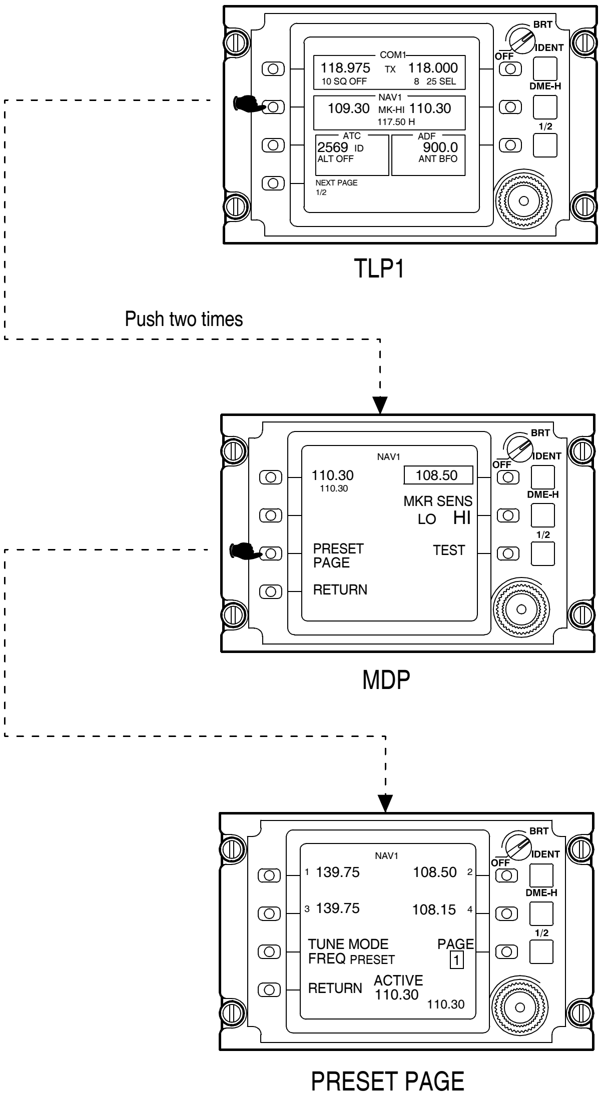

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Figure 7-111 Sequence of VOR1 System Control Pages on RTU Panel

– M letter of black color in a box of yellow color for the middle beacon (Middle Marker) (1 km from the runway)
– I letter of black color in a box of white color for the inner beacon (Inner Marker) or the airway beacon.

If more than one beacon is received by the airborne receiver, the letter of the beacon with the highest priority is only shown.
The priority order is the following: outer, middle and inner.
The letter disappears as soon as the beacon signal is not more available.

The passage over the vertical reference stations is also announced in the headset with signals modulated at 400 Hz for the outer beacon, 1300 Hz for the middle beacon and 3000 Hz for the inner or airway beacon.
The audio signals are received in Morse code and consist of a sequence of lines for the outer beacon, lines and dots for the middle beacon and a continuous signal for the inner or airway beacon.

By accessing to the MDP page of the NAV1 control display on the RTU panel, it is possible to adjust the receiver sensibility and to perform the test of the receiver.

The ADF system is also controlled through the RTU panel of the RMS, on which the TLP1 or TLP2 page must be selected to access the ADF control display.

By pushing twice the multi-function push-button on the right side of the panel and adjacent to the ADF control display, it is possible to access the MDP page and then by pushing the PRESET PAGE push-button the preset page. The display sequence of the ADF system control pages is shown in Figure 7-112.

The ADF system can operate in two modes: ANT and ADF.

In the ANT mode the receiver operates as simple receiver of audio signals that are sent to the headsets through the ICS.

In the ADF mode the system also determines the magnetic heading of the selected station, indicated on the MFDs by the heading pointer.

VOR1/ADF SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-113.

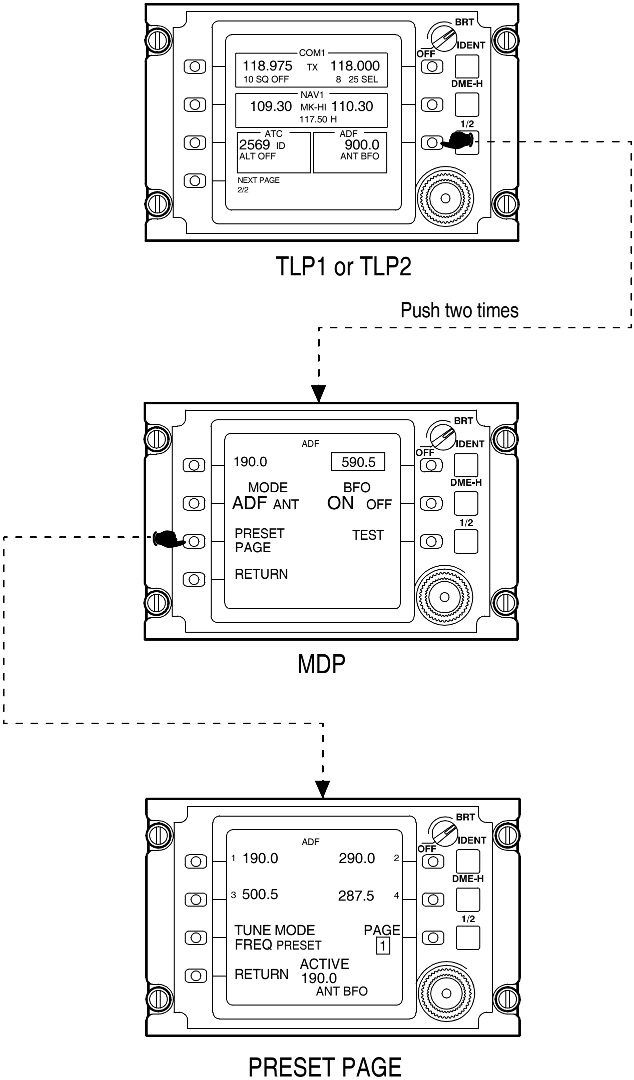

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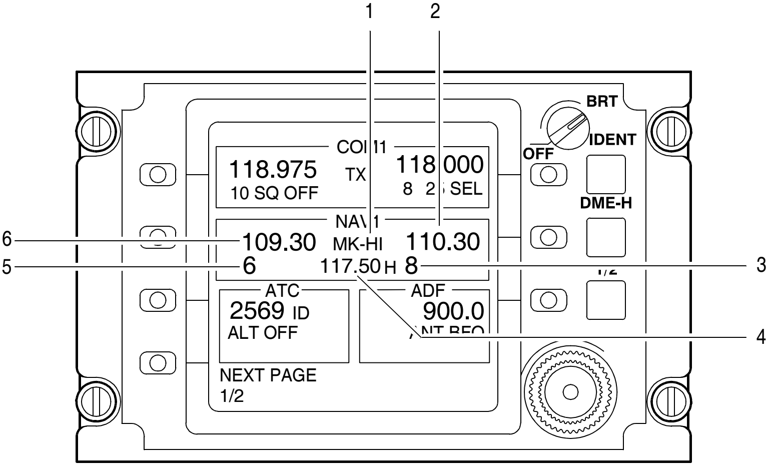
Figure 7-112 Sequence of ADF System Control Pages on RTU Panel

TLP1 PAGE



MDP PAGE - NAV1

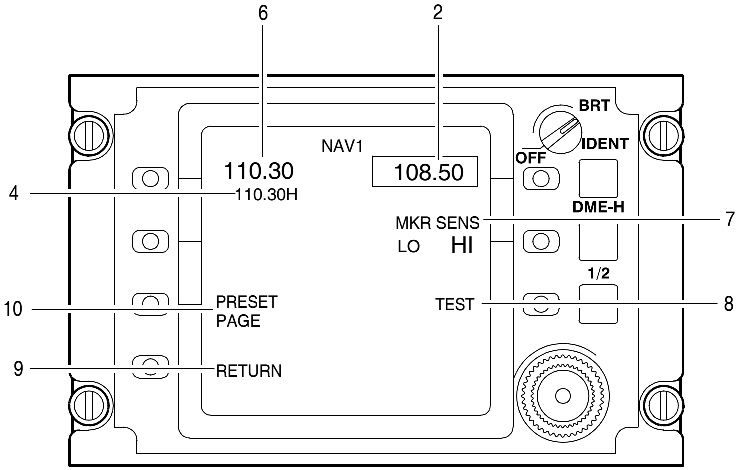


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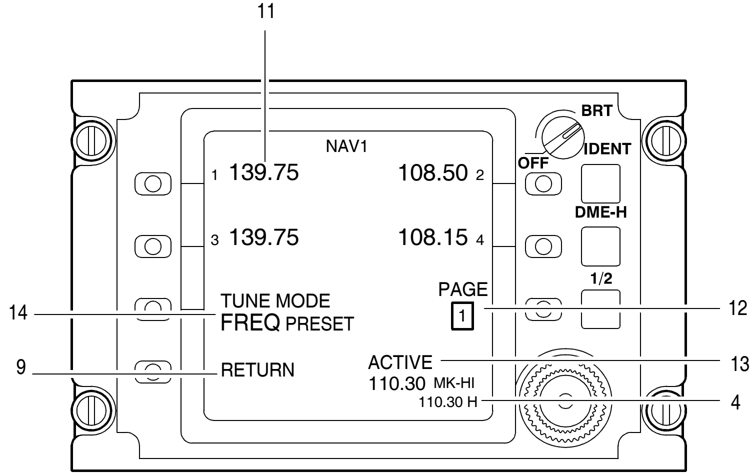
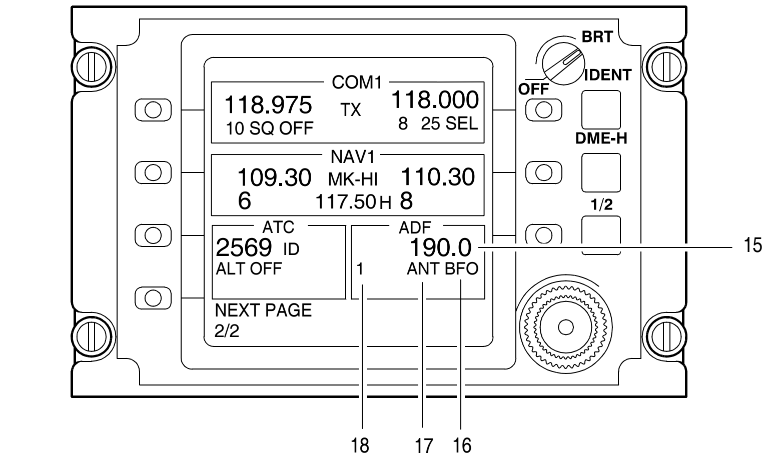
Figure 7-113 VOR1/ADF System - Controls and Displays (Sheet 1 of 5)

PRESET PAGE - NAV1



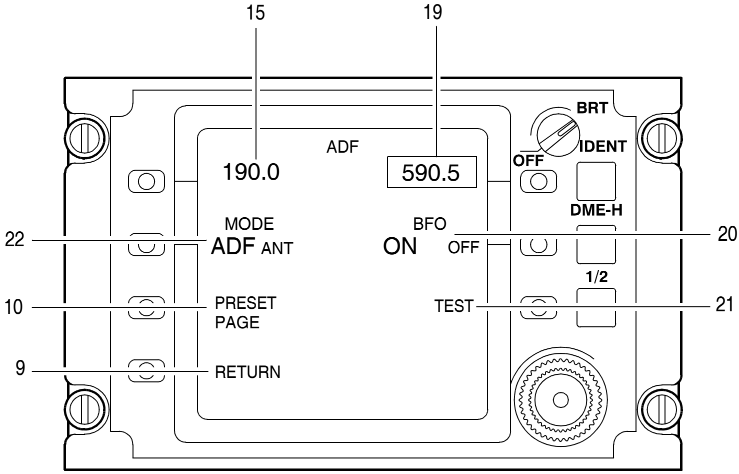
TLP1 PAGE



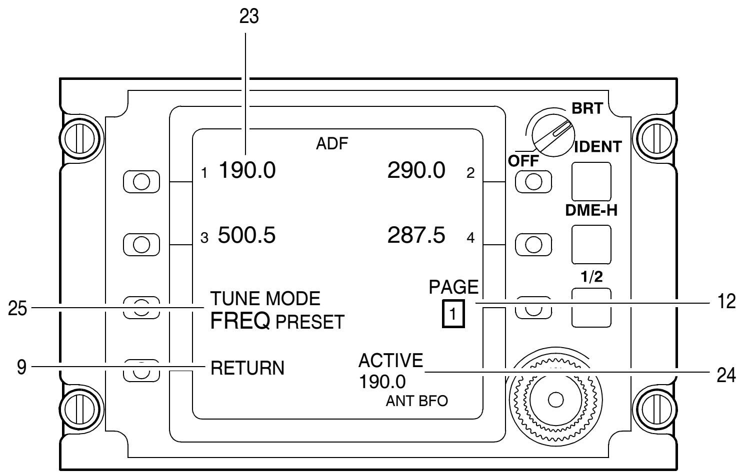
ICN-0B-C-157000-G-00001-02493-A-01-1

Figure 7-113 VOR1/ADF System - Controls and Displays (Sheet 2 of 5)

MDP PAGE - ADF



PRESET PAGE - ADF



ICN-0B-C-157000-G-00001-02494-A-01-1

Figure 7-113 VOR1/ADF System - Controls and Displays (Sheet 3 of 5)

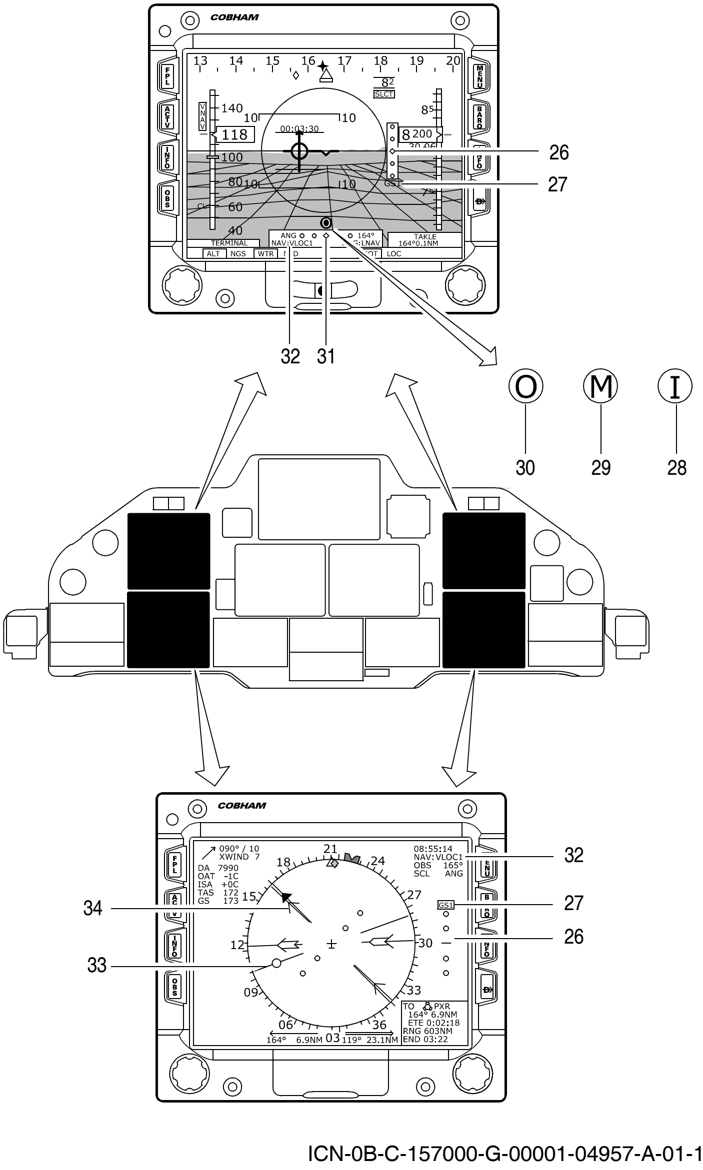
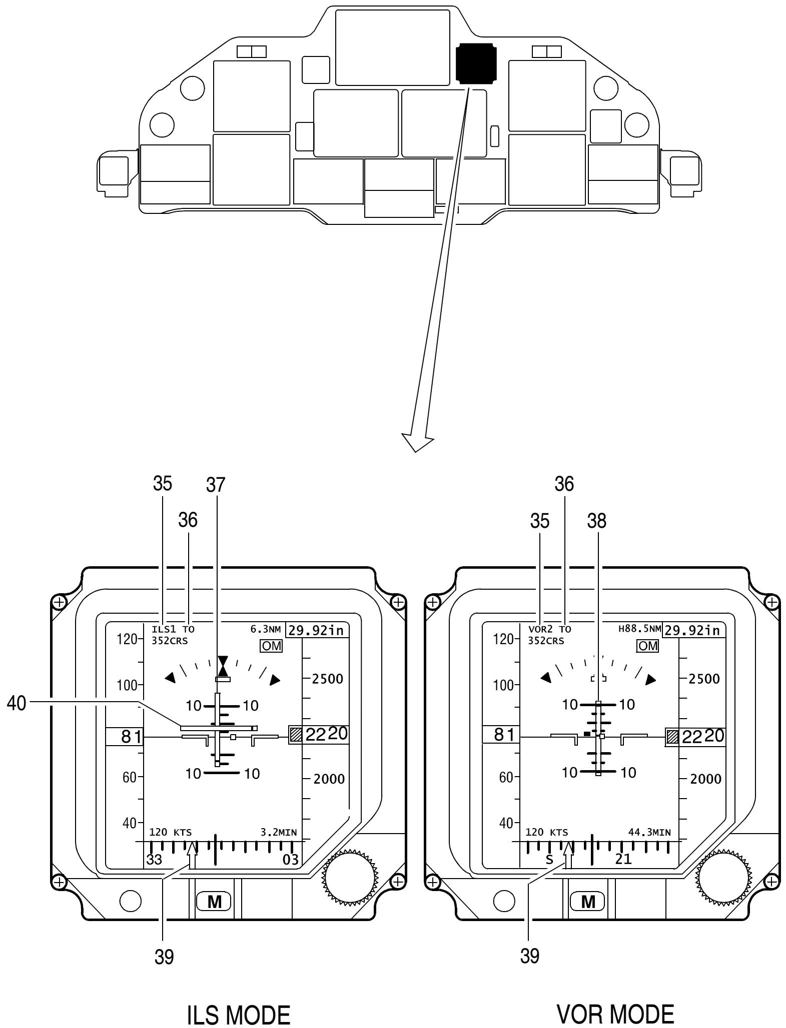


Figure 7-113 VOR1/ADF System - Controls and Displays (Sheet 4 of 5)



ICN-0B-C-157000-G-00001-04958-A-01-1

Figure 7-113 VOR1/ADF System - Controls and Displays (Sheet 5 of 5)

Key to Figure 7-113

Ref	Control/Display	Function
1	"MK-HI" legend	Shown when the marker beacon reception is adjusted to the maximum level of sensibility (see ref. 7).
2	VOR standby frequency indicator	Shows the VOR standby frequency. By pushing the adjacent multi-function push-button, the frequency is highlighted by a white box (tune window), then the frequency can be changed with the tuning knobs in the range from 190 and 1799 / 2088 and 2094 / 2179 and 2195 kHz. The VOR standby frequency can also be selected between the frequencies of the 20 preset channels. By pushing again the adjacent multi-function push-button, the frequency is swapped as active frequency.
3	VOR standby channel indicator	Shows the standby channel number selected with the tuning knobs when the tuning mode is set to PRESET (see ref. 14). The standby channel can be swapped as active channel with the adjacent multi-function push-button. 20 preset channels are available (see ref. 11). **NOTE** When the frequency associated with a channel is swapped as active frequency, the standby channel indicator shows the "RCL" legend. The previous active frequency remains associated with the standby channel. This frequency can become again the active frequency by selecting the "RCL" legend in the standby channel indicator.

Key to Figure 7-113 (Continued)

Ref	Control/Display	Function
4	DME frequency indicator	Shows the frequency on which the DME is tuned to compute the distance to the VOR/ DME station. The letter H, if present, indicates that the DME-H function was selected on the RTU panel or better that DME indication is held referred to the frequency shown before the letter H, even if the active VOR frequency is changed.
5	VOR active channel indicator	Shows the active channel number selected with the tuning knobs when the tuning mode is set to PRESET (see ref. 14).
6	VOR active frequency indicator	Shows the VOR active frequency. By pushing the adjacent multi-function push-button, the frequency is highlighted by a white box (tune window), then the frequency can be changed with the tuning knobs in the range from 190 and 1799 / 2088 and 2094 / 2179 and 2195 kHz.
7	MKR SENS indicator	Shows the marker beacon reception sensibility. With the adjacent multi-function push-button it is possible to select the marker beacon reception sensibility between: – LO - Low sensibility – HI - High sensibility. The selected sensibility is shown in larger letters.
8	TEST identifier	By pushing the adjacent multi-function push-button on the right side, it is possible to perform the test of the VOR/ILS/MKR receiver. The TEST identifier is shown in larger letter while the test is performed. At the end of the test the CODE annunciator and two digit hexadecimal code are shown below the TEST identifier. The code indicates the test result. The code is 00 if no failures are found.

Key to Figure 7-113 (Continued)

Ref	Control/Display	Function
9	RETURN legend	By pushing the adjacent multi-function push-button on the left side, the Menu on the display comes back to TLP1 page.
10	PRESET PAGE legend	By pushing the adjacent multi-function push-button on the left side, it is possible to access the preset page for the programming of the preset channels.
11	Preset channel programming indicator	Shows the channel number and the associated frequency value. By highlighting the channel number with the adjacent multi-function push-button, it is possible to associate in the memory a frequency value selected with the tuning knobs. 20 channels can be programmed. The preset channels are distributed on five pages. The other fields are the same on all the five pages.
12	Page number indicator	Shows which page of the five preset pages is shown on the display. When the indicator is selected by pushing the adjacent multi-function push-button, it is possible to scroll the page number by turning one of the two tuning knobs.
13	VOR active frequency indicator	Shows the VOR active frequency used on the receiver.
14	VOR tuning mode indicator	Shows the VOR active tuning mode. By pushing the adjacent multi-function push-button on the left side, it is possible to change the tuning mode. The active tuning mode is shown in large letters. The selectable tuning modes are as follow: – FREQ: The receiver is tuned on the selected VOR frequency. – PRESET: The receiver is tuned on the frequency associated with one of the preset channels.

Key to Figure 7-113 (Continued)

Ref	Control/Display	Function
15	ADF active frequency indicator	Shows the ADF active frequency. By pushing the adjacent multi-function push-button on the left side, the frequency is high-lighted by a white box (tune window), then the frequency can be changed with the tuning knobs in the range from 108.00 and 117.95 MHz. **NOTE** If the active frequency is the emergency fre-quency (channel 20 = 2182.0 kHz), the EMER legend is shown next to the frequency.
16	BFO annunciator	Shown when the selected NDB station is of interrupted carrier type
17	ANT annunciator	Shown when the ADF receiver operates in audio mode only with no station bearing
18	Active channel indicator	Shows the active channel number selected when the receiver tuning is set to channel (PRESET) mode (see ref.25).
19	ADF standby frequency indicator	Shows the ADF standby frequency. By pushing the adjacent multi-function push-button on the right side, the ADF standby fre-quency is highlighted by a white box (tune win-dow), then the frequency can be changed with the tuning knobs in the range from 190 and 1799 / 2088 and 2094 / 2179 and 2195 kHz. The ADF standby frequency can also be selected between the frequencies of the 20 preset channels. By pushing again the adjacent multi-function push-button, the frequency is swapped as active frequency (see ref. 15).
20	BFO function indicator	Shows if the BFO function is active or not. By pushing the adjacent multi-function push-button, the BFO function changes from the ON condition to the OFF condition and vice-versa. The active condition is shown in large letters.

Key to Figure 7-113 (Continued)

Ref	Control/Display	Function
21	TEST identifier	By pushing the adjacent multi-function push-button on the right side, it is possible to perform the test of the ADF receiver. The TEST identifier is shown in larger letter while the test is performed. At the end of the test the CODE annunciator and two digit hexadecimal code are shown below the TEST identifier. The code indicates the test result. The code is 00 if no failures are found.
22	Indicator of operation mode	Indicates which operation mode is active. By pushing the adjacent multi-function push-button on the left side, it is possible to change the operation mode. The active mode is of cyan color. The selectable operation modes are as follow: – ANT: The receiver operates as simple receiver of audio signals. – ADF: The receiver also determines the magnetic heading of the selected station.
23	Preset channel programming indicator	Shows the channel number and the associated frequency value. By highlighting the channel number with the adjacent multi-function push-button, it is possible to associate in the memory a frequency value selected with the tuning knobs. 19 channels can be programmed, while the channel 20 is reserved to the emergency frequency (2182.0 kHz). The preset channels are distributed on five pages. The other fields are the same on all the five pages.
24	ADF active frequency indicator	Shows the ADF active frequency used on the receiver and the active functions.

Key to Figure 7-113 (Continued)

Ref	Control/Display	Function
25	ADF tuning mode indicator	Shows the ADF active tuning mode. By pushing the adjacent multi-function push-button on the left side, it is possible to change the tuning mode. The active tuning mode is shown in large letters. The ADF selectable tuning modes are as follow: – FREQ: The receiver is tuned on the selected ADF frequency. – PRESET: The receiver is tuned on the frequency associated with one of the preset channels.
26	Vertical deviation indicator	Shows the deviation of the helicopter from the flight path when the glideslope signals are valid. When the needle is within one dot of the vertical deviation scale, the other dots are empty.
27	VNAV source indicator	Shows the source of signals for the vertical navigation of the helicopter. The GS1 legend is shown at the bottom of the vertical deviation scale on the PFDs. The GS1 legend is shown at the top of the vertical deviation scale on the MFDs.
28	Inner marker beacon indicator	I - Shown when the helicopter flights over a inner marker beacon.
29	Middle marker beacon indicator	M - Shown when the helicopter flights over a middle marker beacon.
30	Outer marker beacon indicator	O - Shown when the helicopter flights over an outer marker beacon.

Key to Figure 7-113 (Continued)

Ref	Control/Display	Function
31	Course/Localizer deviation indicator	Shows the deviation of the helicopter from the set course or localizer when the VOR/ILS signals are valid. When the needle is within one dot of the course deviation scale, the other dots are empty. When the needle is at a scale end position, it flashes. A magenta triangle shows the TO/FROM indication. When the triangle points up, it is a TO indication. When the triangle points down, it is a FROM indication.
32	NAV source indicator	Shows the VOR1 or LOC1 legend when the set navigation source is the VOR1/ILS system.
33	ADF pointer	Shows the bearing of a NDB station.
34	VOR1 pointer	Shows the bearing of a VOR station.
35	Mode indicator	Shows the mode of the data presentation. The modes applicable to the VOR1/ADF system are: VOR1 and ILS1.
36	TO/FROM indicator	Shows if the helicopter flights to or away from the set VOR/ILS station.
37	Localizer deviation indicator	Moves left or right to show the left or right deviation from the localizer.
38	Course deviation indicator	Moves left or right to show the left or right deviation from the set course.
39	Course indicator	Shows the course set with the adjustment knob.
40	Glideslope deviation indicator	Moves up or down to show the up or down deviation from the glideslope.

VOR2/ILS SYSTEM

The VOR2/ILS system consists of a VOR/ILS/MKR receiver Rockwell Collins type NAV-4500, installed in the bay below the aft avionic compartment.
The system uses the same three antenna couplers and the same VOR/LOC, glideslope and marker beacon antennas of the VOR1 system.

The receiver receives and processes the VOR signals in the frequency range between 190 and 1799 / 2088 and 2094 / 2179 and 2195 kHz, the localizer signals in the frequency range between 108.10 and 111.95 MHz, the glideslope signals in the frequency range between 329.15 and 335.00 MHz and the marker beacon signals at 75.0 MHz.

The signals are used to provide the bearing of VOR station, the deviation from the selected course, the To/From indications, the localizer and glideslope deviations and the indications of the passage of the helicopter over the marker beacon stations. All these data are shown on the EFIS instruments and on the standby electronic indicator.

The receiver is interfaced with the RMS through ARINC 429 busses, with the ICS for the reception in the headsets of the VOR and Marker Beacon signals, with the DME, the AFCS and the FMS included in the EFIS instruments.

The VOR2/ILS system is supplied with electrical power from the 28 V dc Number 2 emergency bus through the VOR 2 ILS circuit breaker.

Operation of the VOR2/ILS System

The VOR2/ILS system is controlled through the RTU panel of the RMS, on which the TLP2 page must be selected to access the related NAV2 control display.
By pushing twice the multi-function push-button on the left side of the panel and adjacent to the NAV2 control display, it is possible to access the MDP page and then by pushing the PRESET PAGE push-button the preset page.

The display sequence of the VOR2/ILS system control pages is shown in Figure 7-114.

The VOR2/ILS system allows to receive the VOR/ILS signals using the active frequency shown on the left side of the NAV2 control sys-

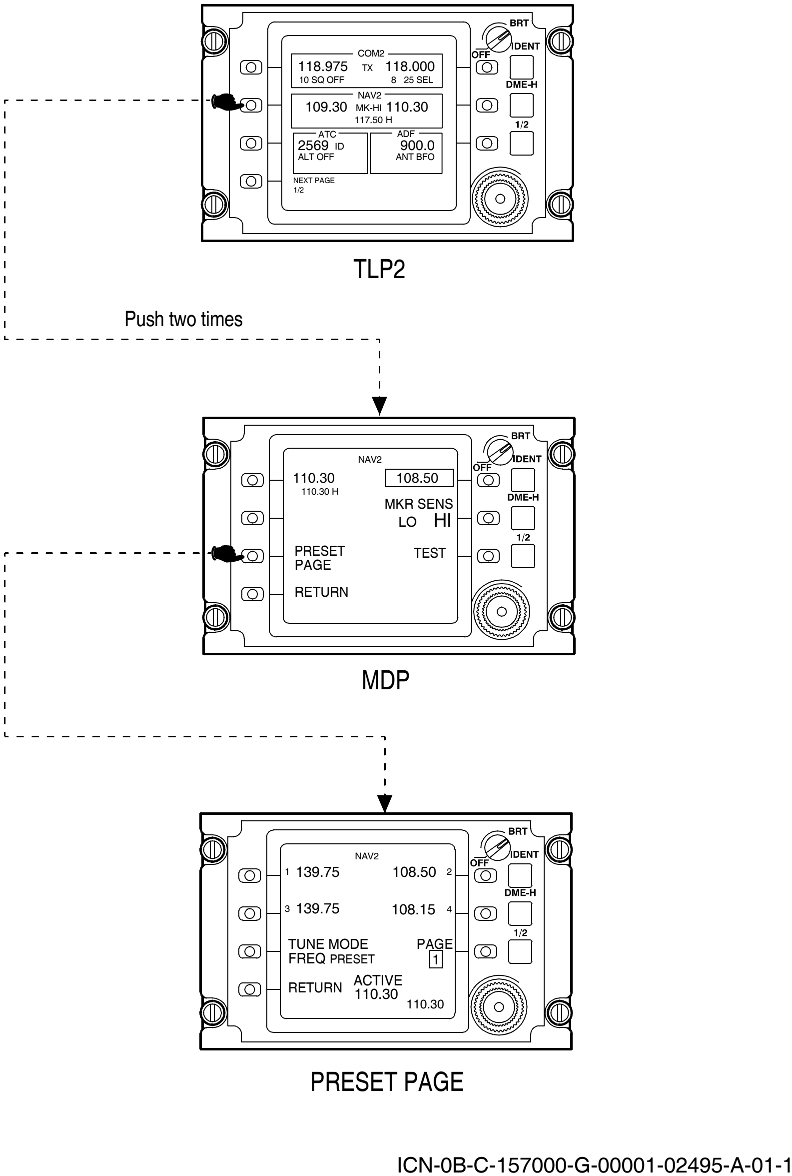


Figure 7-114 Sequence of VOR2/ILS System Control Pages on RTU Panel

tem. The active frequency can be replaced by the standby frequency, shown on the right side of the display.
The standby frequency can be set with the adjacent multi-function push-button and the tuning knobs of the RTU panel or selected between the 20 frequencies memorized in as many channels.
All the channels can be programmed by selecting the preset page.

The VOR2/ILS system allows also to receive the signals of the passage of the helicopter over the Marker Beacon stations.
A flashing letter shown on the bottom part of the PFD displays advises about the passage.

The letters are as follows:

– O letter of white color in a box of cyan color for the outer beacon (Outer Marker) (9 km from the runway)
– M letter of black color in a box of yellow color for the middle beacon (Middle Marker) (1 km from the runway)
– I letter of black color in a box of white color for the inner beacon (Inner Marker) or the airway beacon.

If more than one beacon is received by the airborne receiver, the letter of the beacon with the highest priority is only shown.
The priority order is the following: outer, middle and inner.
The letter disappears as soon as the beacon signal is not more available.

The passage over the vertical reference stations is also announced in the headset with signals modulated at 400 Hz for the outer beacon, 1300 Hz for the middle beacon and 3000 Hz for the inner or airway beacon.
The audio signals are received in Morse code and consist of a sequence of lines for the outer beacon, lines and dots for the inner beacon and a continuous signal for the inner or airway beacon.

By accessing to the MDP page of the NAV2 control display on the RTU panel, it is possible to adjust the receiver sensibility and to perform the test of the receiver.

VOR2/ILS SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-115.

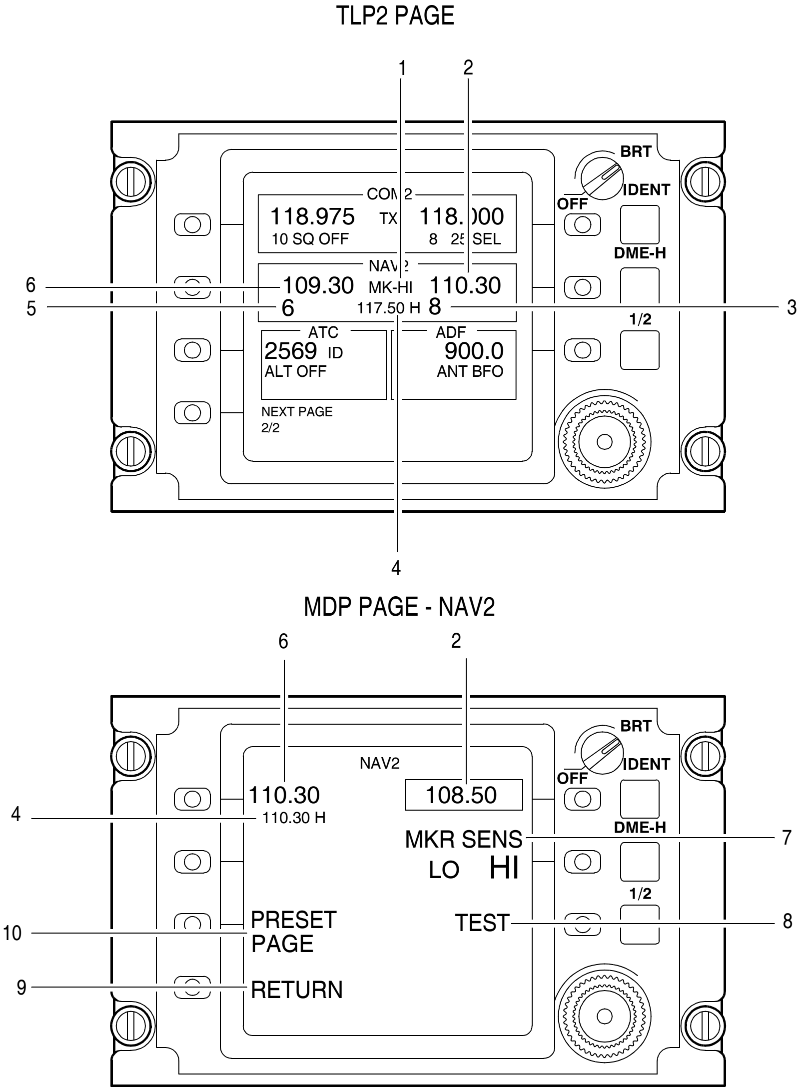


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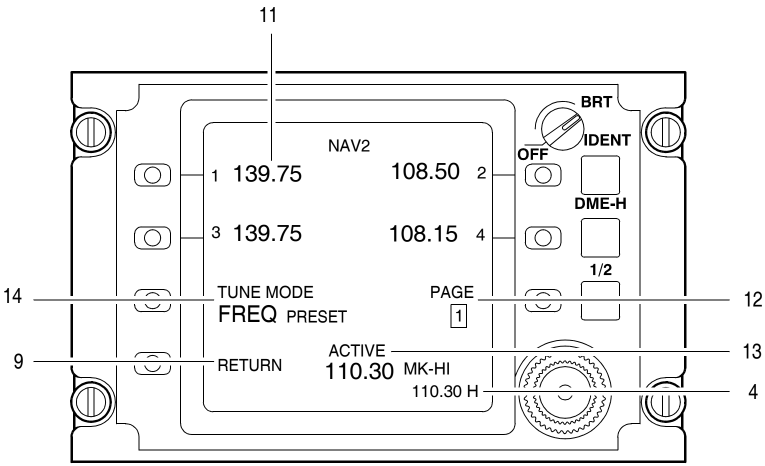
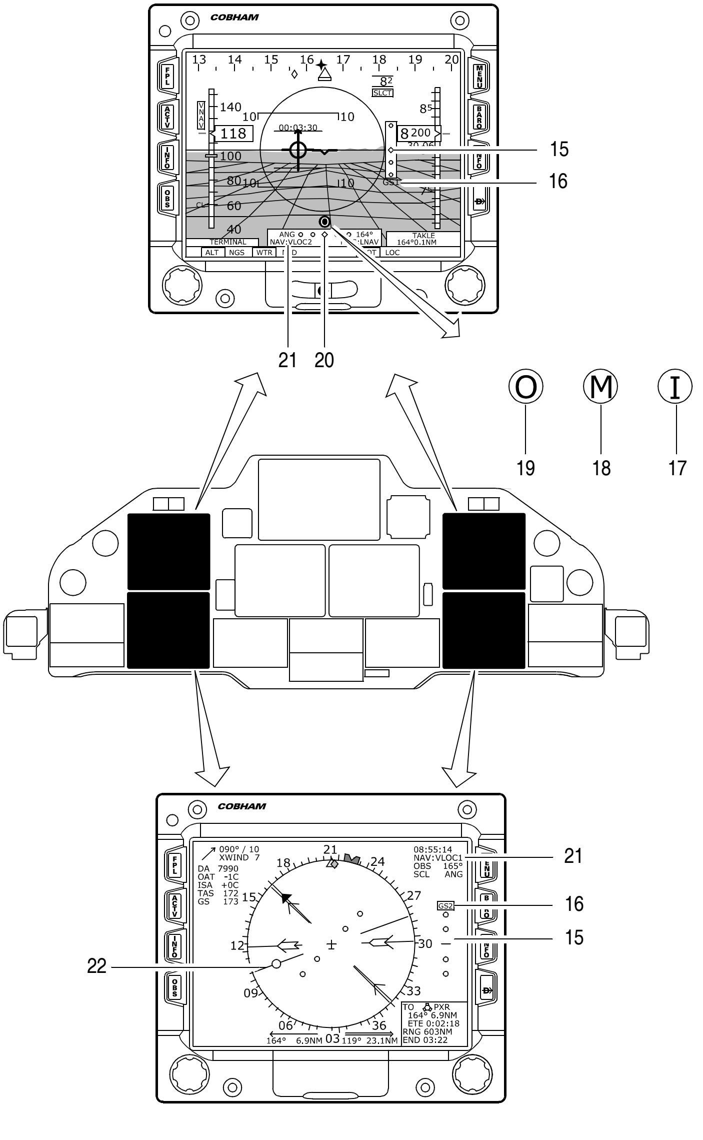
Figure 7-115 VOR2/ILS System - Controls and Displays (Sheet 1 of 4)

PRESET PAGE - NAV2



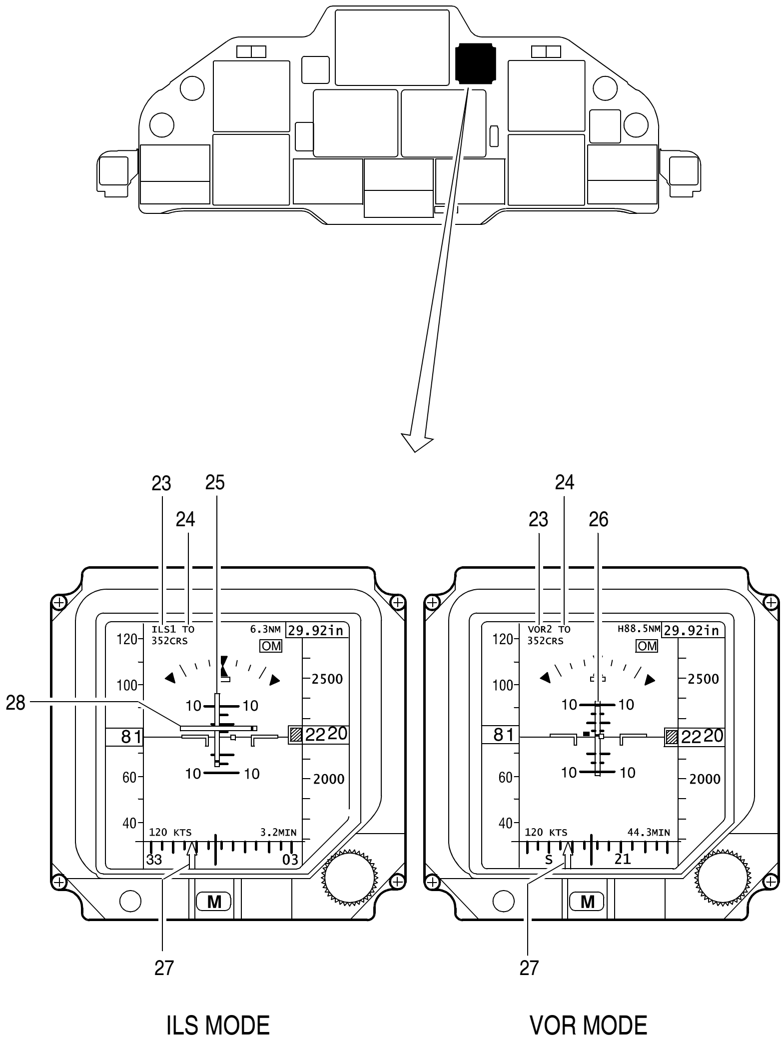
ICN-0B-C-157000-G-00001-02497-A-01-1

Figure 7-115 VOR2/ILS System - Controls and Displays (Sheet 2 of 4)



ICN-0B-C-157000-G-00001-04959-A-01-1

Figure 7-115 VOR2/ILS System - Controls and Displays (Sheet 3 of 4)



ICN-0B-C-157000-G-00001-04960-A-01-1

Figure 7-115 VOR2/ILS System - Controls and Displays (Sheet 4 of 4)

Key to Figure 7-115

Ref	Control/Display	Function
1	MK-HI legend	Shown when the marker beacon reception is adjusted to the maximum level of sensibility (see ref. 7).
2	VOR standby frequency indicator	Shows the VOR standby frequency. By pushing the adjacent multi-function push-button on the right side, the frequency is high-lighted by a white box (tune window), then the frequency can be changed with the tuning knobs in the range from 190 and 1799 / 2088 and 2094 / 2179 and 2195 kHz. The standby frequency can also be selected between the frequencies of the 20 preset channels. By pushing again the adjacent multi-function push-button, the frequency is swapped as active frequency (see ref. 6).
3	VOR standby channel indicator	Shows the standby channel number selected with the tuning knobs when the tuning mode is set to PRESET (see ref. 14). The standby channel can be swapped as active channel with the adjacent multi-function push-button. 20 preset channels are available (see ref. 11). **NOTE** When the frequency associated with a channel is swapped as active frequency, the standby channel indicator shows the "RCL" legend. The previous active frequency remains associated with the standby channel. This frequency can become again the active frequency by selecting the "RCL" legend in the standby channel indicator.

Key to Figure 7-115 (Continued)

Ref	Control/Display	Function
4	DME frequency indicator	Shows the frequency on which the DME is tuned to compute the distance to the VOR/ DME station. The letter H, if present, indicates that the DME-H function was selected on the RTU panel or better that DME indication is held referred to the frequency shown before the H letter, even if the active VOR frequency is changed.
5	VOR active channel indicator	Shows the active channel number selected with the tuning knobs when the tuning is set to PRESET (see ref. 14).
6	VOR active frequency indicator	Shows the VOR active frequency. By pushing the adjacent multi-function push-button on the left side, the frequency is high-lighted by a white box (tune window), then the frequency can be changed with the tuning knobs in the range from 190 and 1799 / 2088 and 2094 / 2179 and 2195 kHz.
7	MKR SENS indicator	Shows the marker beacon reception sensibility. With the adjacent multi-function push-button it is possible to select the marker beacon reception sensibility between: – LO - Low sensibility – HI - High sensibility. The selected sensibility is shown in larger letters.
8	TEST identifier	By pushing the adjacent multi-function push-button on the right side, it is possible to perform the test of the VOR/ILS/MKR receiver. The TEST identifier is shown in larger letter while the test is performed. At the end of the test the CODE annunciator and two digit hexadecimal code are shown below the TEST identifier. The code indicates the test result. The code is 00 if no failures are found.

Key to Figure 7-115 (Continued)

Ref	Control/Display	Function
9	RETURN legend	By pushing the adjacent multi-function push-button on the left side, the Menu on the display comes back to TLP1 page.
10	PRESET PAGE legend	By pushing the adjacent multi-function push-button on the left side, it is possible to access the preset page for the programming of the preset channels.
11	Preset channel programming indicator	Shows the channel number and the associated frequency value. By highlighting the channel number with the adjacent multi-function push-button, it is possible to associate in the memory a frequency value selected with the tuning knobs. 20 channels can be programmed. The preset channels are distributed on five pages. The other fields are the same on all the five pages.
12	Page number indicator	Shows which page of the five preset pages is shown. When the indicator is selected by pushing the adjacent multi-function push-button on the right side, it is possible to scroll the page number by turning one of the two tuning knobs.
13	VOR active frequency indicator	Shows the VOR active frequency used on the receiver.
14	VOR tuning mode indicator	Shows the active tuning mode. By pushing the adjacent multi-function push-button on the left side, it is possible to change the tuning mode. The active tuning mode is shown in large letters. The selectable tuning modes are as follows: – FREQ: The receiver is tuned on the selected VOR frequency. – PRESET: The receiver is tuned on the frequency associated with one of the preset channels.

Key to Figure 7-115 (Continued)

Ref	Control/Display	Function
15	Vertical deviation indicator	Shows the deviation of the helicopter from the flight path when the glideslope signals are valid. When the needle is within one dot of the vertical deviation scale, the other dots are empty.
16	VNAV source indicator	Shows the source of signals for the vertical navigation of the helicopter. The GS1 legend is shown at the bottom of the vertical deviation scale on the PFDs. The GS1 legend is shown at the top of the vertical deviation scale on the MFDs.
17	Inner marker beacon indicator	I - Shown when the helicopter flights over a inner marker beacon.
18	Middle marker beacon indicator	M - Shown when the helicopter flights over a middle marker beacon.
19	Outer marker beacon indicator	O - Shown when the helicopter flights over an outer marker beacon.
20	Course/Localizer deviation indicator	Shows the deviation of the helicopter from the set course or localizer when the VOR/ILS signals are valid. When the needle is within one dot of the course deviation scale, the other dots are empty. When the needle is at a scale end position, it flashes. A magenta triangle shows the TO/FROM indication. When the triangle points up, it is a TO indication. When the triangle points down, it is a FROM indication.
21	NAV source indicator	Shows the VOR1 or LOC1 legend when the set navigation source is the VOR1/ILS system.
22	VOR2 pointer	Shows the bearing of a VOR station.
23	Mode indicator	Shows the mode of the data presentation. The modes applicable to the VOR1/ADF system are: VOR1 and ILS1.
24	TO/FROM indicator	Shows if the helicopter flights to or away from the set VOR/ILS station.

Key to Figure 7-115 (Continued)

Ref	Control/Display	Function
25	Localizer deviation indicator	Moves left or right to show the left or right deviation from the localizer.
26	Course deviation indicator	Moves left or right to show the left or right deviation from the set course.
27	Course indicator	Shows the course set with the adjustment knob.
28	Glideslope deviation indicator	Moves up or down to show the up or down deviation from the glideslope.

DME SYSTEM

The DME system consists of a DME transceiver Rockwell Collins type DME-4000, installed in the bay below the aft avionic compartment, and an antenna, installed on the bottom of the tail boom.

The DME system is controlled through the RTU panel of the RMS, on which the TLP1 page must be selected to access the related NAV1 control display or the TLP2 page must be selected to access the NAV2 control display.

Because the DME system uses channels conventionally paired to VOR/LOC channels, the DME transceiver automatically tunes on the related paired channel, when the desired VOR/LOC frequency is tuned on the preferred NAV control display.

The transceiver is able to manage in the same time three of the 252 available channels in the frequency range of 960 to 1215 MHz.

When a VOR or ILS frequency is selected on the NAV1 control display, the first channel tunes automatically on the frequency of the related DME station.
The same occurs for the second channel tuned on the frequency selected on the NAV2 control display.
The third channel is managed by the FMS function of EFIS instruments to tune the DME stations selected by the FMS function and used for the triangulation of the position.

For each channel the DME transceiver processes and provides the information of slant distance, time-to/from-station, station identifier, also when the DME is installed in an ILS station.

The maximum range of the DME is 300 nautical miles; this value is however variable and limited by the helicopter altitude, obstructions such as hills or mountains and the curvature of the earth.
The accuracy of the shown distance value is of 0.1 nautical miles from 0 to 199.9 nautical miles and of 1 mile from 200 to 300 nautical miles.

On the RTU panel it is possible to activate the DME-H function, which permits to hold a transceiver channel paired to the old selected VOR frequency, without following the new frequency selected for the in-use VOR station.
If the signal corresponding to the selected frequency or channel is not valid, dashes are shown on the control display to avoid false indications.

If the signal is weak or momentarily lost, the internal memory of the DME transceiver avoids that the dashes are shown for a duration of 10 to 14 seconds.

The audio signal of the DME station is sent to the ICS so as the identifier code of the ground station, transmitted every 30 seconds, can be heard in the pilot headset.

The DME system is supplied with electrical power from the 28 V dc Number 2 auxiliary bus through the DME circuit breaker.

DME SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-116.

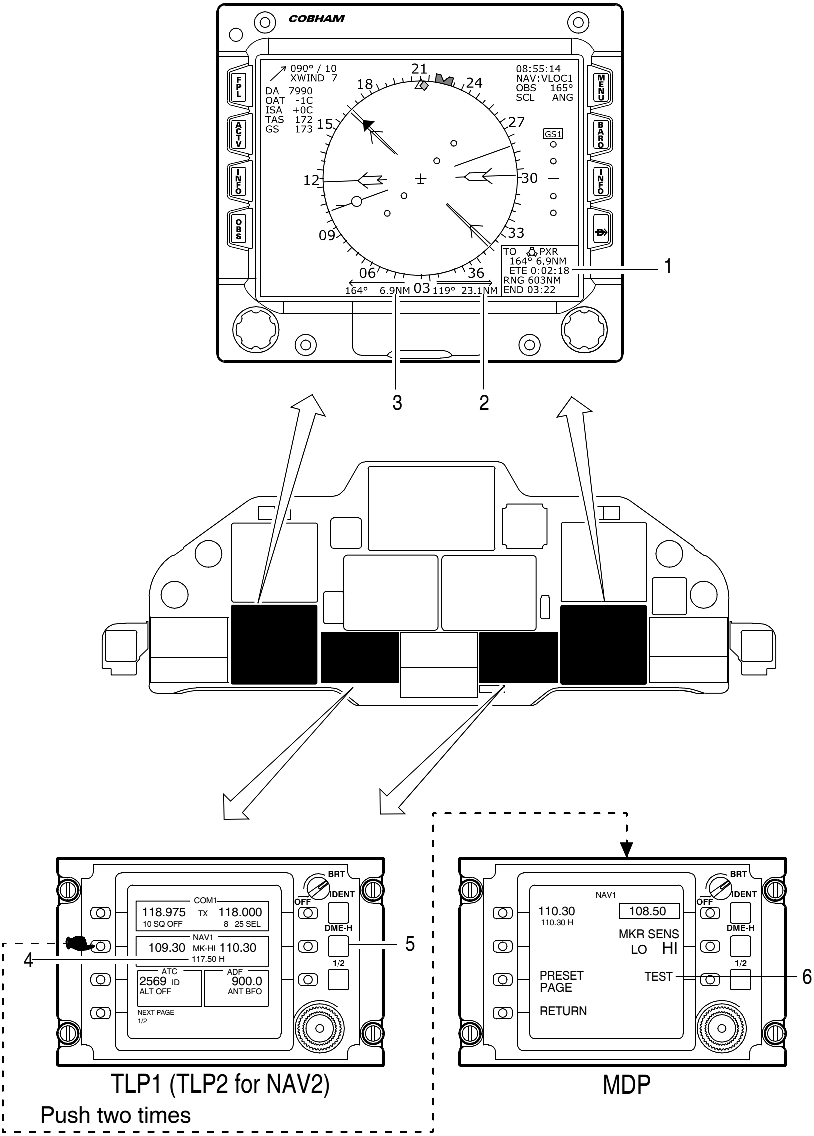


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Figure 7-116 DME System - Controls and Displays

Key to Figure 7-116

Ref	Control/Display	Function
1	ETE indicator	Indicates the time necessary to reach the selected VOR/DME station
2	Distance readout (NAV2)	Indicates the distance from the VOR/DME station set on NAV2 control display. If the DME channel is in hold mode (see ref. 5), the related distance readout is shown in yellow and the letter H is shown above the distance readout.
3	Distance readout (NAV1)	Indicates the distance from the VOR/DME station set on NAV1 control display. If the DME channel is in hold mode (see ref. 5), the related distance readout is shown in yellow and the letter H is shown above the distance readout.
4	DME frequency indicator	Shows the frequency on which the DME is tuned to determine the distance from the VOR/DME station. The letter H, if present, indicates that the DME-H function has been selected on the RTU panel (see ref. 5)
5	DME-H push-button	Pushed - Operates the DME-H function. This function holds active the DME frequency on which the selected VOR receiver was tuned. When a new frequency is set to tune a new VOR station, the old frequency remains active to supply the DME data. The old frequency is shown in the DME frequency indicator (see ref. 4). The DME-H function operates, until the DME-H push-button is pushed again.

Key to Figure 7-116 (Continued)

Ref	Control/Display	Function
6	TEST identifier	By pushing the adjacent multi-function push-button on the right side, it is possible to perform the test of the VOR/ILS/MKR and DME receivers. The TEST identifier is shown in larger letter while the test is performed. At the end of the test the CODE annunciator and two digit hexadecimal code are shown below the TEST identifier. The code indicates the test result. The code is 00 if no failures are found.

ATC TRANSPONDER SYSTEM

The Air Traffic Control (ATC) transponder system consists of a transponder Rockwell Collins type TDR-94, installed in the forward avionic compartment, and an antenna, installed on the bottom of the fuselage nose.

The system provides the identification signal and the altitude of the helicopter when interrogated by a radar station of the Air Traffic Control Radar Beacon System (ATCRBS).

The transponder is interfaced with the RMS and the ADUs through ARINC 429 busses.
It is also connected to the RCP for the selection of the ADU as source of altitude data.

A discrete suppression signal is exchanged with the DME system.

The ATC transponder system is supplied with electrical power from the 28 V dc Number 1 emergency bus through the XPDR circuit breaker.

Operation of ATC transponder system

The ATC transponder system is controlled through the RTU panel of the RMS, on which the TLP1 or TLP2 page must be selected to access the related ATC control display.
By pushing twice the multi-function push-button on the left side of the panel and adjacent to the ATC control display, it is possible to access the MDP page.

The sequence of the control pages on the displays is shown in Figure 7-117.

The transponder can operates in the modes that follow:

– mode A
– mode C
– mode S.

When the transponder receives a mode A interrogation from a ground radar facility, it transmits a coded reply which is shown on the display of the ground radar permitting the air traffic controller to identify the helicopter and to determine its position.

The coded reply sent by the transponder consists of a four digit identification number (octave-based), variable from 0000 to 7777, which is assigned by the air traffic controller and entered in the transponder by the pilot through the RTU panel.

When the transponder is interrogated in the mode C by a qualified station, it transmits a reply with also the helicopter altitude information, provided to the transponder by the selected ADU.

The transponder can also operate in the mode S.
In this mode the transponder must provide an identification code unique to each individual helicopter (assigned to each transponder and thus to each helicopter) in response to interrogations from ground station or airborne equipment.

The transponder is also able to operate as data link; this allows the transponder to perform additional functions of Air Traffic Control (ATC) and of Aircraft Separation Assurance (ASA).

The assignment of the unique identification code is done by connecting the transponder aft connector in a predetermined configuration.
An another connection determines the maximum speed parameter.

The mode S operations also require that the transponder is connected to the WOW switch so as the flight condition of the helicopter (on ground/in flight) is determined.

ATC TRANSPONDER SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-118.

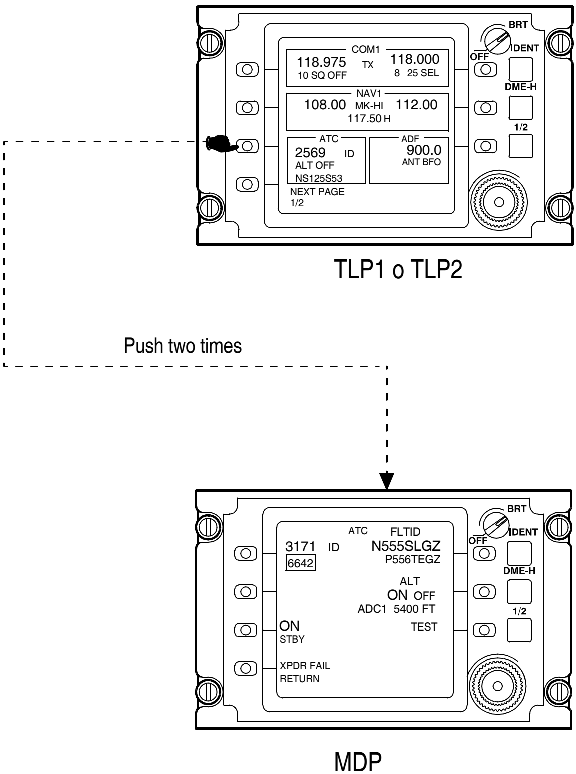

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Figure 7-117 Sequence of ATC Transponder System Control Pages on RTU Panel

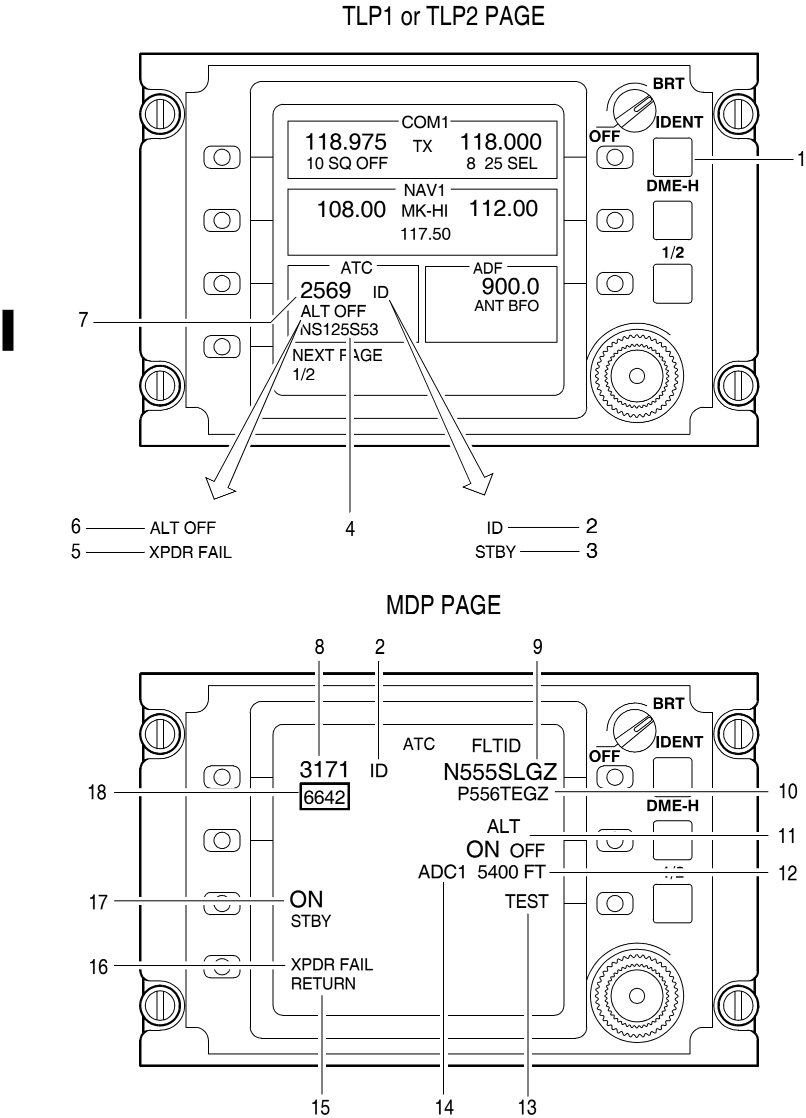

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Figure 7-118 ATC Transponder System - Controls and Displays

Key to Figure 7-118

Ref	Control/Display	Function
1	IDENT push-button	Pushed - Causes the helicopter identification code to be transmitted in response to the radar station interrogation. During the transmission the ID annunciator is shown on the ATC control display (see ref. 2).
2	ID annunciator	Shown for some seconds during the transmission of the helicopter identification code in response to the radar station interrogation.
3	STBY annunciator	Shown when the ATC transponder is in standby (see ref. 17).
4	Identification code indicator	Shows the helicopter identification code. The code consists of eight alphanumeric characters, white space included. It can be changed by selecting it with the adjacent multi-function push-button and then by turning the tuning knobs. The code is shown in green color when the transponder is active or in white color when it is in standby.
5	XPDR FAIL message	Shown when the CBIT detects a failure in the transponder. This message has the priority respect to the ALT OFF message (see ref. 6).
6	ALT OFF message	Shown when the function of the altitude transmission (mode C) is not active (see ref.11).
7	Active code indicator	Shows which code is active for modes A and C. The code is a octave-based number, consisting of four digits from 0000 to 7777. It can be changed by selecting it with the adjacent multi-function push-button and then by turning the tuning knobs. The outer knob permits to change the first two digits, the inner knob the other two digits. The code is shown in green color when the transponder is active or in white color when it is in standby.

Key to Figure 7-118 (Continued)

Ref	Control/Display	Function
8	Active code indicator	Shows which code is active for modes A and C. The code is a octave-based number, consisting of four digits from 0000 to 7777. The code is shown in green color when the transponder is active or in white color when it is in standby. **NOTE** The active code cannot be changed but only swapped with the standby code (see ref. 18).
9	Active identification code	Shows the helicopter active identification code. The code consists of eight alphanumeric characters, white space included. This code cannot be changed in this menu page. This code can only be interchanged with the standby identification code when the adjacent multi-function push-button is pushed. The code is shown in green color when the transponder is active or in white color when it is in standby
10	Standby identification code indicator	Shows the helicopter standby identification code. The code consists of eight alphanumeric characters, white space included. It can be changed by selecting it with the adjacent multi-function push-button and then by turning the tuning knobs. The code is shown in green color when the transponder is active or in white color when it is in standby.

Key to Figure 7-118 (Continued)

Ref	Control/Display	Function
11	ALT indicator	Shows if the function of the altitude transmission is active or not. The condition (ON, OFF) can be selected by pushing the adjacent multi-function push-button. The active condition is highlighted in larger characters
12	Altitude indicator	Indicates the altitude value detected by the selected ADU through the RCP. The altitude is shown when the mode C is activated and the selected ADU transmits valid data. If the data are invalid, the altitude is replaced by white dashes. The altitude value is shown in green color when the transponder is active or in white color when it is in standby.
13	TEST identifier	By pushing the adjacent multi-function push-button on the right side, it is possible to perform the IBIT of the ATC transponder. The TEST identifier is shown in larger letter while the test is performed. At the end of the test the CODE annunciator and two digit hexadecimal code are shown below the TEST identifier. The code indicates the test result. The code is 00 if no failures are found.
14	ADC annunciator	Shows which ADU (ADU1 or ADU2) is selected on the RCP.
15	"RETURN" legend	By pushing the adjacent multi-function push-button on the left side, the Menu on the display comes back to TLP1 page.
16	XPDR FAIL message	Shown when the CBIT detects a failure in the transponder.

Key to Figure 7-118 (Continued)

Ref	Control/Display	Function
17	ON/STBY condition indicator	Shows if the transponder is active or in standby. By pushing the adjacent multi-function push-button, the transponder changes from the ON condition to the STBY condition and vice-versa.
18	Standby code indicator	Shows which standby code is set for modes A and C. The code is a octave-based number, consisting of four digits from 0000 to 7777. It can be changed by selecting it with the adjacent multi-function push-button and then by turning the tuning knobs. The outer knob permits to change the first two digits, the inner knob the other two digits. The standby code can be swapped with the active code (see ref. 8).

DIGITAL MAP GENERATION SYSTEM

The Digital Map Generation (DMG) system Euroavionics type EuroNav V provides the flight crew with a visual moving map to enhance situational awareness.
The digital map generation system consists of a processor type RN6, a processor mounting tray, an USB connector panel and a DMG control panel. The processor, the mounting tray, the USB connector panel are installed in the aft avionic compartment. The DMG control panel is installed in the interseat console.
The processor provides the interface to all system components and contains a 32 Gb flash Disk Memory. This disk memory stores the dedicated configuration software and the vector/raster/DTED maps.
By removing the front panel of the processor, it is possible to access to two hard disks, two PCMCIA cards, the GPS battery and the battery fuse.
The DMG system is interfaced with the EFIS instruments through an ARINC 429 bus. From the EFIS instruments the DMG system receives the helicopter data (position, ground speed, altitude) and the active flight plan.
The digital moving map images and the navigation data are displayed on the cockpit monitor.
The system is supplied with electrical power from the 28Vdc Number 2 main bus through the DMG PWR circuit breaker.

DMG system databases

The operation of the DMG system requires many databases that must be stored in the hard disk memory of the processor. This can be done by unloading the data through the USB port on the USB connector panel, in the aft avionic compartment.

MAPPING DATABASE

The mapping database includes either vector or scanned raster maps of the entire operation area and its surroundings. The map can have different scales from 1: 2.000.000 to 1:5.000. All maps can be shown in full color and in North-up, Track-up or Heading-up modes with the helicopter position set at bottom or at center of the screen. As the moving map is linked to helicopter position, the current position is constantly updated. The map can be zoomed in or out in small increments. Map labels move with the map, but remain upright for easy

reading. In case of need or in IFR conditions, only the aeronautical information can be shown on the screen.
The terrain maps are of raster type and can appear in different color modes with dynamic sun shading and ground proximity warnings. They are based on DTED or other digital terrain data.

JEPPESEN DATABASE

Jeppesen charts also can be used. They show airspace structures, airports information such as runways alignment, radio frequencies, navigation aids and restricted airspace.

ADDRESS DATABASE

A full geographic database with a complete Town List for the country or application can be stored. Any address can be viewed by selecting a position on the map.

USER DATABASES

Up to six user databases, of unlimited size, can be used.

Map overlays

The digital moving map can be overlaid with various flight data described below.

WAYPOINTS

The waypoints are normally used to hold Town reference points, hospital, landing sites, points of interest, drill platform, farms etc. The waypoint data can be stored in six user databases as permanent or temporary waypoints (all temporary waypoints of one database can be deleted with one command).
Each waypoint is identified by a name/number and its Latitude and Longitude coordinates. Each waypoint has a parameter setting of show or hide (they can be faded in or out). Each waypoint can be displayed as NATO mil symbology as well as user defined bitmap symbol. They can have up to three range rings and be warning sensitive.

FLIGHT PLANS

The flight plans can be overlaid as received from the EFIS instruments or created with the DMG system. The system offers two possibilities of creating a Direct To plan, the other one is to use the Flight Plans editor for more complex flight planning. An info window is displayed at the

lower left frame containing indicators such as waypoint name, bearing, distance, ETE,ETA and DTK.

OBJECTS

Objects are normally used to mark warning or danger area, wires, obstacles, etc. The pre-setup in the object editor allows up to 5000 objects be configured in the system database.

SEARCH PATTERNS

The search patterns are in principal normal routes with predefined selectable shapes, such as Rising Ladder, Race Track, Orbit, Sector Search, Expanding Square.

GEOGRAPHICAL DATA

By selecting a position om the map, the related geographical data can be viewed. A destination can be selected by Town name, street name and house number.

RADIALS / DIRECTION FINDER

Radials are lines from a selected point with a selected bearing and length. Three radials can be defined for each flight plan bank. The line of a direction finder that is linked to the system can be shown and fixed as a radial.

MOVING OBJECTS

Up to 200 moving objects (ground vehicles, aircraft, vessels, etc.) can be shown on the screen. Moving objects are fed into the system by manual setup. They can have various symbols and colors. A speed vector shows heading and speed of movement. Moving objects can be used as waypoints in flight plans. Interception points are calculated.

CAMERA FOOTPRINT / LINE OF SIGHT

The camera viewing area can be shown as a footprint or as a line of sight in the map. Optionally the system can show the coordinates of the camera look-to-point and pull the street name next to the picture center from the vector map.

ESCAPE RINGS

Escape-Rings give an idea of the area a moving target (person, vehicle) could be found in. Three escape-rings (white, green, yellow) are drawn into the map around the selected destination point. The growth

speed can be set according to the specific needs using the Escape Rings Setup menu.

TERRAIN COLLISION WARNING

When the system contains an installed terrain database (DTED), collision warning can be shown provided that the related function has been activated through the Terrain Warning menu.

Operation of digital map generation system

The DMG system lets the flight crew perform many functions related to the flight management as follow:

– Flight Planning: The flight can be planned and previewed on the screen. The DMG system has the capability to overlay flight plans received from the EFIS instruments, to show the user defined waypoints and to transmit the user defined waypoints to the EFIS instruments. Each flight plan can be changed during the flight and all the planned flights can be stored and reactivated on demand.
– Flight Planning Info: The DMG system can show all the flight information by input of the ICAO letter code, coordinates or name. When the helicopter is in flight, the GPS1 or the GPS2 provide the position and the distance to airports, VORs or any other point in the database.
– Direct-To Navigation: The DMG system can show a line from the helicopter or/and the start point to the destination point.
– Searching and Marking: The DMG system allows searching for any particular place, right down to a street or an object by interrogating all available databases. Once the place has been found, immediately a Direct To can be set to fly directly to the scene. Colored flags can be used to pinpoint anything.
– Object and Terrain Warning: Objects like towers, pipelines, lifts etc. can be integrated into the map and can be set to a warning mode, issuing a warning when the helicopter reaches the defined warning range. During flight this function can be set to off. When the warning function is activated, the system automatically searches in the map for objects to which the warning mode has been assigned. A certain section area ahead of the aircraft symbol and next to it is continuously searched through to detect such objects. In case the helicopter approaches an obstacle, the COLL

warning message, shown in the lower frame, flashes white and grey.
Terrain collision warnings can also be activated, if the system contains an installed terrain database (DTED data).

- Elevation mode: The system can operate in the Elevation Mode if a terrain database (DTED) has been installed. When in elevation mode a terrain profile window is shown in the frame upper right corner indicating the vertical terrain profile in front of the helicopter. The profile height, width and length and the color coding can be set through the DTED Setup menu.
- Emergency mode: In case of an emergency the emergency function shows the pilot the direct track to the nearest suitable landing field together with the associated navigational information about the displayed location that enables prompt reaction according to the system emergency checklist.

DIGITAL MAP GENERATION SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-119.

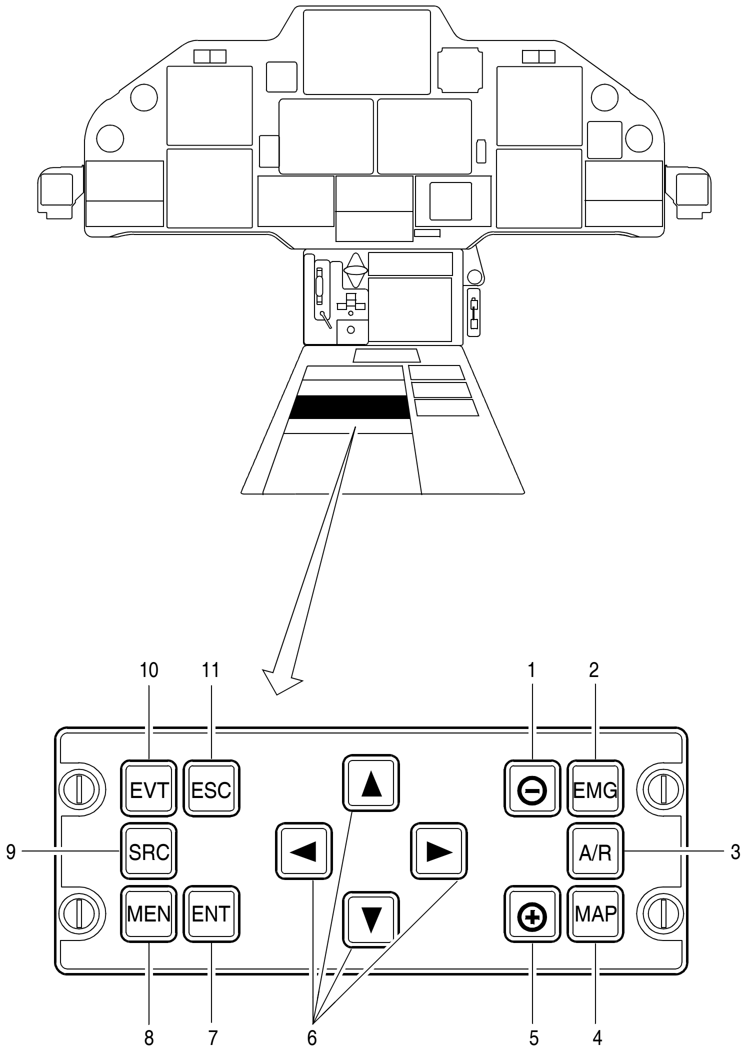

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Figure 7-119 Digital Map Generation System - Controls and Displays (Sheet 1 of 2)

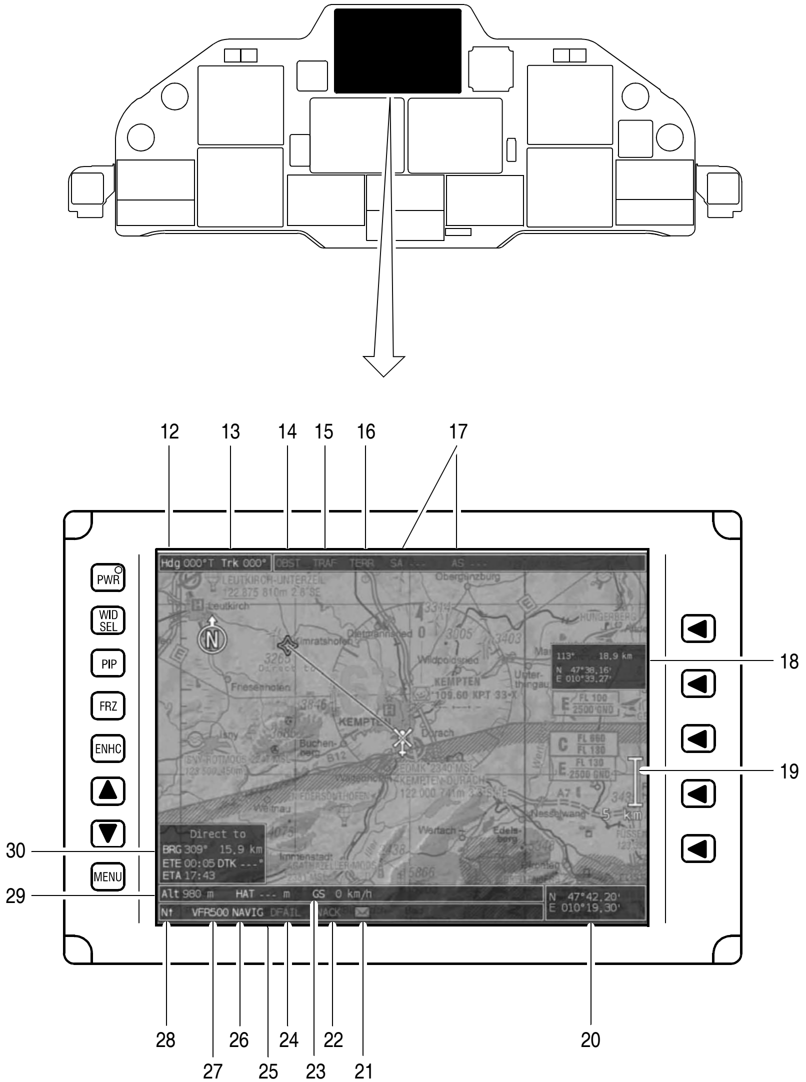

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Figure 7-119 Digital Map Generation System - Controls and Displays (Sheet 2 of 2)

Key to Figure 7-119

Ref	Control/Display	Function
1	Zoom out key	Pushed - Decreases the zoom of the screen.
2	EMG key	Pushed - Gets access to the data and sale modes that follow: – Emergency Mode – Emergency waypoints or Address Points.
3	A/R key	Pushed - Gets access to the submenus that follow: Rose, Range Rings, Track Line, Panning, Bearing Line, Flight Trace, Rectangular Grid and Range Arc.
4	MAP key	Pushed - Gets access to the submenus that follow: DTED Profile, Map Line, Labels, Maps, Map Mode, Map Design.
5	Zoom in key	Pushed - Increase the zoom of the screen.
6	Arrow keys	The keys are used for the map, menu and text window navigation.
7	ENT key	Pushed - Enters next or previous waypoint.
8	MEN key	Pushed - Gets access to the submenus that follow: Flight Plan, Flight Info, Checklists, I/O Status, Flight Recorders, Data Com, Maintenance, Shut Down. Pushed during character insertion - Shows a virtual keyboard on the screen to make easier the data insertion.
9	SRC key	Pushed - Gets access to the submenus that follow: View Next Waypoint, Search Map, Search Database, Auto Line, Search Air Info.
10	EVT key	Pushed - Gets access to the submenus that follow: Record Event, User Database, Quick Edit, Point Types, Vector Types, Radials, Escape Rings, Screenshot.
11	ESC key	Pushed - Gets access to the Quick Access Menu. If pushed when a menu is already shown, the key performs the abort/escape procedure.

Key to Figure 7-119 (Continued)

Ref	Control/Display	Function
12	Heading indicator (Hdg)	Shown on the monitor only when selected through HDG/TRK menu.
13	Track indicator (Trk)	Shown on the monitor only when selected through HDG/TRK menu.
14	Obstacle warning function indicator (OBST)	Blinks white/red when there is an impending obstacle collision. It is grey when the function is deactivated.
15	Traffic advisory system indicator (TRAF)	Used for presentation of traffic advisory system applications.
16	Terrain awareness and warning system indicator (TERR)	Is white when the TAWS operates. It blinks white/red in case of an impending terrain collision. It is grey when the function is deactivated.
17	SA / AS indicator	Blinks in case of approaching Special Area (SA) or Air Spaces (AS) and shows the related denomination.
18	Additional coordinate window	In the Panning mode, shows the information that follow: – The Latitude and Longitude of the map center point – The distance between the helicopter and the map center point – The bearing of Direct To track toward the map center point (0° indicates North).
19	Map scale	Shows the current map scale. It is deactivated when the Range Rings option of the ARC/ Rose menu is selected.
20	Position window	Shows the Latitude (North) and Longitude (East) coordinates of the helicopter.
21	Message indicator	Does not operate
22	Not Acknowledged indicator (NACK)	Does not operate
23	Ground speed indicator (GS)	Shows the ground speed value.

Key to Figure 7-119 (Continued)

Ref	Control/Display	Function
24	Fail indicator (DFAIL)	Is white when a data communication fail occurs. It is gray when there are no communication problems.
25	Height Above Terrain (HAT) indicator	Shows the height value above the terrain.
26	Navigation vector indicator (NAVIG)	The indication is white when the navigation vector operates. It is grey when the navigation vector is deactivated.
27	Map indicator	Shows the selected map in white color.
28	Map mode indicator	Shows the mode selected through the Map Mode menu. One of the indicators that follow is shown: – N↑ North Up mode – T Track Up mode – H Heading Up mode.
29	Altitude indicator	Shows the altitude value of the helicopter.
30	Direct To / Next Waypoint Window	Shows the data related to the next waypoint of the flight plan or to the Direct To destination. The data that follow are shown: waypoint name, bearing, distance, ETE, DTK and ETA. When the Direct To navigation operates, the “Direct To” label replaces the waypoint name.

DOORS

(Chapter 52)

GENERAL

The helicopter has the following doors:

– The passenger/crew compartment doors
– The baggage compartment door
– The service doors.

Refer to Figure 7-120 for the location of the doors.

PASSENGER/CREW COMPARTMENT DOORS

This compartment includes the cockpit doors for crew access and the cabin doors. Four sensor switches, connected to the DAU, send the signal that causes the CABIN DOOR caution message to come in view on the EDU when a door is not closed and locked.

COCKPIT DOORS

Refer to Figure 7-121.

The two cockpit doors are identical and each consists of a composite material structure containing a window. Each door is attached to the structure with two fittings. The hinge-pin of each fitting is connected to an emergency door release mechanism which is actuated by a jettison handle located on the windshield side post. The handle permits a quick-release of the door by extracting the pins from the hinges.

The door latching mechanism consists of two handles (one internal and one external handle) that operate a roller assembly to secure the door to the airframe. Each cockpit door is provided with a strut that holds the door open when necessary.

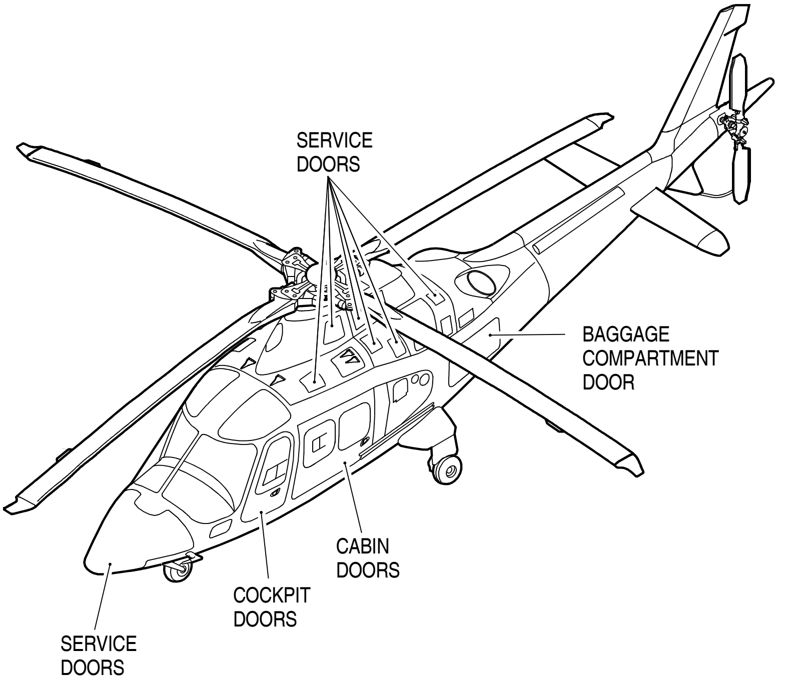

Figure 7-120 Doors Location

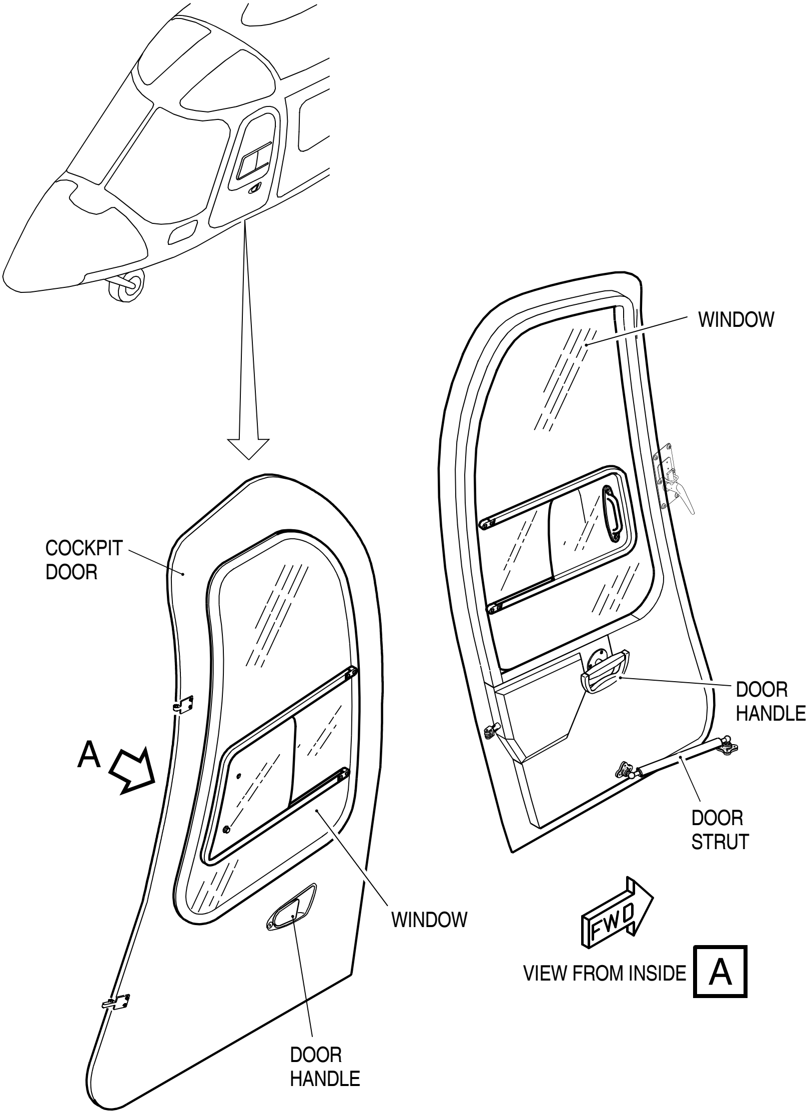

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Figure 7-121 Cockpit Doors

PASSENGER CABIN DOORS

Refer to Figure 7-122.

Two large sliding doors are fitted, one on each side of the forward fuselage, that provide access to the passenger/cargo compartment. The doors are attached to two tracks allowing the doors to be slid aft. A latching mechanism enables the doors to be locked fully open. A lever, located on the aft end of the door frame, releases the mechanism when pressed inward.
The door latching mechanism consists of two handles (one internal and one external handle) that operate a roller assembly to secure the door to the helicopter structure.

Two windows are installed in the door structure by means of a rubber seal. In the event of an emergency, the windows can be released by pulling the red strap fitted at the internal edge of each window.
As optional, the open pax windows can be installed on the helicopter instead the forward windows. Refer to Figure 7-122.
The two hinges hold the open pax window in its position on the structure. The opening is inward and the maximum opening angle is 105 degrees. The four rotating blocks lock the window in the closed position. The handle stop and the spring clip hold the window in the open position.

Passenger sliding door(s) opening in flight is prohibited if passenger door modification P/N 109-0814-35 is not installed.

BAGGAGE COMPARTMENT DOOR

The baggage compartment door is installed on the left side of the fuselage. The door is hinged to the fuselage structure in two points. The door is held closed by two push-button latches and is provided with a key-operated lock.
A sensor switch, connected to the DAU, send the signal that causes the BAG DOOR caution message, to come in view on the EDU when a door is not closed and locked.

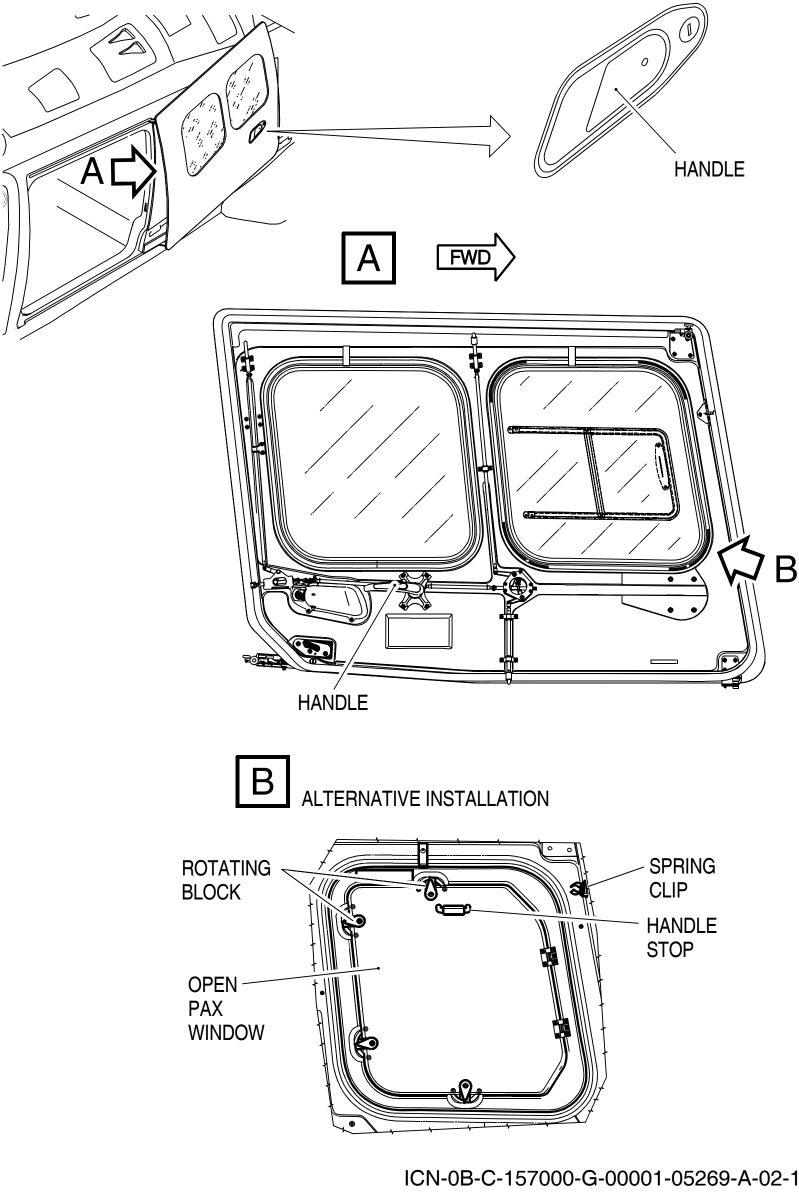

Figure 7-122 Passenger Cabin Doors

SERVICE DOORS

Service doors on the helicopter consist of access panels, cowlings and fairings. The function of the service doors is to provide access to equipment for maintenance and inspection.
The nose door and the external power receptacle door have sensor switches that send the signal that causes the NOSE DOOR and UTIL DOOR caution messages to come in view on the EDU when the related door is not closed and locked.

CABIN FOLDABLE FOOTBOARDS

Two cabin foldable footboards are installed on the two sides of the fuselage in correspondance of the passenger cabin doors. They help the persons to go in and out the cabin. The system is composed by the left and right footboard and two actuators for each footboard.

The pilot controls the extension and retraction of the left and right cabin footboard through two lighted push-button switches placed on the interseat console. When the cabin foldable footboards are fully extended, the related indication on the lighted push-button switch is on. The cabin footboards retract automatically after the helicopter takeoff.

Refer to Figure 7-122A.

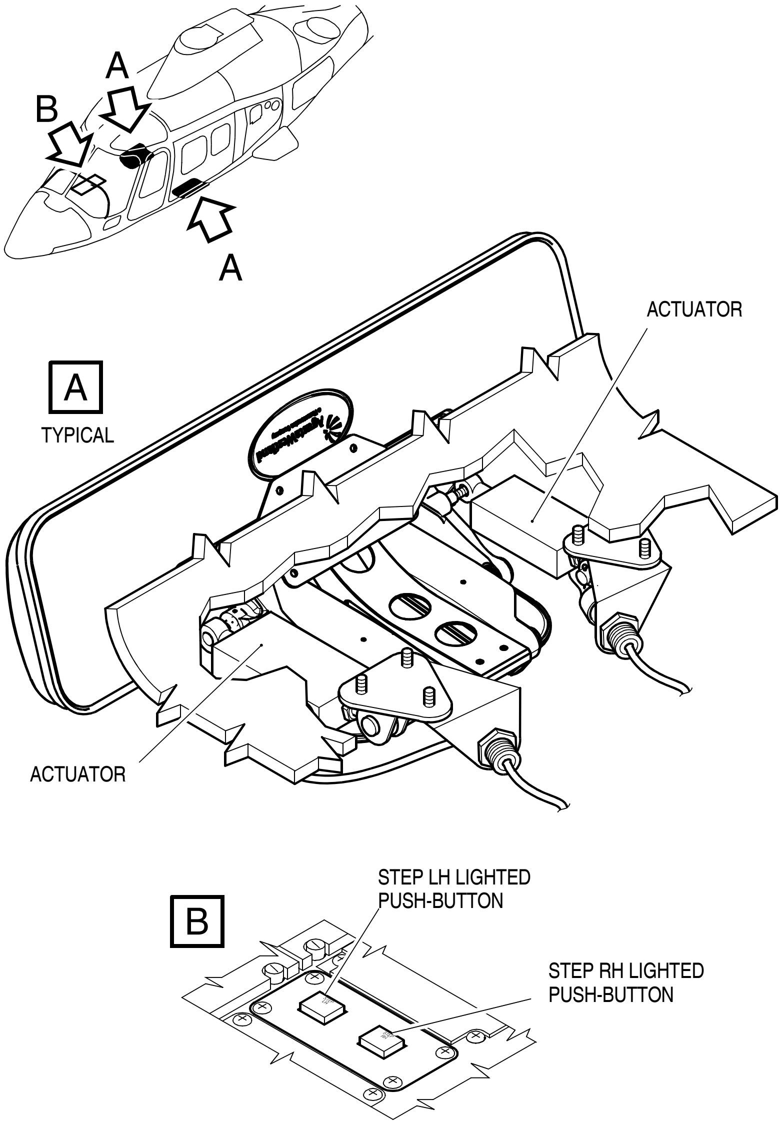

Figure 7-122A Cabin Foldable Footboards - Component Location

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MAIN ROTOR

(Chapter 62)

GENERAL

The main rotor system includes:

– The main rotor blades
– The main rotor head
– The rotating controls
– The main rotor indicating system.

Refer to Figure 7-123 for the general layout of the main rotor system.

MAIN ROTOR BLADES

Refer to Figure 7-124.

The main rotor blades are of composite material. Each blade consists of:

– A fiberglass spar
– A trailing edge made of a graphite-fiber skin and a Nomex core
– A stainless steel abrasion strip attached to the leading edge
– A fiberglass tip cap with a nickel anti-abrasive strip.

The blades are statically balanced during construction by means of weights fitted to the blade end and trailing edge.
The blade retention bolts are hollow and allow the addition of weights during hub balancing. A trim tab is bonded to the trailing edge to permit rotor tracking.

MAIN ROTOR HEAD

Refer to Figure 7-125.
The Main Rotor Head (MRH) includes:

- The hub
- The levers
- The grips
- The dampers
- The flapping
- The droop restrain mechanisms
- The elastomeric bearings.

HUB

The main rotor hub is made of titanium. It forms a quatrefoil structure (four rotor blades) internally bound through a spline and is coupled to the rotor mast by which it is driven.

LEVERS

Four aluminium alloy levers are attached to the grips with bolts. The levers permit torsional movement through two arms connected to the related pitch change link and damper.

GRIPS

The grips provide the attachment points between the blade and the rotor hub. They transmit the aerodynamic and inertial loads to the blades.

DAMPERS

The dampers provide damping of the main rotor blades during take-off and landing and during flight. The dampers provide the damping necessary to avoid a blade shock in lag-lead motion during rotor starting, rotor brake application or flight maneuvers that involve blade operation close to end stroke.

FLAPPING AND DROOP RESTRAINT MECHANISMS

The flapping and droop restraint mechanisms limit blade flapping at normal operating speed (RPM), and support the blades when the rotor is stationary. They consist mainly of two separate groups: the flap restraint mechanism and the droop restraint mechanism. The flap restraint mechanism, installed between the blade grip and the blade lever, consists of a loaded spring and inertia weight fixed to a bracket connected to the elastomeric bearing and a stop limiter installed on the blade lever. The droop restraint mechanism, installed on the lower side of the hub, consists of a ring and reference plate that can float in a groove on the hub and is attached to the blade grip.

ELASTOMERIC BEARINGS

The elastomeric bearings (one for each arm) are spherical bearings housed in the concave part of the grip and secured by bolts. They transfer the aerodynamic and inertial load of the blade to the hub, allowing flapping, lead-lag and pitch motion of the blade.

ROTATING CONTROLS

Refer to Figure 7-126.
The main rotor rotating controls consist of:

– The pitch links
– The rotating scissors
– The non rotating scissors
– The swashplate and support assembly
– The boot assembly.

PITCH CHANGE LINKS

The pitch change links connect the rotating parts of the swashplate to the root end of the blade and change the angle of the swashplate in accordance with the cyclic and collective control inputs.

ROTATING SCISSORS

The four pitch change links are installed between the control lever and the bracket of the swashplate of the main rotor head.

The rotating scissors include an upper and a lower hinged link. The lower link is attached to the main rotor hub flange bracket, while the upper link is attached to the swashplate outer ring. The rotating scissors drive the swashplate outer ring and permit the outer ring to tilt and move vertically in response to the swashplate pivot sleeve movements.

NON-ROTATING SCISSORS

The non-rotating scissors includes an upper and a lower hinged link. The lower link is attached to the transmission upper case fitting, the upper link is attached to the swashplate pivot sleeve.

SWASHPLATE

The swashplate assembly transmits the flight control controls inputs to the rotor blades. The assembly includes a support, a pivot sleeve, a non-rotating inner ring and a rotating outer ring.
The support is attached to the transmission upper case studs and provides a guide for the collective pivot sleeve. The pivot sleeve carries the swashplate rings.
The swashplate inner ring is connected to the hydraulic servo actuators. The inner ring moves vertically and/or tilts on its ball-end with the sleeve, in response to collective and cyclic control inputs. The rotating outer ring is separated from the inner ring by bearings and is connected to the pitch change links. The outer ring changes the blade pitch following the movement of the inner ring.

BOOT ASSEMBLY

The boot assembly includes the top boot, the bottom boot and the boot locking ring.
The top boot is installed above the boot support of the swashplate assembly. It is in the red silicone rubber type. The lock wire safeties the top boot above the boot support and to the swashplate. The top boot is resistant to the oils and grease. The top boot has an operation range of - 54 thru 60 °C.
The bottom boot is a protection from moisture and other contamination.
The bottom boot prevents the contamination of the areas between the swashplate and the spherical pivot.

Also, a boot support is installed on the groove of the swashplate. This support lets the extension and retraction of the top boot. This occurs when the actuators move up and down during the operation.

The boot locking ring attach the top boot to the main rotor mast. The boot locking ring has four lugs, and each lug has an attaching hole. Through this holes that four bolts and four washers connect the boot locking ring to the rotating scissor attachment flange on the main rotor head.

MAIN ROTOR INDICATING SYSTEM

Refer to Figure 7-127.

The main rotor indicating system indicates the rotational speed of the rotor in percent of the maximum rating. The system consists of the rotor RPM sensor (magnetic pick-up) installed on the main transmission and electrically connected to the pilot's EDU.

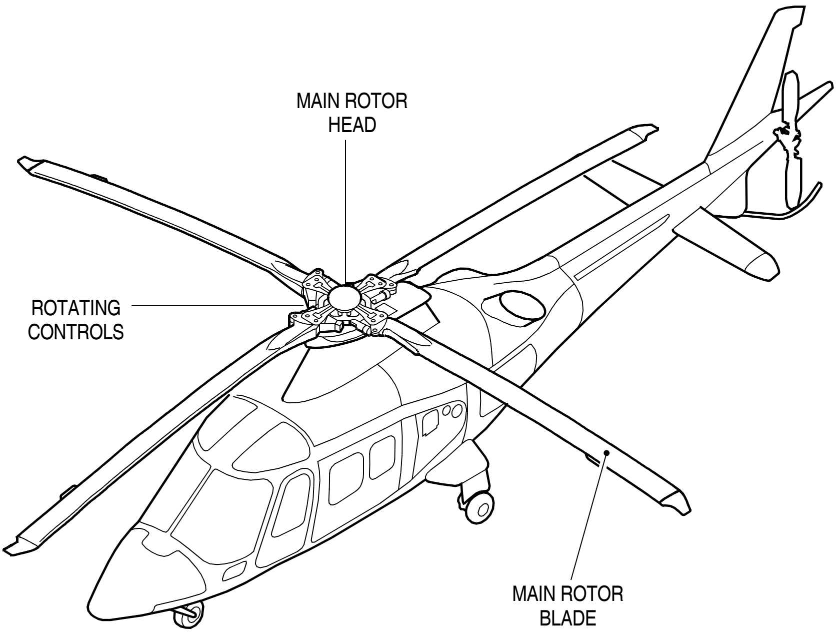

Figure 7-123 Main Rotor - Components Location

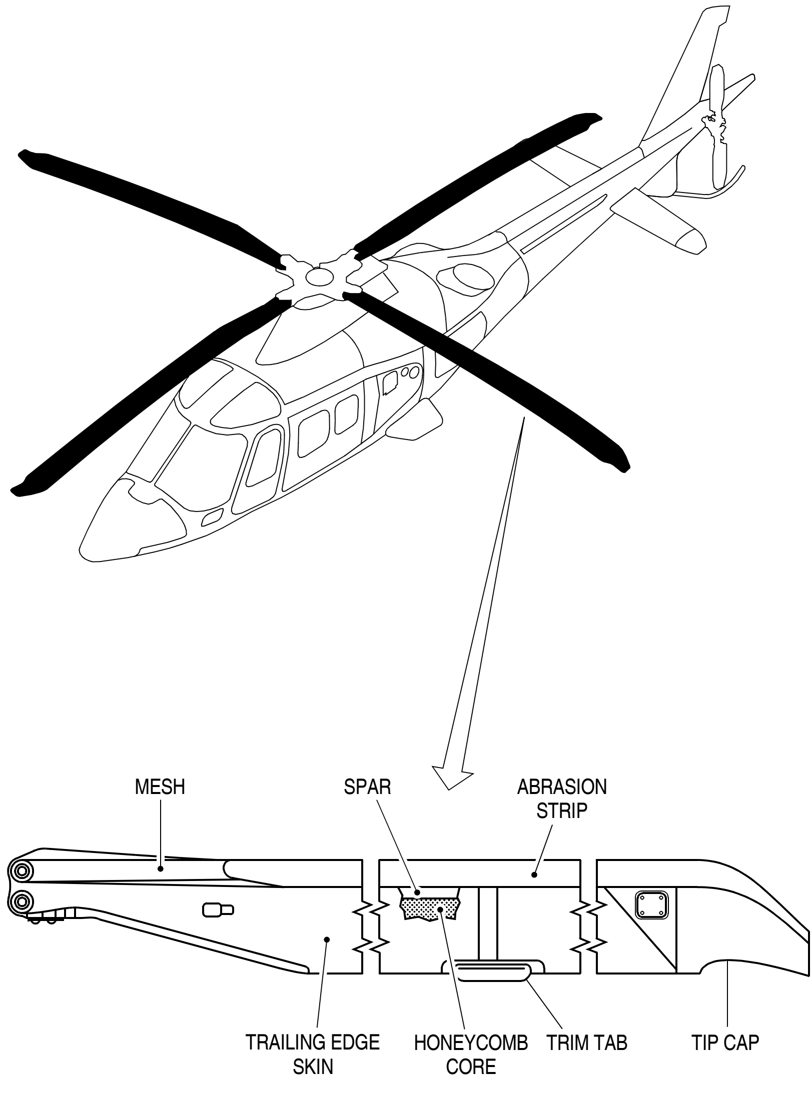

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Figure 7-124 Main Rotor Blade

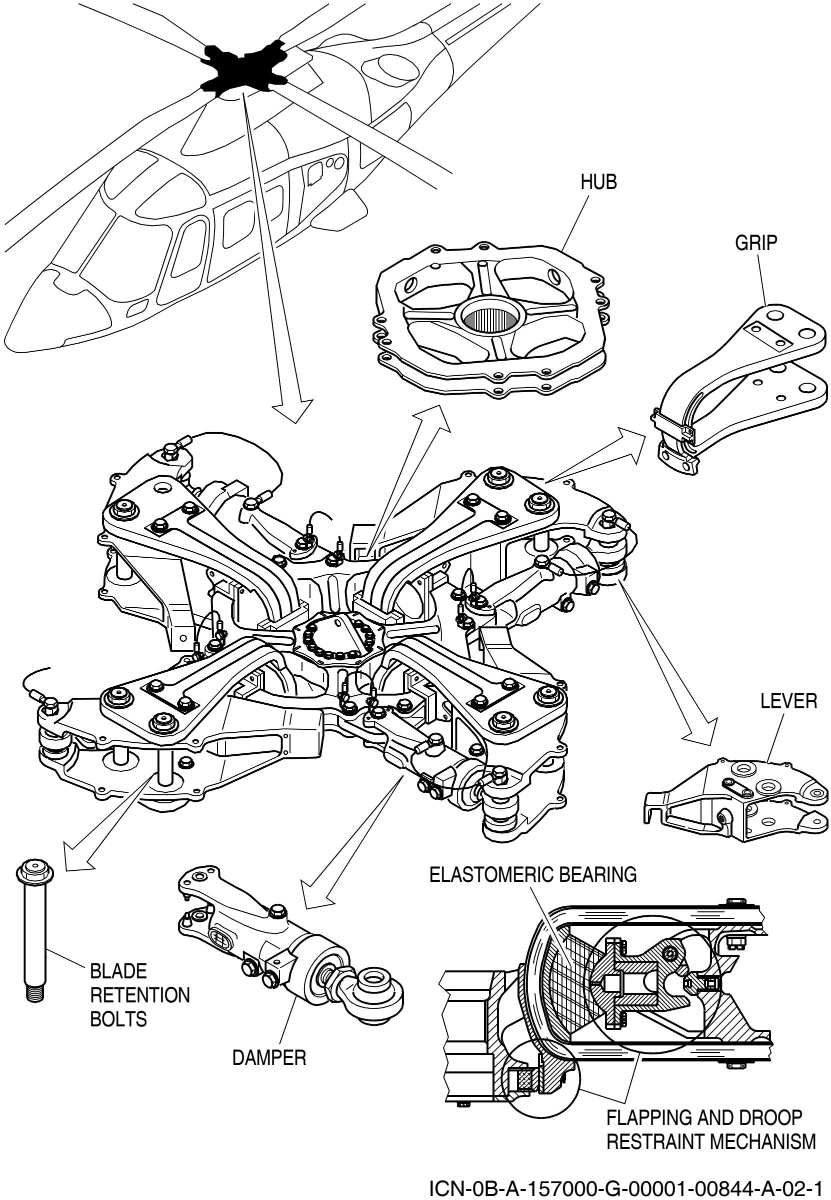

Figure 7-125 Main Rotor Head - Component Location

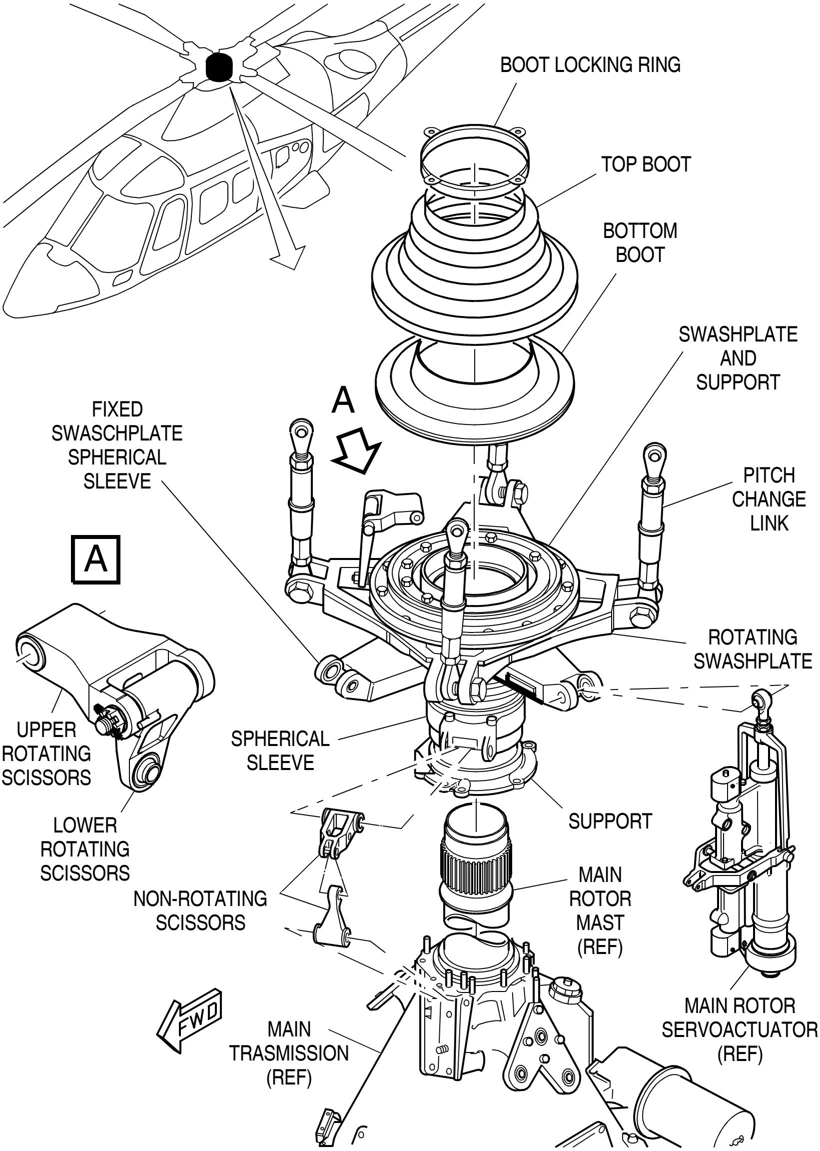

Figure 7-126 Rotating Controls - Component Location

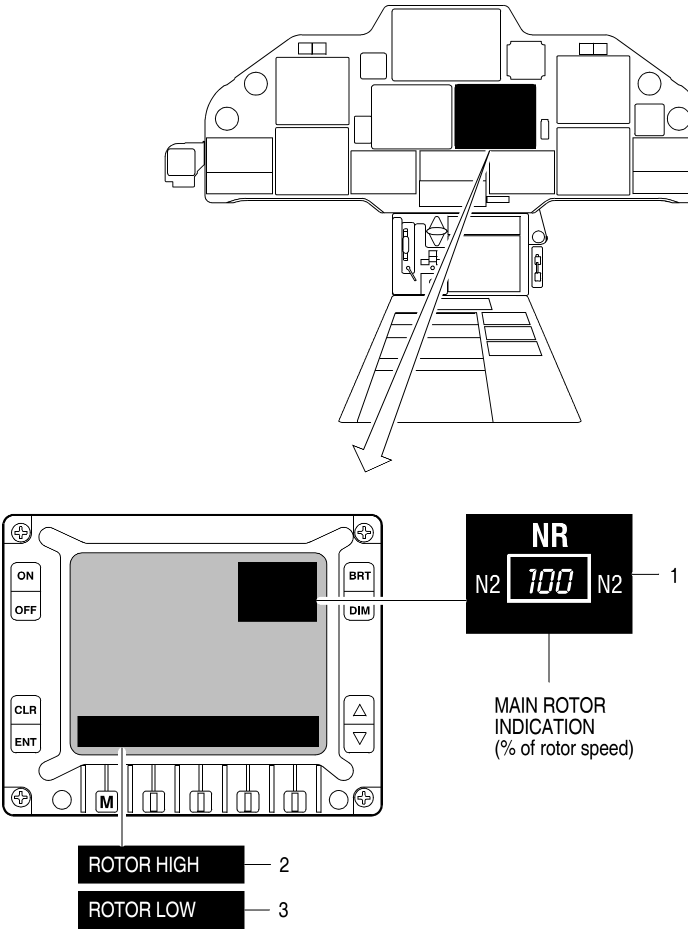

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Figure 7-127 Main Rotor Indicating System - EDU Indications

Key to Figure 7-127

Ref.	Control/Display	Function
1	NR	Indicates the rotational speed of the main rotor in percent of the maximum rating.
2	ROTOR LOW warning message	Displayed when the main rotor speed is less than 96% (power on) or 95% (power off - autorotation). The warning message is associated with a warning audio tone and with the aural message “rotor low”. The warning audio tone and the aural message reset when the NR is less than 80%.
3	ROTOR HIGH warning message	Displayed when the main rotor speed is more than 105% (power on) or 110% (power off - autorotation). The warning message is associated with a warning audio tone and with the aural message “rotor high”.

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MAIN ROTOR DRIVE

(Chapter 63)

GENERAL

The main rotor drive system includes:

- The main rotor shafts
- The main transmission (MGB)
- The main transmission lubricating system
- The oil cooling system
- The rotor brake system.
- The chip burner system (optional).

MAIN ROTOR DRIVE SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-128.

MAIN ROTOR SHAFTS

Refer to Figure 7-129.

The main rotor shafts transmit engine torque to the main transmission, they are installed between the engine power drives and the freewheel units of the main transmission.

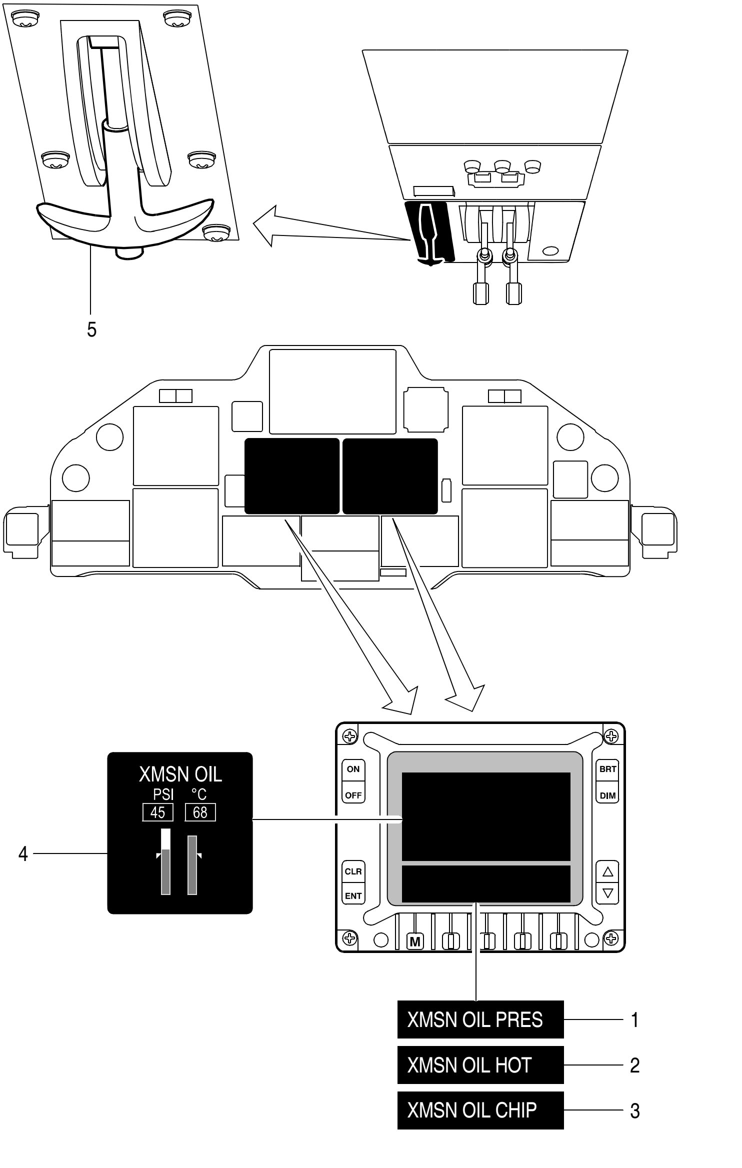

Figure 7-128 Main Rotor Drive - Controls and Displays

Key to Figure 7-128

Ref.	Control/Display	Function
1	XMSN OIL PRES warning message	Displayed when the pressure of the MGB oil is less then 30 psi. The warning message is associated with a warning tone and with the aural message.
2	XMSN OIL HOT warning message	Displayed when the temperature of the MGB oil exceeds 120 °C. The warning message is associated with a warning tone and with the aural message.
3	XMSN OIL CHIP caution message	Displayed when metal chips are detected in the MGB oil.
4	XMSN OIL indicator	Indicates the temperature and pressure of the MGB oil.
5	Rotor brake control lever	Fully forward - the rotor brake is released. Fully aft - the rotor brake is applied. The push-button on the brake lever must be pressed to allow the control lever to be moved.

MAIN TRANSMISSION

Refer to Figure 7-130.

The main transmission is mounted on the cabin upper deck forward of the two engines.

The main transmission reduces the speed of 6000 RPM from the main drive shafts to a speed of 380 RPM (100% NR) in the main rotor mast, with three stages of RPM reduction:

– The first stage of reduction includes two freewheels and associated input gear shafts which drive two symmetrical idler gears, and one gear installed on the external splines of a main input pinion.
– The second stage of reduction includes a pinion which drives a bevel gear shaft.
– The third stage of reduction includes a planetary gear train. The planetary gear train drives in turn the main rotor mast installed in the internal splines of the hub of the planetary gear-train.

The main rotor mast and associated roller and spherical bearings are installed in the upper case and mast assembly bolted to the upper side of the transmission main case.
The upper case is also the reservoir of the oil system.
The main case houses the input pinion and associated roller bearing and triplex spherical bearing, the bevel gear shaft and associated roller bearing, the sun gear and the planetary assembly. Mounted externally, on the left and right sides of the main case, there are a hydraulic pump drive and a hydraulic pump and tachometer-generator drive which are driven by bevel gear shaft. The freewheels, the input gear shafts, the idler gears, the tail rotor drive and the oil pump are installed in the coupling gearbox bolted to the lower aft side of the main case. The upper case, the main case and the coupling gearbox are provided with internal oil passages and oil jets for the lubrication of the main transmission components.

MAIN TRANSMISSION LUBRICATING SYSTEM

Refer to Figure 7-131.

The main transmission is lubricated by a system that includes a dual pump installed on the coupling gearbox.

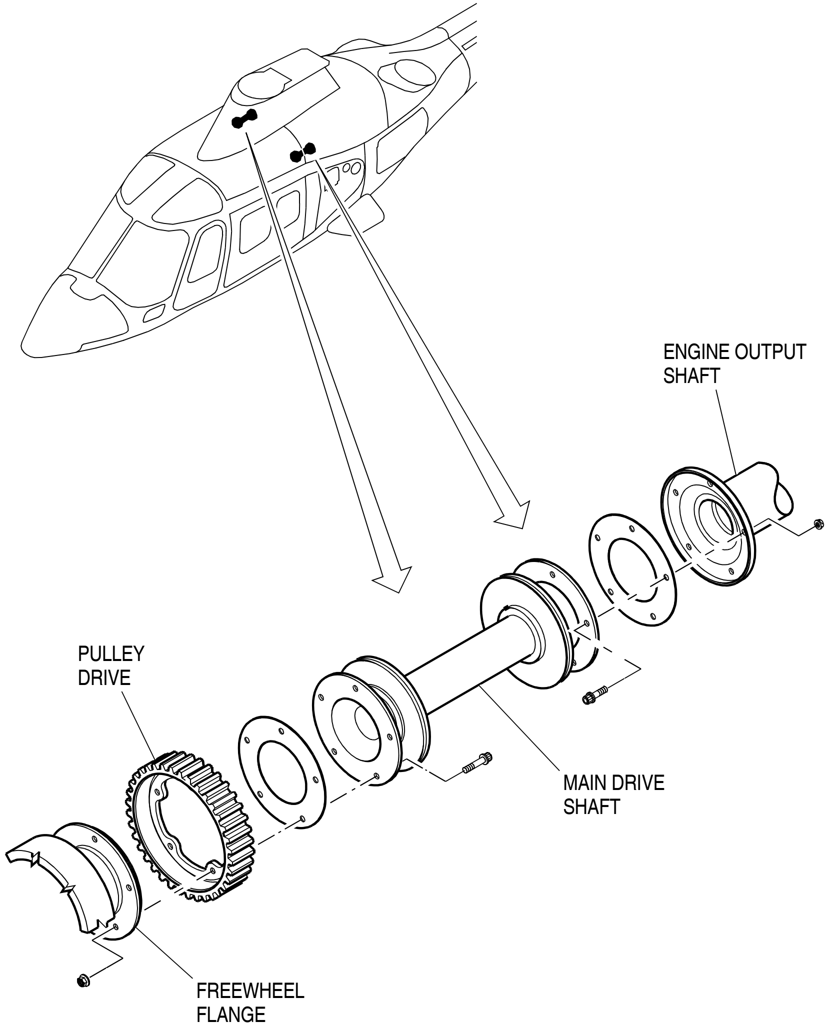

Figure 7-129 Main Drive Shaft

The lubricating oil contained in the upper case and mast assembly flows to one section of the dual pump (pressure pump), to the external filter, to all internal passages and oil jets of the transmission cases, to the bottom of main case and coupling gearbox and to the other section of the dual pump (scavenge pump).

From the scavenge pump the oil flows to the return lines and to the upper case through the thermostatic bypass valve and the oil coolers. A bypass valve on the external oil filter permits the passage of the oil if the filter is blocked. A pressure regulating valve installed on the pressure pump adjusts the oil operating pressure.

OIL COOLING SYSTEM

Refer to Figure 7-132.

The main-transmission oil-cooling system includes two fans installed on the accessory gearbox, two oil coolers mounted on either side of the transmission and a thermostatic bypass valve installed on a housing.

An oil return line is connected between the outlet port of the scavenge pump and the thermostatic bypass valve. When the oil temperature is below 54 °C, the thermostatic valve is completely open and oil flows directly to the upper case. When the temperature rises above 54 °C, the valve begins to close and the oil is directed to the oil coolers. The flow of oil to the oil coolers is proportional to the oil temperature and the differential pressure between the inlet and the outlet port. When the temperature reaches 74 °C, the valve is completely closed and all the oil is directed to the oil coolers. When the differential pressure is 20 psi, the valve begins to close. When the differential pressure reaches 80 psi, the valve is completely closed.

An air intake duct is connected to the forward part of each oil cooler. External ram air enters through the two NACA air intakes located on the transmission forward cowling. The air enters the intake duct and is drawn through the oil cooler by a fan fitted to the aft end of each oil cooler. The exhaust air is expelled to the atmosphere through an exhaust duct connected to the fan.

Each fan is driven by a belt connected to the pulley drive on the main drive shafts.

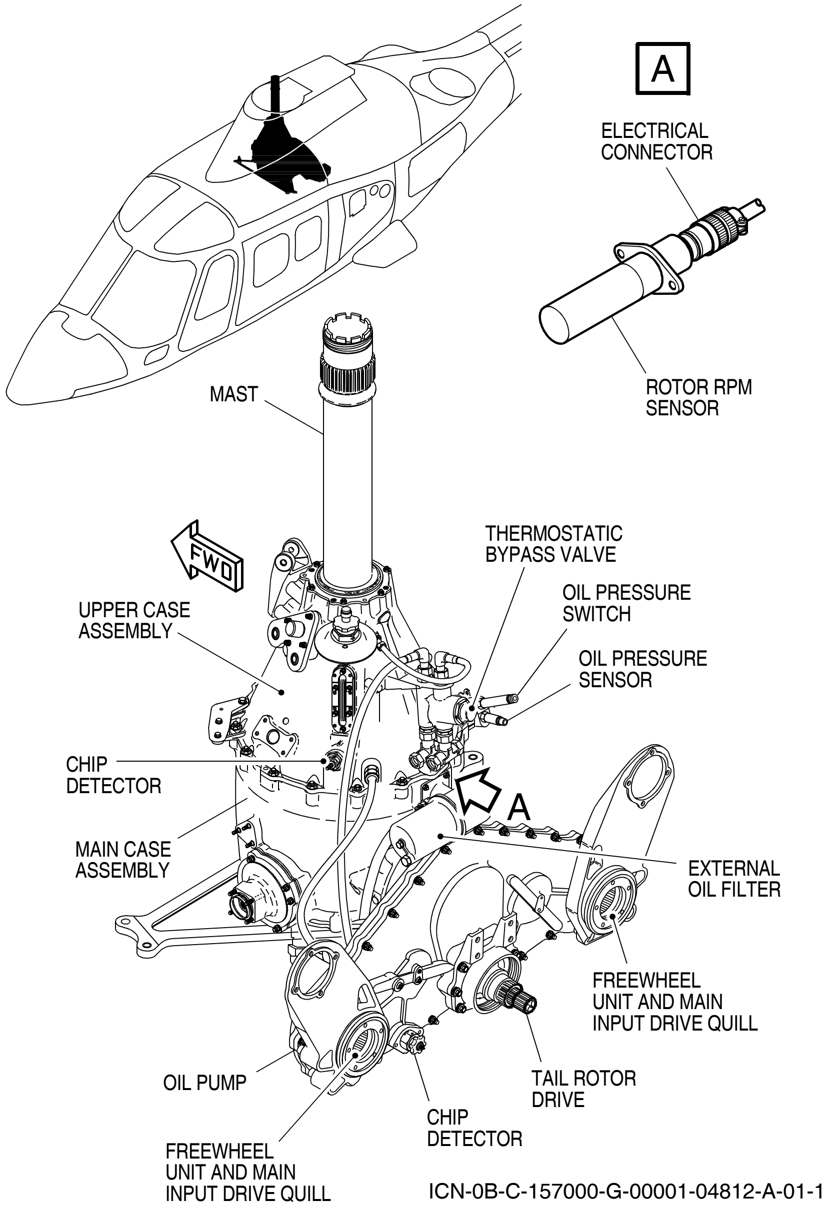

Figure 7-130 Main Transmission - Component Location

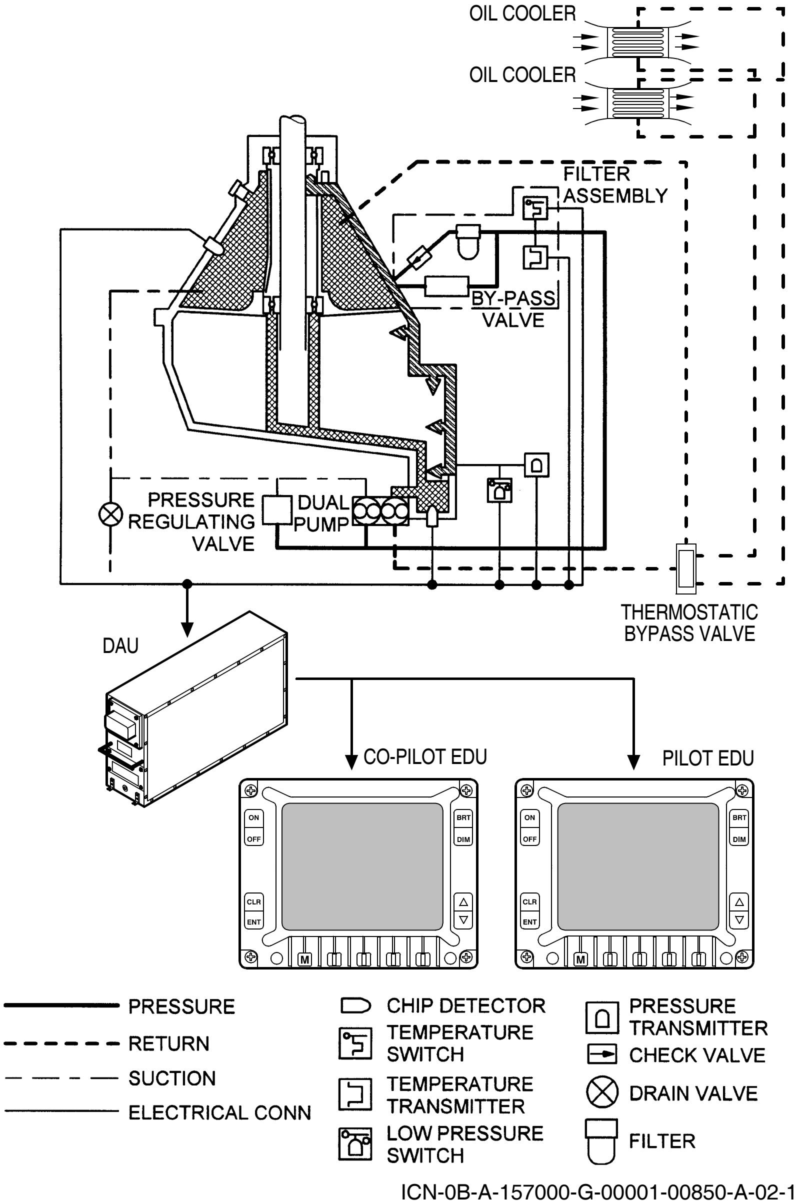

Figure 7-131 Main Transmission Lubricating System - Schematic Diagram

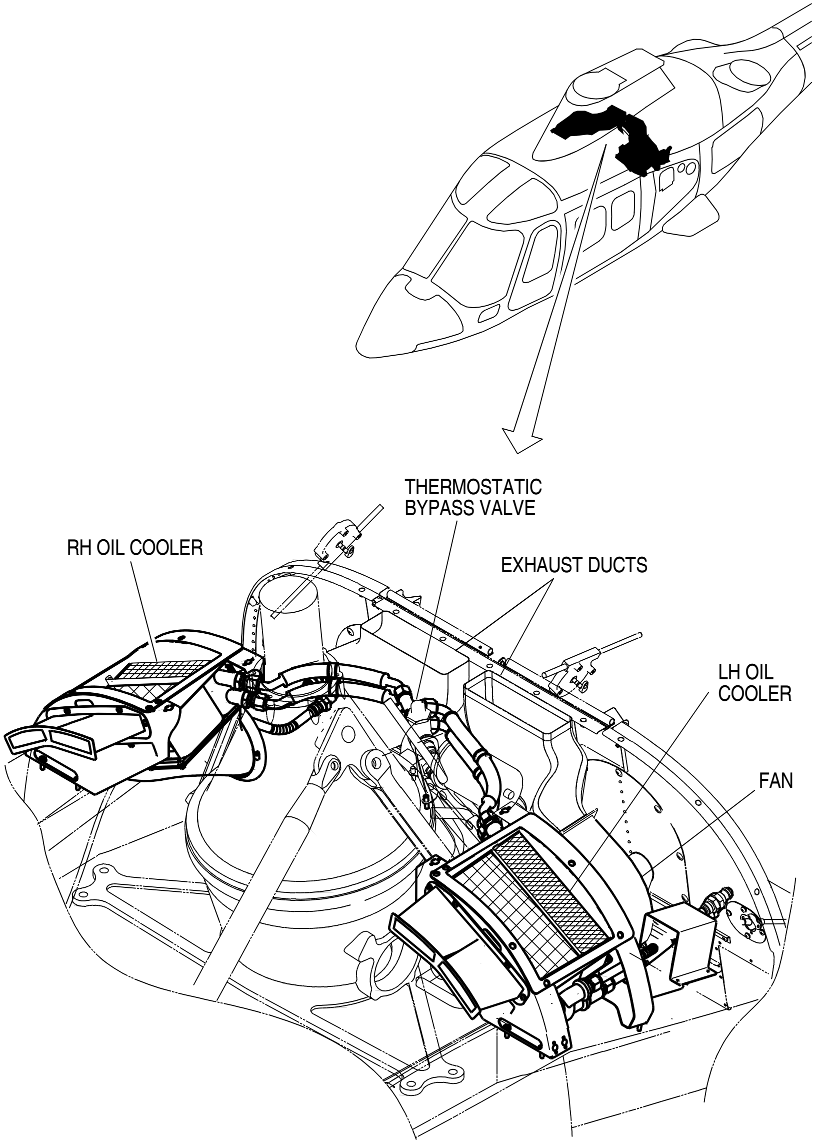

Figure 7-132 Oil Cooling System - Component Location

ROTOR BRAKE SYSTEM

The rotor brake system is used to stop the rotation of the rotor during engine power-off. The rotor brake system includes a hydraulically operated caliper acting on a disc secured to the tail rotor drive pinion. Hydraulic fluid supplied from the utility hydraulic system operates the caliper. The brake is applied by actuating the control lever located on the overhead console. The control lever is connected to the nose wheel centering lock and the brake selector valve by means of a metal cable.
The main components of the rotor brake system are shown in Figure 7-133.
Refer to Figure 7-134 for the schematic diagram of the rotor brake system.

CHIP BURNER SYSTEM

The chip burner system, in addition to the chip detector caution function, enables the pilot to burn metal particles which are deposited on the chip detectors of the main transmission and 90-degree gearbox.

The system consists of three chip detectors, a power module and the relevant wire harness.

Two chip detectors are located in main transmission and one in 90-degree gearbox. The power module, located in the baggage compartment, contains all circuitry required for operation of the system.

The chip burner system push-button is installed on the overhead console. Refer to Figure 7-134.

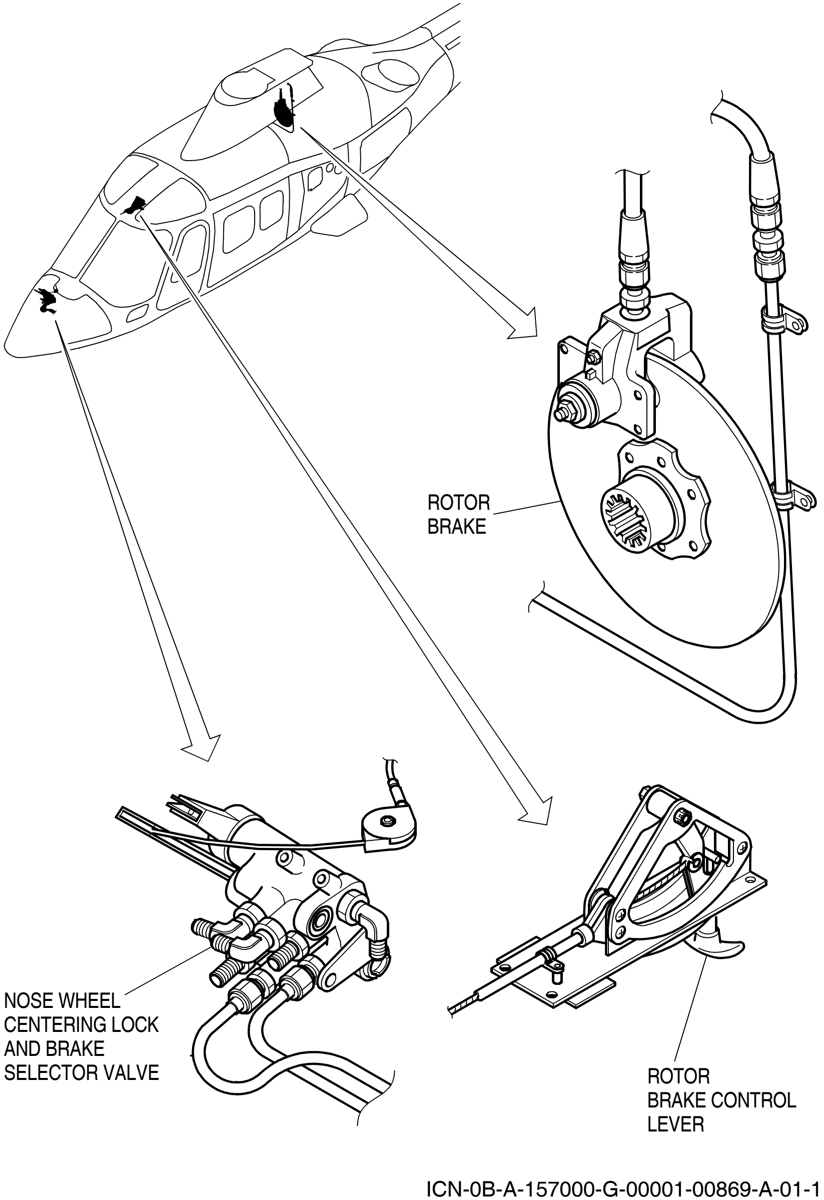

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Figure 7-133 Rotor Brake System - Component Location

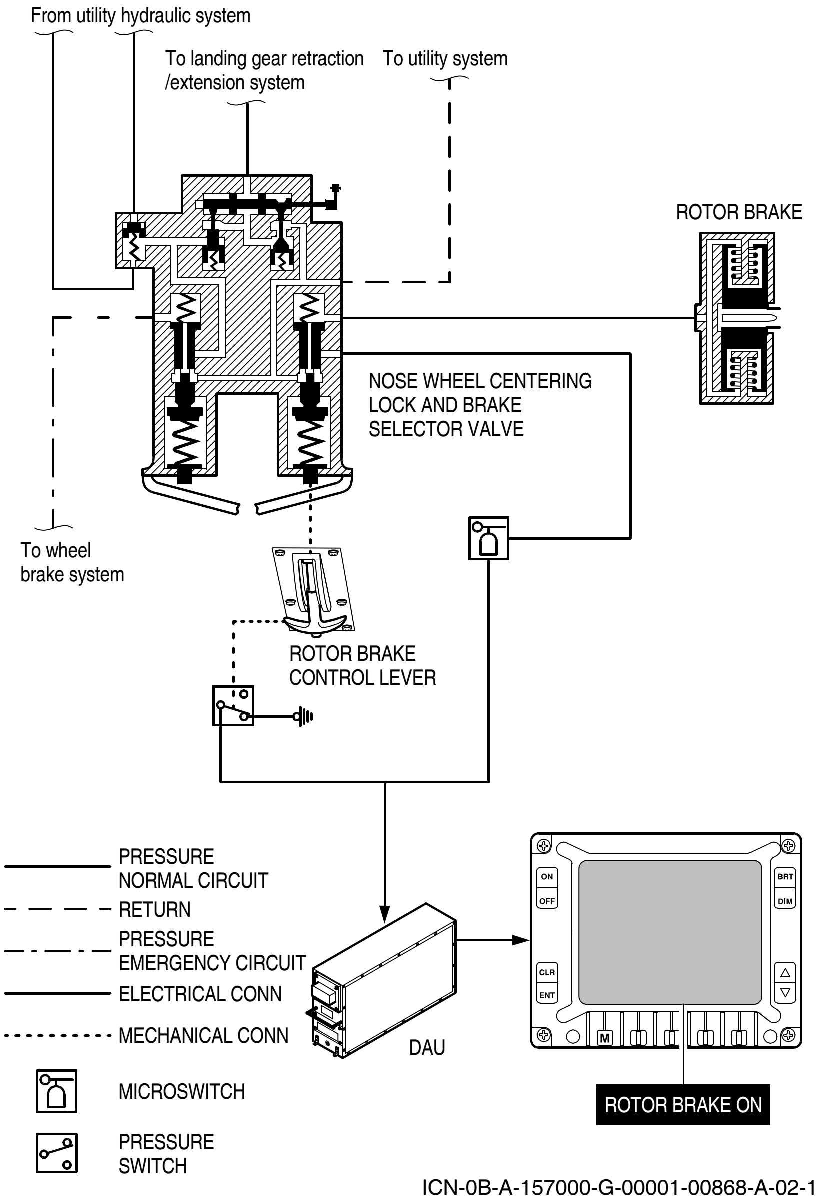

Figure 7-134 Rotor Brake System - Schematic Diagram

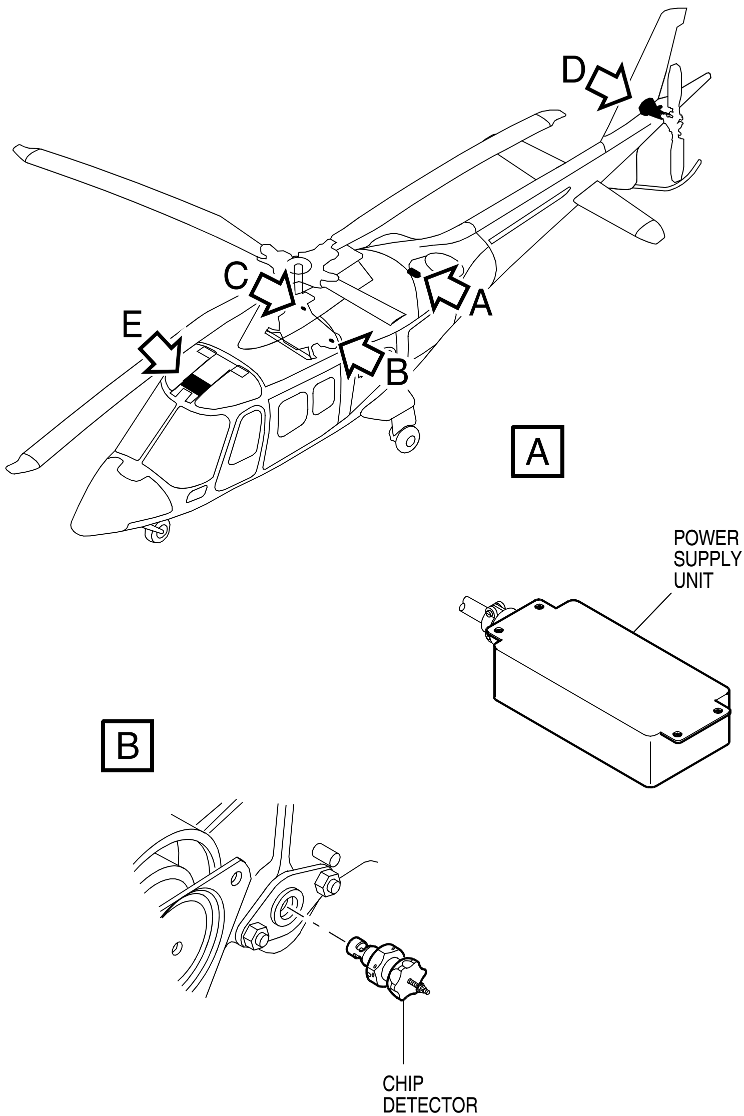

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Figure 7-134A Chip Burner System - Component Location (Sheet 1 of 2)

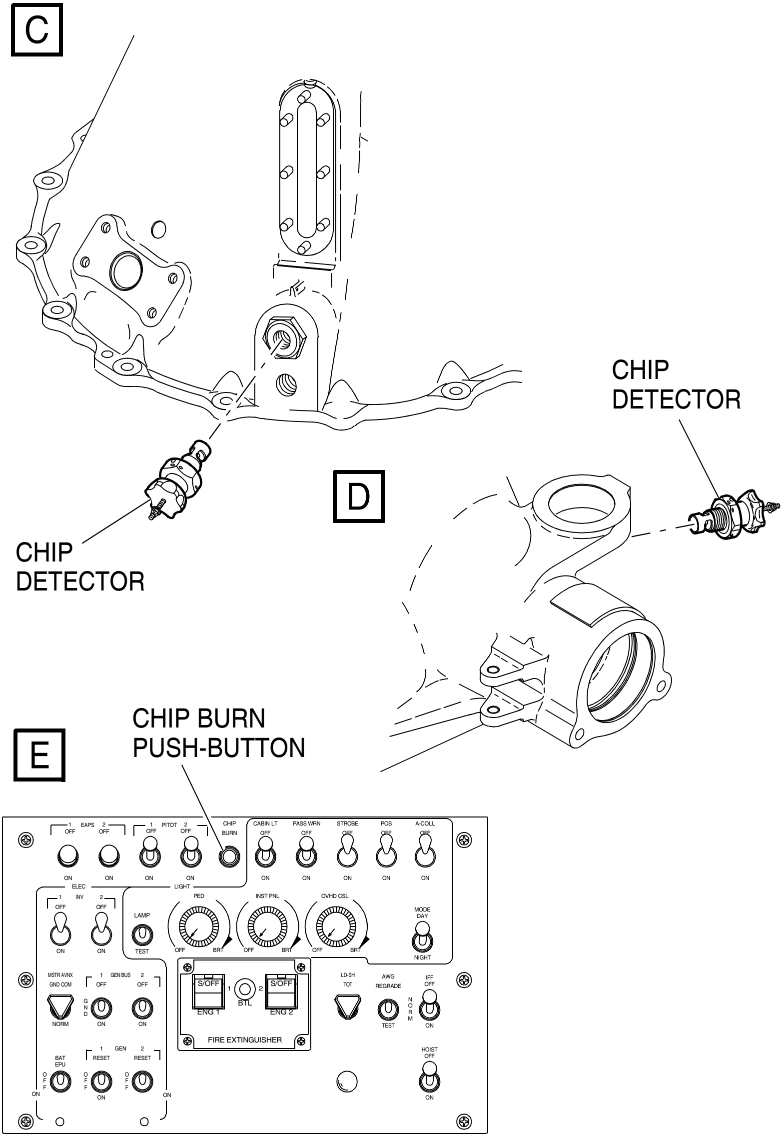

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Figure 7-134A Chip Burner System - Component Location (Sheet 2 of 2)

TAIL ROTOR
(Chapter 64)

GENERAL

The tail rotor system includes:

– The tail rotor hub
– The tail rotor blades
– The pitch change mechanism.

Refer to Figure 7-135.

TAIL ROTOR HUB

Refer to Figure 7-136.

The tail rotor hub installation includes:

– The trunnion
– The retention strap assembly (2 off)
– The hub bushing (2 off)
– The cover flange (2 off)
– The cover (2 off).

TAIL ROTOR HUB

The tail rotor hub provides the support for the trunnion and for the strap assembly. It is made of stainless steel alloy.
The tail rotor hub has four cylindrical arms. Two arms are mounted at the root of the blades and permit the pitch change movement.
The other two arms of the hub, the axis of which is at an angle of 45° with respect to the axis of the former arms, house the splined trunnion and permits flapping of the blades.
The hub and blade assembly comprises an internally splined trunnion installed on the 90-degree gearbox output shaft, a hub mounted on the trunnion, two blades linked to the hub with two tension-torsion straps and two retaining bolts.

TRUNNION

The trunnion is installed in the hub bushings of the tail rotor hub. At the ends of the trunnion the cover flange close the access to the hub bushing and the trunnion.
The trunnion provides the attachment for the hub of the output shaft and the pivot for the flapping hinge.

RETENTION STRAP ASSEMBLY

Two retention strap assemblies keep the blades on the tail rotor hub and absorb the axial centrifugal loads. The retention straps are made of stainless steel laminates.
The retention strap pin and the pin hold each retention strap in the tail rotor hub.

HUB BUSHING

Two hub bushings, one for each side of the hub (small cylindrical arm), are installed on the hub.
The hub bushings hold the trunnion in its position on the hub.

COVER FLANGE

The cover flange (one for each side of the hub) is installed on the hub bushing and the trunnion.
Three elastic pins hold the cover flange and the shim in position on the trunnion.

COVER

The cover (one for each side of the hub) is installed on the longer arm of the tail rotor hub.
The adhesive attaches the cover to the hub arm.

TAIL ROTOR BLADES

Refer to Figure 7-137.

The tail rotor blades are made of composite material.
The leading edge of the blades is protected by an electro-deposited nickel abrasion strip that has variable thickness.
The root of each blade has the holes for the pin that attaches it to the pitch link.
Each blade is secured to the hub by one bolt provided with washers of different weight for the static balancing of the tail rotor.
Each blade is statically balanced during manufacture by means of weights fitted on the blade root and tip.

PITCH CHANGE MECHANISM

The pitch change mechanism, mounted on the 90° output shaft, is connected to the tail rotor blades by a slider and two pitch links.

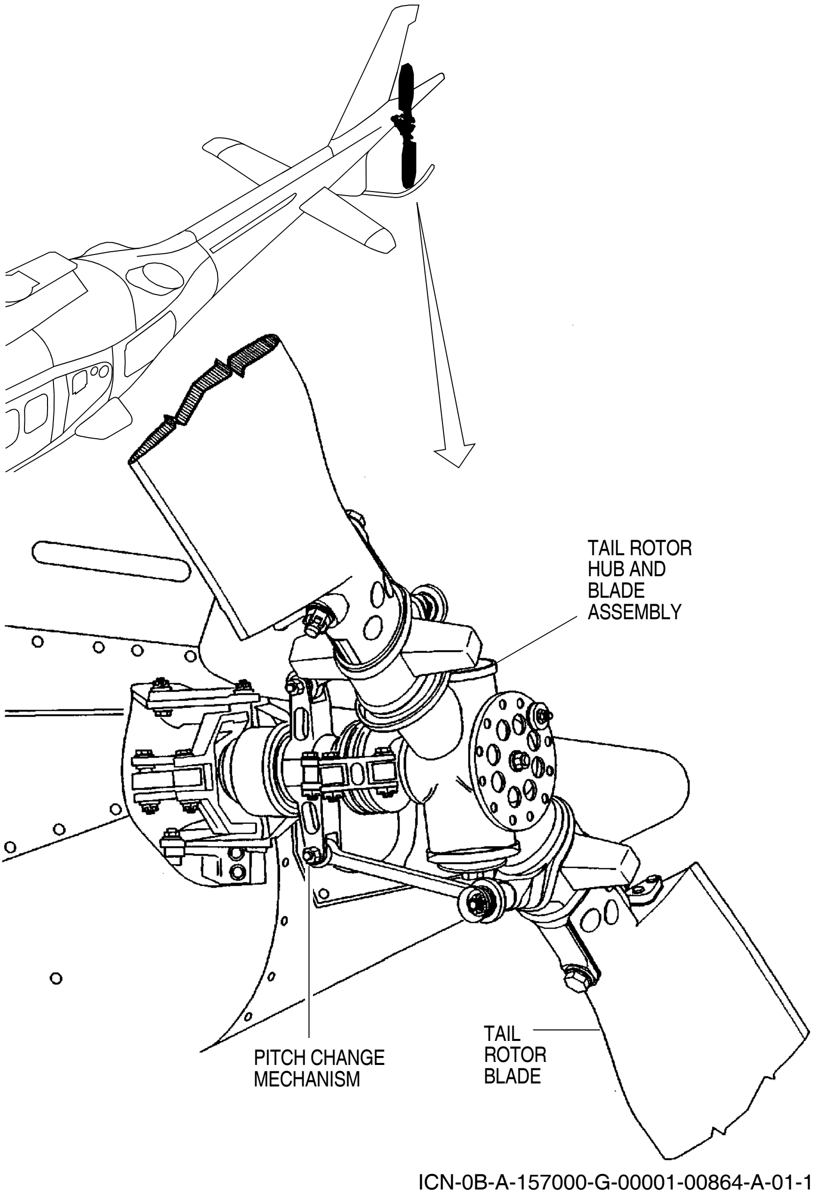

Figure 7-135 Tail Rotor - Component Location

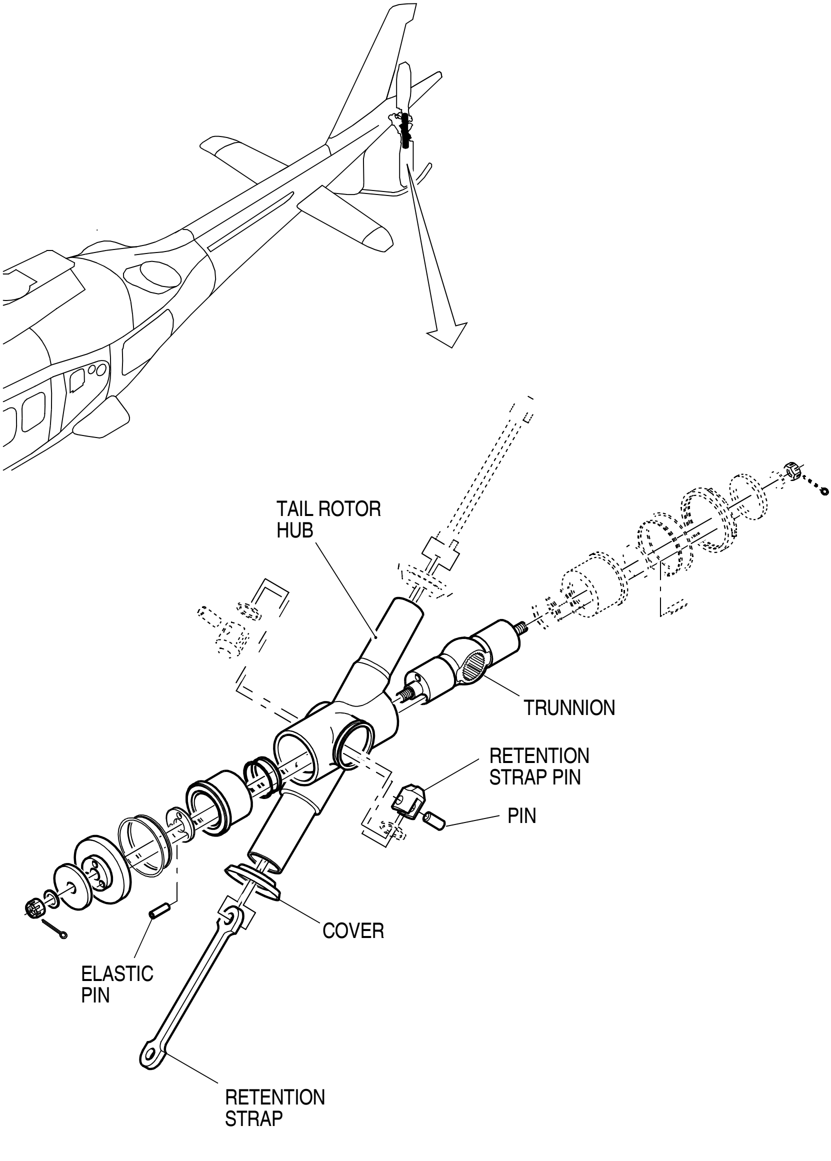

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Figure 7-136 Tail Rotor hub - Component Location

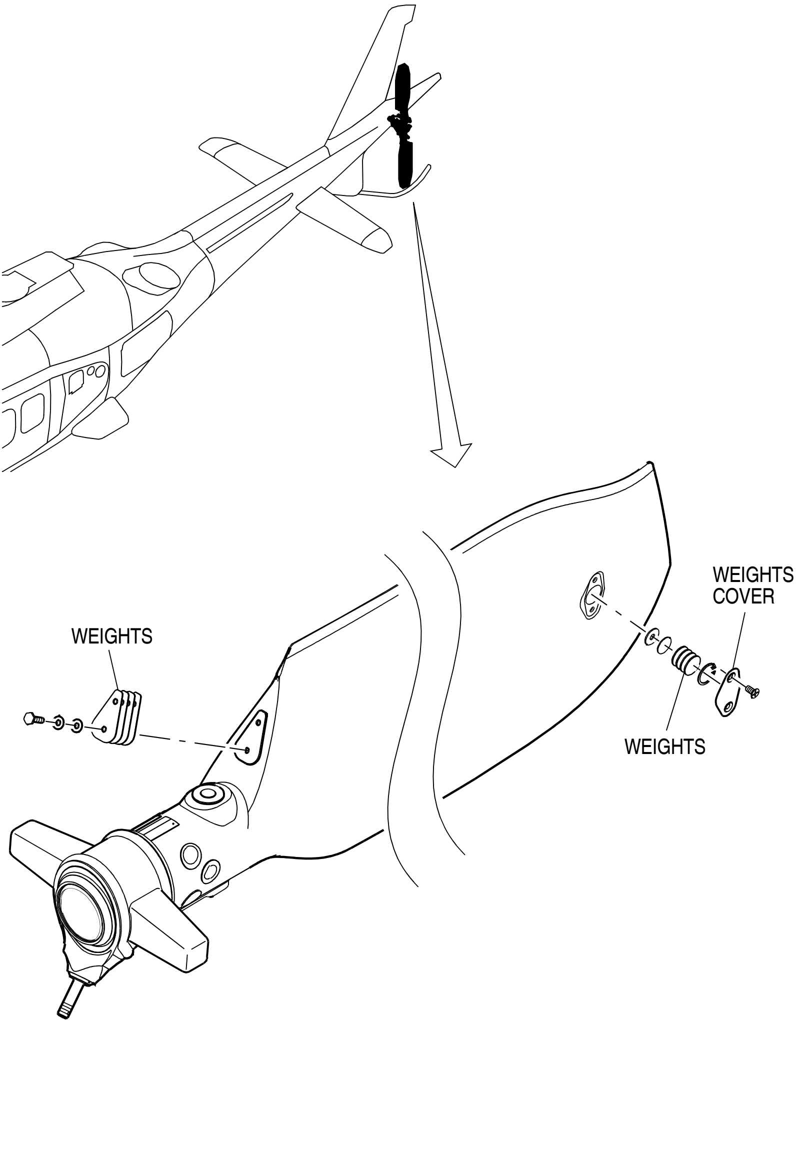

Figure 7-137 Tail Rotor Blade

TAIL ROTOR DRIVE

(Chapter 65)

GENERAL

The tail rotor drive system transmits power from a drive on the main transmission to the tail rotor through three drive shafts and the 90-degree gearbox. The tail rotor drive system includes:

– The tail rotor drive shafts
– The 90-degree gearbox.

TAIL ROTOR DRIVE SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-138.

TAIL ROTOR DRIVE SHAFTS

Refer to Figure 7-139.

The tail rotor has three drive shafts (Number 1 drive shaft, Number 2 drive shaft and Number 3 drive shaft).
The Number 1 drive shaft transmits the torque from the main gearbox to the Number 2 drive shaft. The Number 2 and Number 3 drive shafts transmits the torque to the 90-degree gearbox.

90-DEGREE GEARBOX

The 90-degree gearbox provides a 90° change in the direction of drive and 2.8 to 1 speed reduction between the input shaft and the output shaft on which the tail rotor is mounted.
The 90-degree gearbox consists mainly of the case assembly, input pinion assembly and output shaft assembly.
The 90-degree gearbox is attached to the structure through a mounting sleeve.

The case assembly is provided with a breaker-type filter-cap, an oil level sight, a magnetic chip detector, an oil drain plug, and the mounting lugs for the tail rotor pitch change mechanism.
The input pinion assembly, provided with a spiral bevel gear, is supported by a duplex ball bearing and a single-row roller-bearing.
The output shaft assembly consists mainly of a flanged shaft and a bevel gear bolted to the flange. The shaft is supported by a duplex ball bearing and a single row roller bearing.

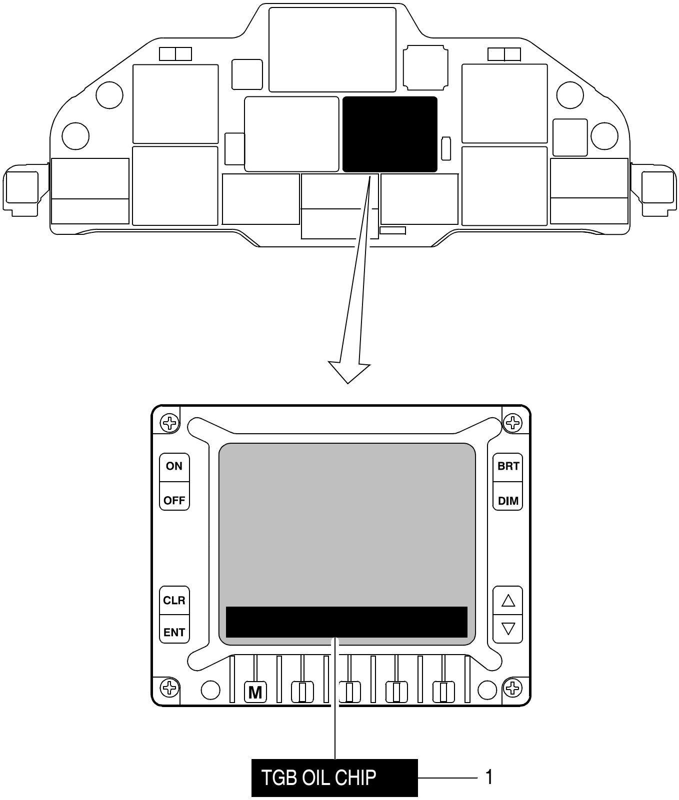

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Figure 7-138 Tail Rotor Drive - Controls and Displays

Key to Figure 7-138

Ref.	Control/Display	Function
1	TGB OIL CHIP caution message	Shown when the chip detector detects metal chips in the lubricating oil of the 90-degree gearbox.

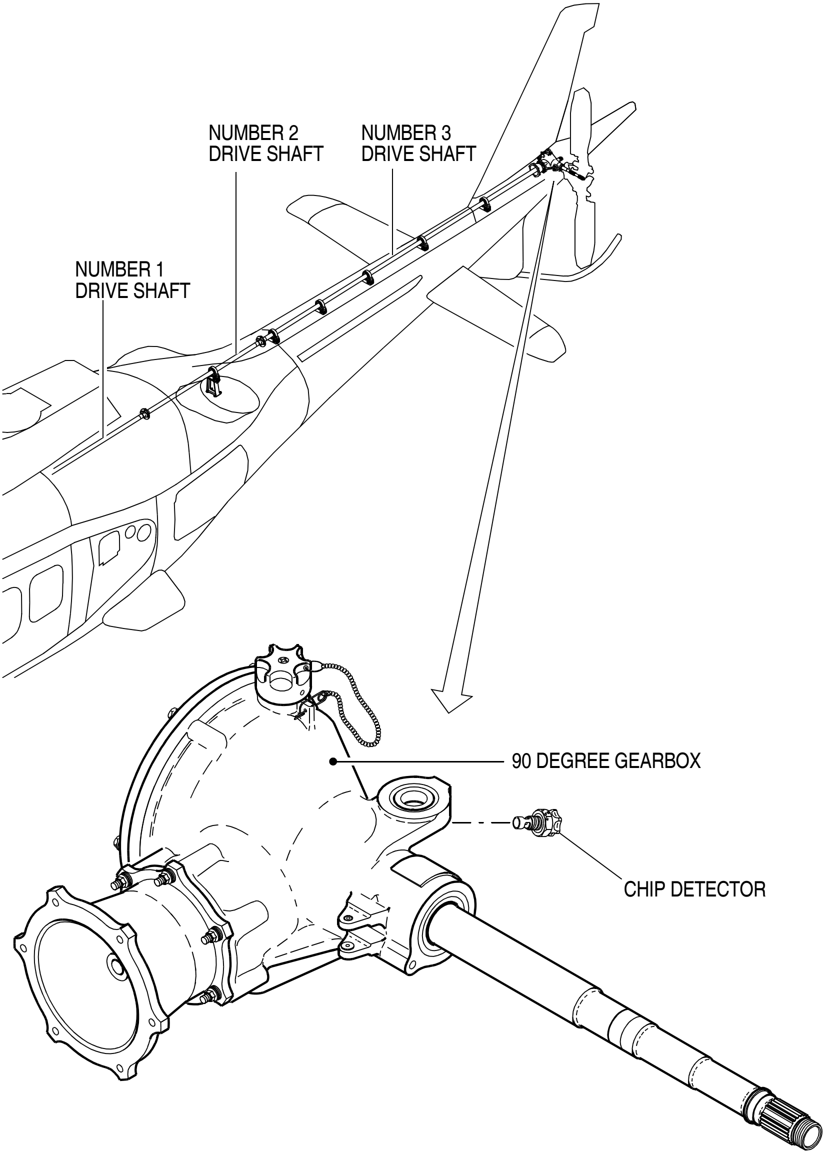

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Figure 7-139 Tail Rotor Drive - Component Location

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ROTOR FLIGHT CONTROLS

(Chapter 67)

GENERAL

The rotor flight control system gives positive control of attitude, speed and altitude of the helicopter.
The system includes:

- The main rotor control system
- The tail rotor control system
- The servo control system.

ROTOR FLIGHT CONTROLS SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-140.

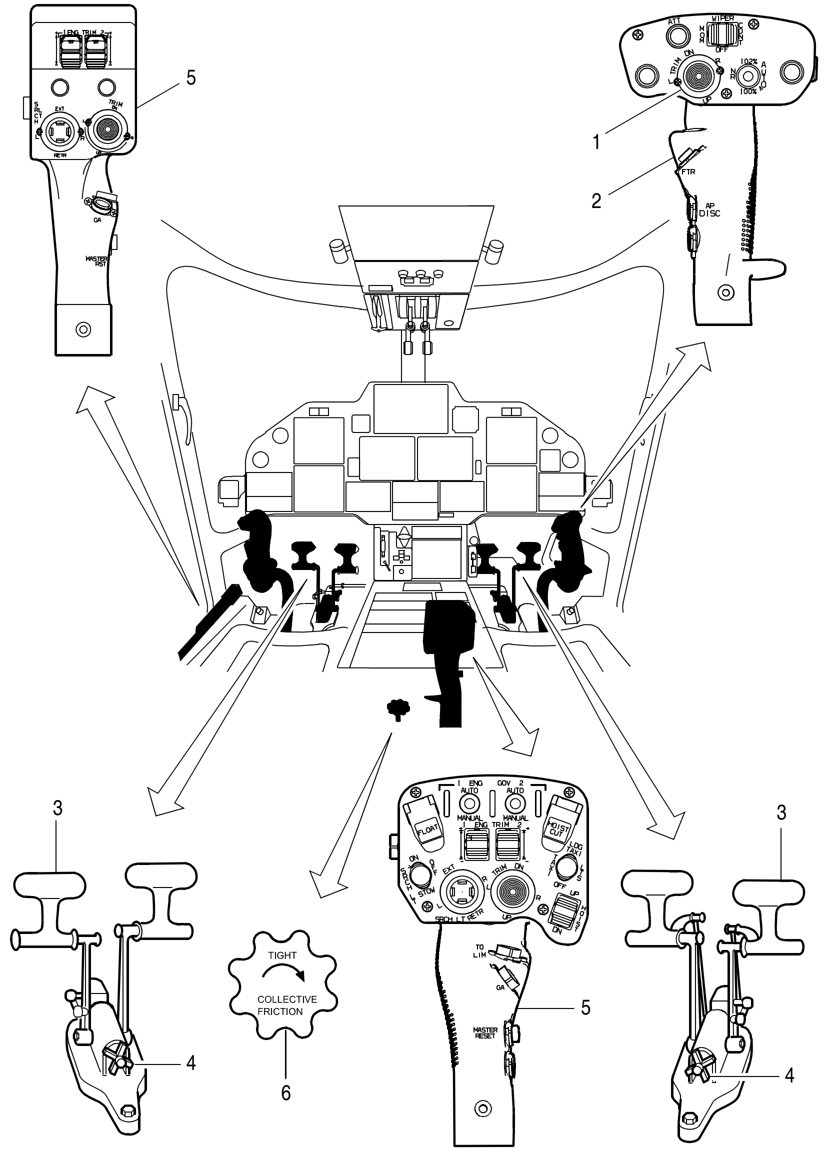

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Figure 7-140 Rotor Flight Control System - Controls and Displays

Key to Figure 7-140

Ref.	Control/Display	Function
1	BEEP TRIM switch	Moved right or left - operates the actuator to obtain lateral trimming of the helicopter. Moved forward or aft - operates the actuator to obtain longitudinal trimming of the helicopter.
2	Pilot’s and Co-pilot’s cyclic control stick	Controls the cyclic pitch of the main rotor for longitudinal and lateral control of the helicopter.
3	Tail rotor pedals	Controls the pitch of the tail rotor for directional control of the helicopter.
4	Tail rotor pedals adjustment knob	Permits the adjustment of the tail rotor pedals.
5	Collective control lever	Controls the collective pitch of the main rotor.
6	Friction knob	Permits the adjustment of the force necessary to operate the collective control lever.

MAIN ROTOR CONTROL SYSTEM

The main rotor control system controls the helicopter in pitch and roll, climb and descent.
The main rotor control system includes:

– The collective control system
– The cyclic control system
– The magnetic-brake artificial-feel and trim units
– The stabilization actuators
– The mixing control system.

COLLECTIVE CONTROL SYSTEM

The collective control system is a conventional rigid control tube type, and is controlled by the pilot and copilot through the collective control levers installed on the left side of the pilot’s and copilot’s seats (Figure 7-141).
The system transmits the control movement to the mixing group. The friction knob permits the adjustment of the force necessary to operate the controls levers. The control levers can be locked in any position by fully tightening the knob.
The pilot’s and copilot’s controls are the same, except for the handgrip which are different.
The main components of the system are the collective control levers, the torque tube and the mixing group.

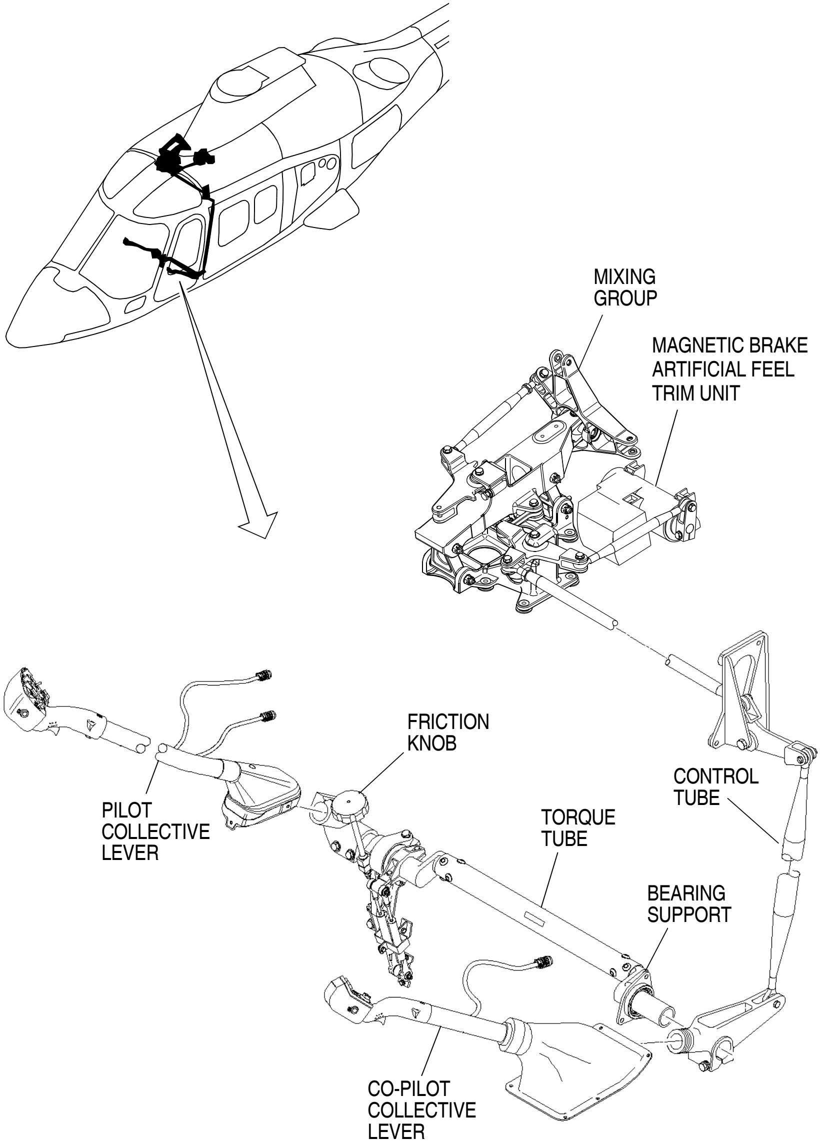

Figure 7-141 Collective Control System - Component Location

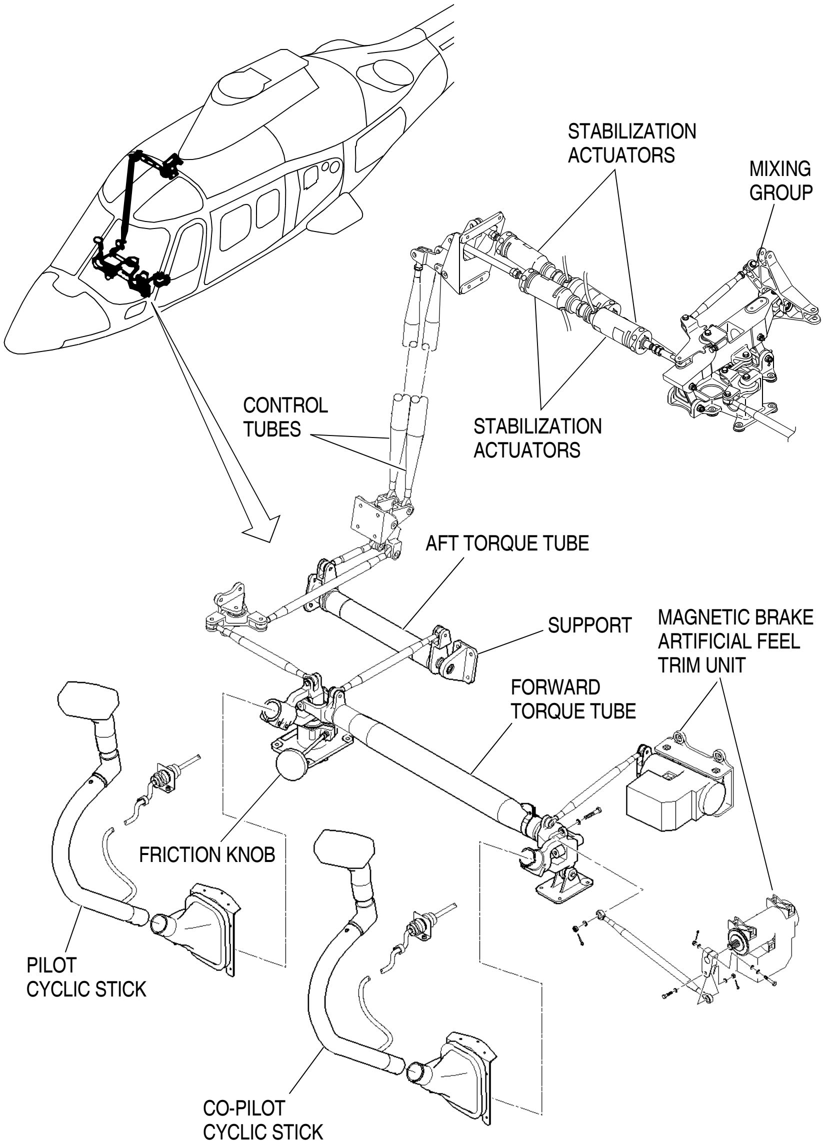

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Figure 7-142 Cyclic Control System - Component Location

Collective control lever

The collective control lever, located on the left side of the pilot's seat consists of a shaped light alloy tube. The lower end of the lever is protected by a boot and is attached to the bearing of the support to allow control movements. A handgrip is fitted to the top of each lever. The two handgrips are different. The pilot's handgrip is larger and includes several controls, the copilot's handgrip carries only the engines trim switches and the master caution warning message reset push-button (Figure 7-141).
A torque tube interconnects the two collective control levers.

CYCLIC CONTROL SYSTEM

The cyclic control system is a conventional rigid control tube type and is controlled by the pilot's and copilot's through the cyclic control sticks. The system transmits the control movements to the mixing group. The friction knob permits the adjustment of the force necessary to operate the control sticks. The control sticks can be locked in any position by fully tightening the knob. The system incorporates a magnetic brake artificial feel and trim unit and two control tubes with stabilization actuators.
The main components of the cyclic control system are the cyclic control sticks, cyclic control stick handgrips, cyclic support group and torque tubes.
The pilot and copilot's controls are the same.
Refer to Figure 7-142 for the location of the components of the cyclic control system.

Cyclic control stick

Each cyclic control stick, located in front of the pilot's and co-pilot's seat, consists of a light alloy shaped tube. The lower end of the stick is protected by a boot and is attached to the self-aligning bearing of the support to allow control movements in all directions. An anatomic handgrip made of synthetic material incorporates flight control and utility switches.

MAGNETIC BRAKE ARTIFICIAL FEEL AND TRIM UNIT

The cyclic and collective control system are equipped with a magnetic brake artificial feel and trim units and control tubes with stabilization actuators. Refer to Chapter 22 Auto Flight for more information.

STABILIZATION ACTUATORS

The stabilization actuators stabilize the helicopter in response to electrical control signals supplied by the stabilization system.

MIXING CONTROL SYSTEM

The mixing control system is a conventional rigid type and comprises a mixing group, control tubes.
The mixing group is installed on the upper deck of the helicopter. It receives the movements from the cyclic and collective control systems and transmits them to actuate the servo-actuators.
The main components of the mixing group are the carriage, the support and the mixing lever. The support is attached to the helicopter structure. The carriage is connected to the support through two links.
The movements transmitted to the carriage from the collective control system through the control tube cause lenghtwise movement of the carriage and the change of position of the mixing lever.

The mixing lever has a central body and four arms. A universal joint and two control tubes connected to two arms of the mixing lever connect the mixing lever to the carriage.

Three control tubes are connected to the arms of the mixing lever to transmit the movement to the actuator levers.
Refer to Figure 7-143 and Figure 7-145 for the location of the components of the mixing control system and the main rotor servo-actuator.

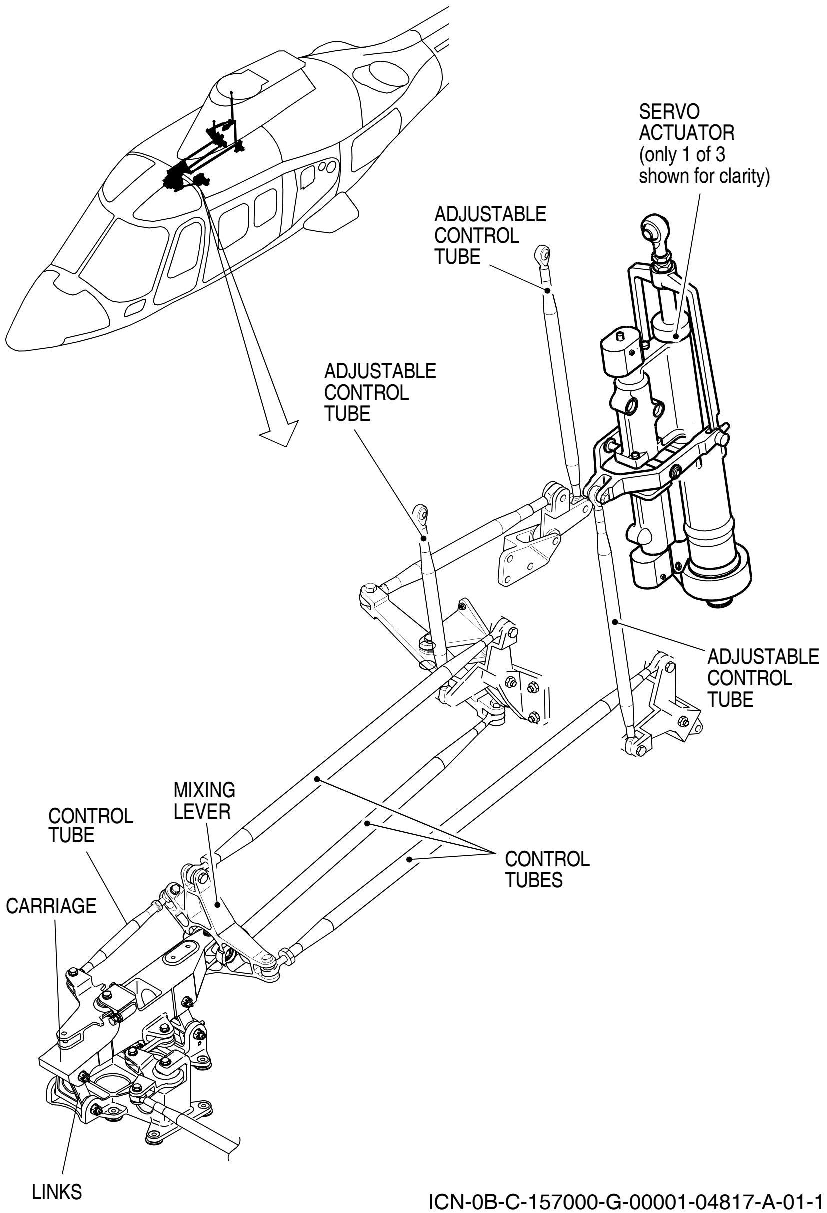

Figure 7-143 Mixing Control System - Component Location

TAIL ROTOR CONTROL SYSTEM

The tail rotor control system controls the direction (yaw axis) of the helicopter.
The tail rotor control system is controlled by two sets of adjustable tail rotor pedals that are connected by a series of levers, push-pull tubes and bellcranks to the tail rotor hydraulic servo-actuator. The system incorporates a control tube provided with a stabilization actuator and magnetic brake artificial feel unit (Figure 7-144).

TAIL ROTOR PEDALS

The tail rotor pedals are secured to the cabin floor facing the pilot and co-pilot's seat. The pedals are attached to a support and are connected to link rods that actuate the lever. The pedals are fitted with a device to adjust the distance between the pedals and the crew seats.
A friction adjustment unit is preset during rigging to get minimum friction on the system.
The tail rotor pedals are connected, through control tubes to a torque tubes.
The torque tube is a light alloy tube installed below the left door post of the crew cabin. The torque tube is mounted on supports secured to the helicopter structure, and incorporates two end levers for control tube connection.
The pilot's and copilot's tail rotor pedals are different. The pilot's pedals are fitted with the brake pedals to brake the main landing gear wheels, while the copilot's pedals are not provided with them.

MAGNETIC BRAKE ARTIFICIAL FEEL UNIT

The magnetic brake artificial feel unit installed in the tail rotor control system is similar to the cyclic magnetic brake artificial feel unit.
The magnetic brake is controlled by the F-trim switch located on the pedestal. The FTR switch on the cyclic control handgrip allows a temporary release of the magnetic brake.

STABILIZATION ACTUATOR

The tail rotor control system incorporates a control tube with a stabilization actuator. The actuator stabilizes the helicopter in response to electrical control signals supplied by the stabilization system.
Refer to Chapter 22 Auto Flight for more information.

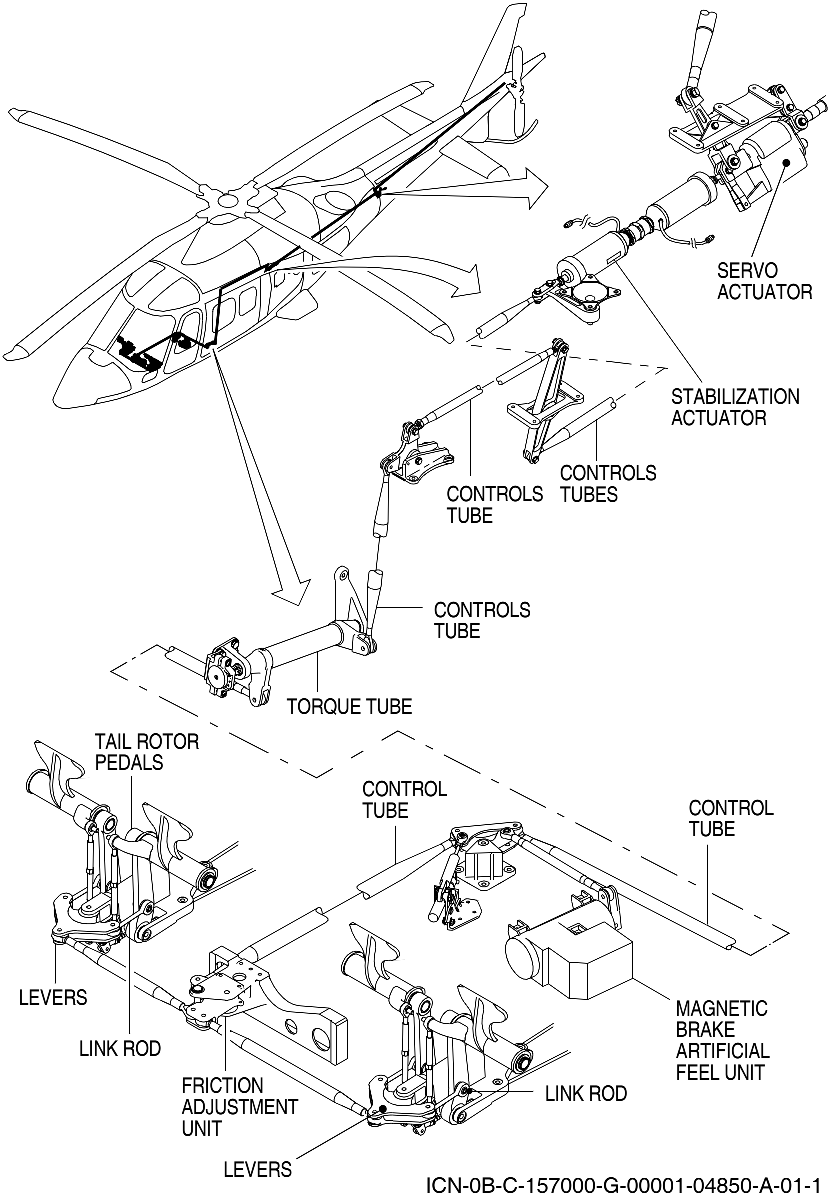

Figure 7-144 Tail Rotor Control System - Component Location

SERVO CONTROL SYSTEM

The servo control system comprises the main rotor servo-actuators and the tail rotor servo-actuator. The servo-actuators are powered from the flight controls hydraulic system.

MAIN ROTOR SERVO-ACTUATORS

The main rotor servo-actuator system includes three hydraulic servo-actuators. The servo-actuators are connected to the mixing control system and to the inner ring of the swashplate. The actuators are attached with supports to the top case.
The main rotor servo-actuators are of the tandem type, energized by two hydraulic systems. Hydraulic pressure operates the servo-actuator piston causing retraction or extension depending on control input to the servo-actuator distribution group valves via the mixing control system. In the event of a seizure of a servo-actuator distribution group valve, a backup valve, within the distribution group, will cut in to keep the effort required to operate the control within acceptable limits.
If a servo valve in one of the three main rotor servo-actuators is jammed, the caution message MAIN RTR SERVO (yellow) comes in view on the EDU1.
Refer to Figure 7-145 for the main rotor actuators.

TAIL ROTOR SERVO-ACTUATOR

The tail-rotor servo-actuator installation comprises a hydraulic, single body servo-actuator. The servo-actuator is supplied the No 1 flight control hydraulic system. The servo-actuator is located aft of the rear of the baggage compartment and is connected to the link, lever and adjustable control tube.
Hydraulic pressure operates the tail rotor servo-actuator piston causing retraction or extension depending on control input to the servo-actuator valve group via the adjustable control tube. In the event of seizure of the distribution valve, a backup valve, located within the valve group, will cut in to keep the effort required to operate the control within acceptable limits. .

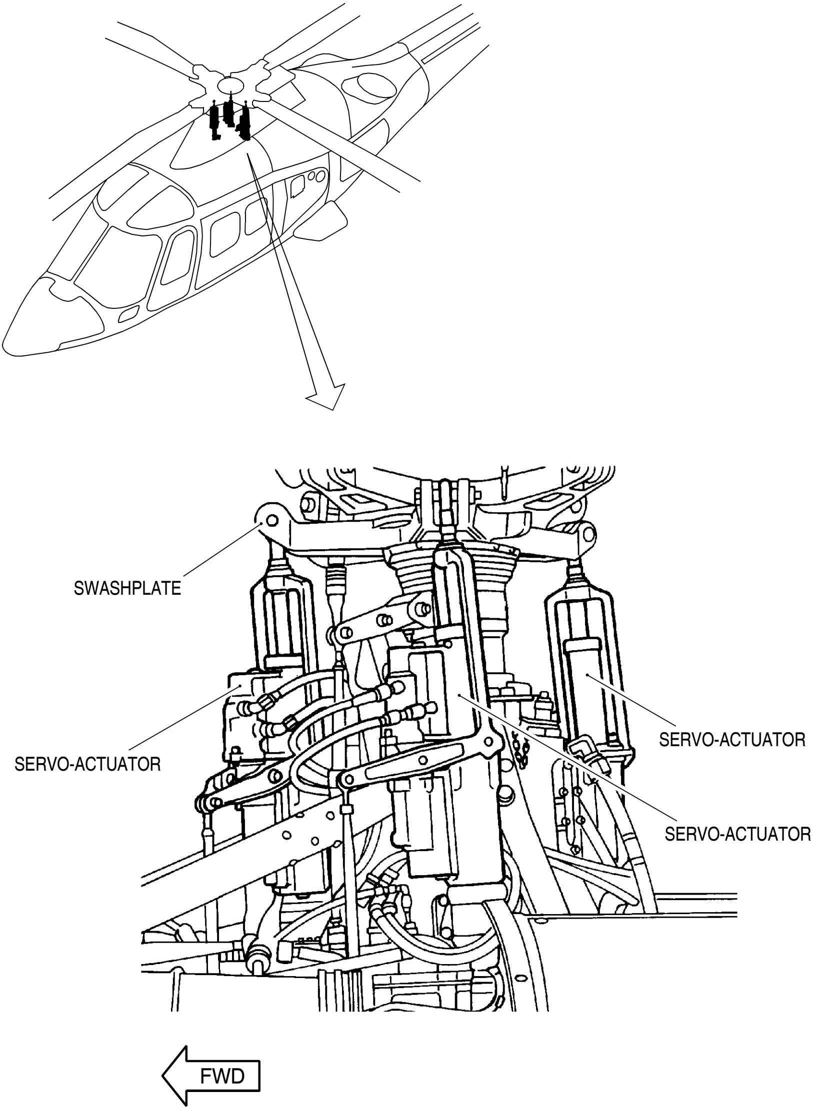

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Figure 7-145 Main Rotor Servo-actuators - Component Location

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POWER PLANT

(Chapter 71)

GENERAL

The power plant installation includes:

– The engine installation
– The engine cowling installation
– The engine mount installation
– The engine firewall installation
– The engine air intakes installation
– The engine drain installation.

The two engines are housed in a compartment located aft of the transmission. A front and rear firewall enclose the engine compartment and a central firewall divides it in two parts, thus separating the two engines. The engines are protected by cowlings and are attached to the relevant mount at three points.
The engines are equipped with drain tubes which protrude from the underside of the helicopter and allow the drainage of the fuel and lubricant.

ENGINE INSTALLATION

The engine installation consists of the two Pratt & Whitney Canada 207C engines (Figure 7-146).
The PW207C is a lightweight, free-turbine, turboshaft engine incorporating a single stage centrifugal compressor driven by a single stage turbine. Metered fuel from the Fuel Management Module (FMM) is sprayed into a reverse flow annular combustion-chamber through twelve fuel nozzles mounted around the gas generator case. A high ignition unit and fuel spark igniters are used to start combustion. A single-channel Full Authority Digital Electronic Control (FADEC) system with a mechanical back up FMM ensures accurate control of the engine output speed and fast response to changes in power demand.
An electrical torque motor located whiting the FMM works in conjunction with the electronic Engine Control Unit (ECU).

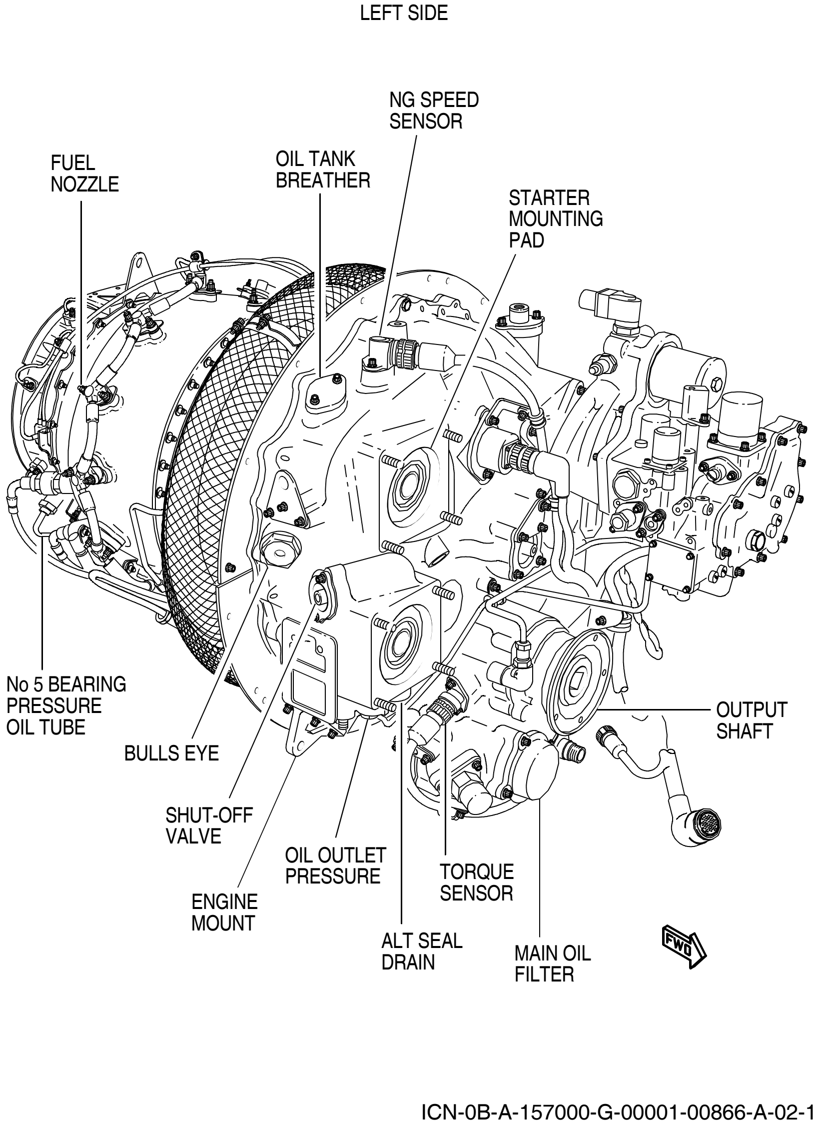

Figure 7-146 Engine Installation (Sheet 1 of 2)

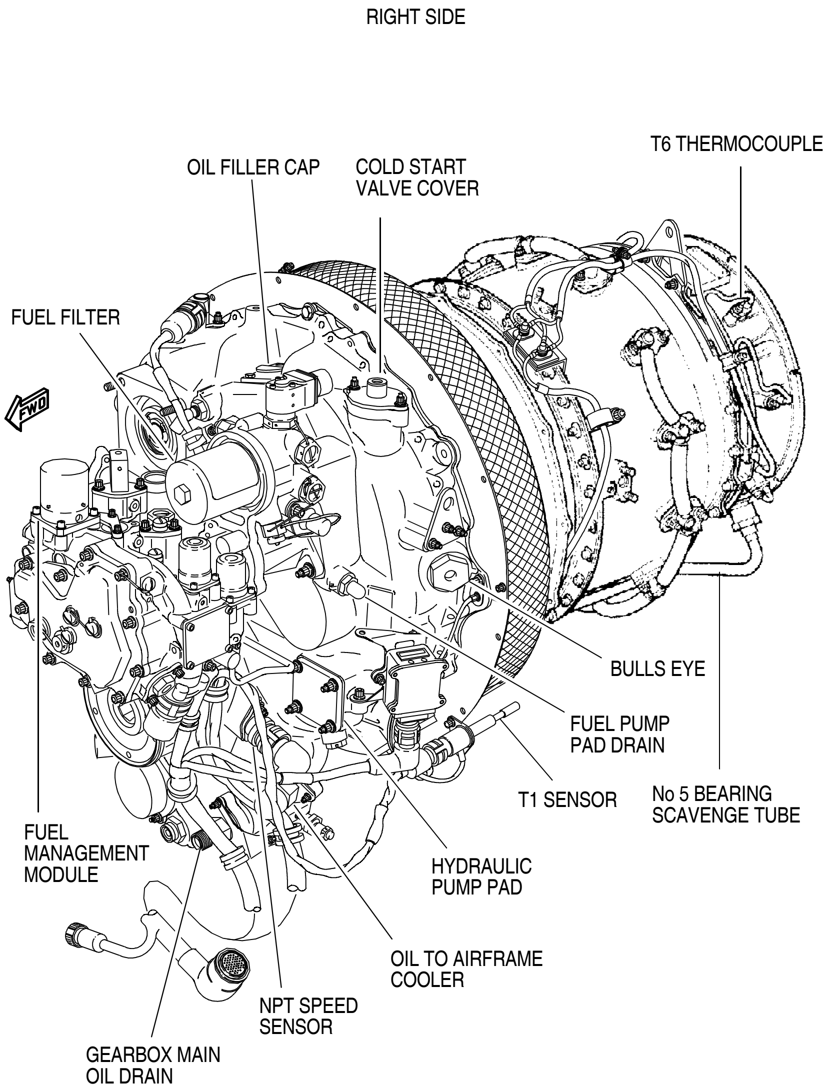

Figure 7-146 Engine Installation (Sheet 2 of 2)

Inlet air enters through a radial inlet plenum chamber, formed by the compressor inlet case, where it is directed rearward to the centrifugal impeller. The high pressure air from the centrifugal impeller passes through diffuser tubes which turn the air through 90 degrees in direction and convert velocity to static pressure. This high pressure air sorrounds the combustion chamber.
The combustion chamber consists of two annular welded sections bolted together at the rear end; the outer liner incorporates an integral exit duct. The liner assembly has perforations of various sizes that allow entry of compressor delivery air. The flow of air reverses direction as it goes through the compressor and the power turbines.
Fuel is injected into the combustion chamber liner through twelve fuel nozzles arranged evenly around the gas generator case. The nozzles are supplied with fuel by a flow divider. The combustion gases expand from the combustion chamber, reverse direction in the duct zone and go through the compressor turbine stator and turbine vanes to the single stage compressor turbine. The stator vanes ensure that the expanding gases impinge on the turbine blades at the correct angle, with a minimum loss of energy. The still expanding gases continue rearward to the power turbine stator and turbine. The exhaust gases from the power turbine are directed through an annular exhaust chamber to the atmosphere.

The power turbine drives the output shaft through a two-stage reduction gearbox located at the front of the engine. The gearbox incorporates a phase-shift torque meter device with sensors which provides inputs to the ECU for control cockpit indication of engine torque.

All engines driven accessories are mounted on the reduction gearbox. A gas generator speed sensor is mounted on the upper left section of the reduction gearbox front housing. This sensor is located directly over the starter generator drive spur gear. When the spur gear passes by the sensor, it creates impulses which are transmitted to the ECU. The operation of the power turbine sensor is the same as the gas generator speed sensor, but this sensor receives its signal from the torque shaft.
The engine oil supply comes from an integral oil tank.
The engines are installed in a compartment located aft of the main transmission by means of mounts. The engine compartment is encloses between two transverse firewalls, while a longitudinal firewall separates the two engines.

ENGINE COWLING INSTALLATION

The engine cowling installation consists of the engine compartment side cowling, upper cowling, rear cowling and exhaust ejectors (Figure 7-147).
The side cowlings are hinged to the central beam of the engine compartment and are secured to the structure by latches.
Four quick release pins allow the central beam to be disconnected from the fittings on the engine front and rear firewalls to which it is attached.
Each side cowling is also provided with a supporting rod that permits the cowling to be held open. The upper and rear cowlings are attached to the rear firewall and to the fuselage structure with fasteners.
The rear cowling is provided with an inspection door.
The exhaust ejectors, one for each engine, protrude from the rear cowling and are secured to the rear firewall by bolts.

ENGINE MOUNT INSTALLATION

The helicopter is equipped with the mounts for each engine. Each mount installation consists of two V-shaped support and two rear leg supports with fittings (Figure 7-148).
The V-shaped support is attached to the helicopter structure with bolts and nuts; it provides the engine inboard attachment point.
The supports with the fitting, secure the engine outboard attachment points.
The rear leg provides the rear engine attachment point.

ENGINE FIREWALL INSTALLATION

The engine firewall installation consists of a forward, a central and a rear firewall (Figure 7-149).
The firewalls are made of titanium alloy and are riveted to the structure.
The forward firewall has removable panels for the maintenance of the main drive shafts. The rear firewall has the openings for the engine exhaust duct.
The upper edge of the firewalls provides the attachment points for the cowlings.
The central firewall is secured to the forward and rear firewalls by bolts.

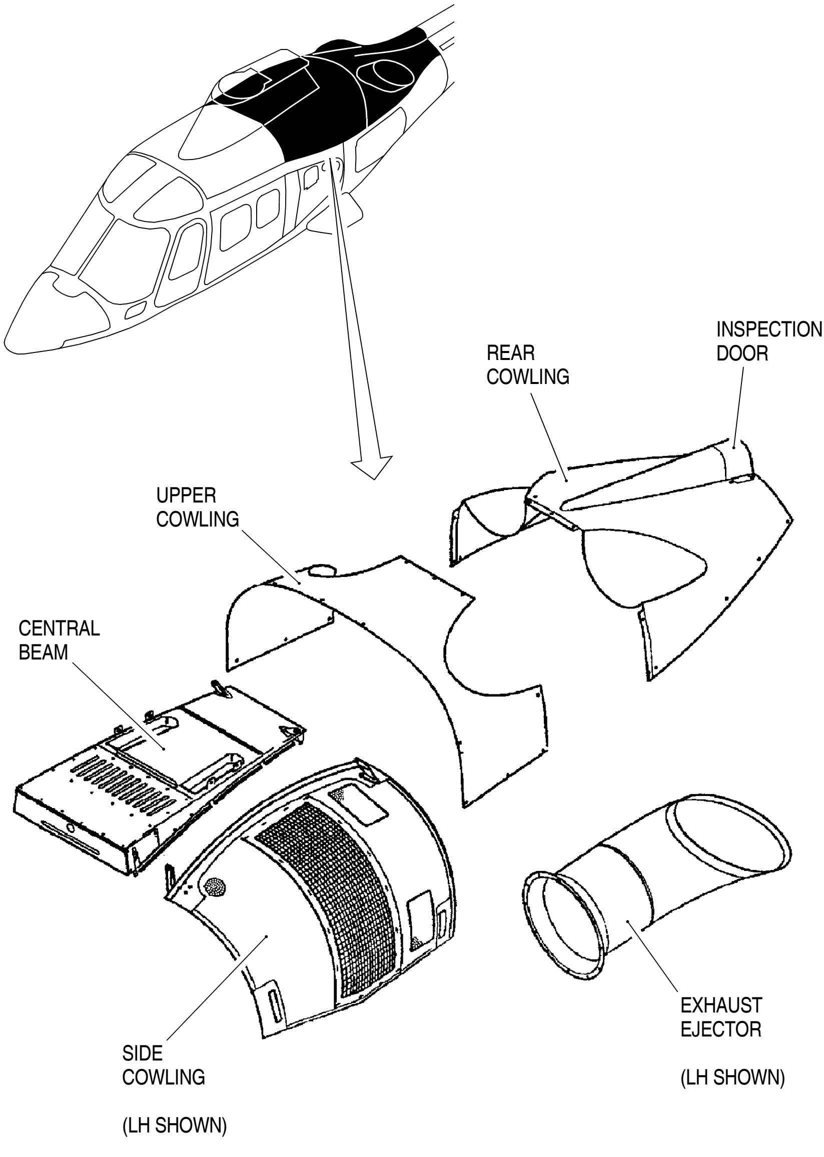

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Figure 7-147 Engine Cowling Installation

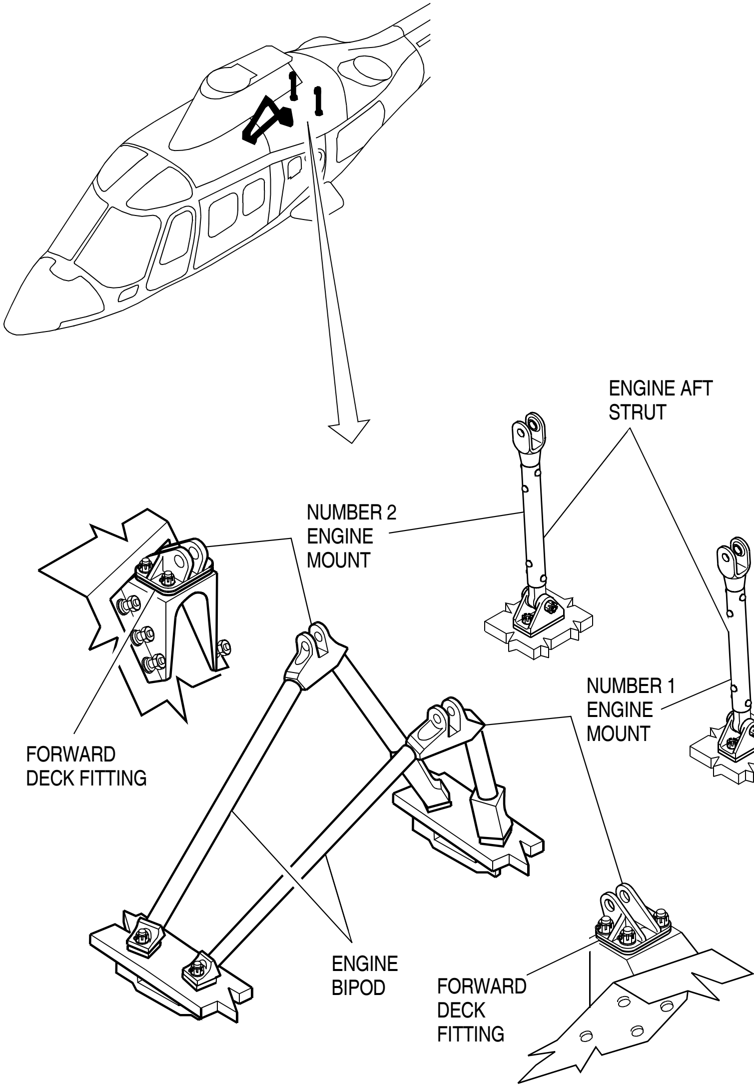

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Figure 7-148 Engine Mount Installation

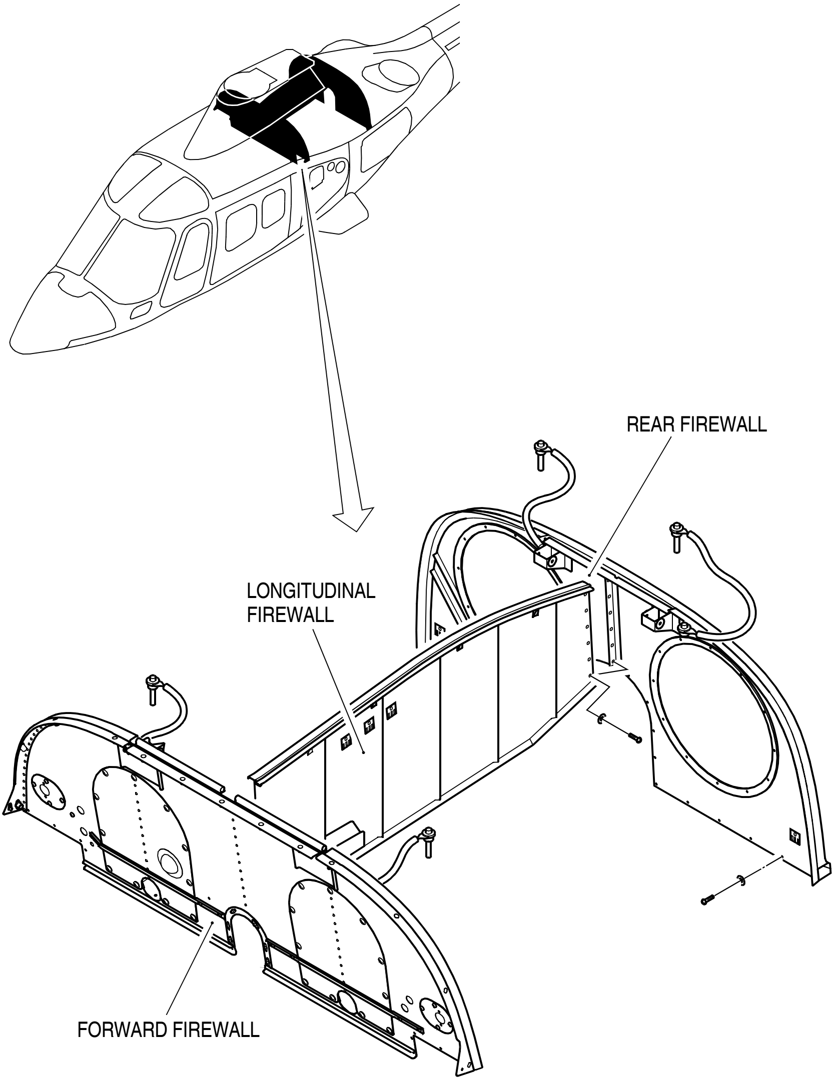

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Figure 7-149 Engine Firewall Installation

AIR INTAKE INSTALLATION

Each engine air intake consists of a vane, made in two halves for installation purposes, installed on the engine around its inlet screen. The air intake is closed by the side cowling on which is installed the air intake net. The net is attached to the side cowling with screws.
The side cowling is hinged on a central beam and can easily be opened by means of the latches.

ENGINE DRAIN INSTALLATION

Each engine is fitted with pipes that allow lubricant and fuel to be drained. The fluid from these pipes is collected in two drain cups installed on the engine compartment floor.
The cups are interconnected by a tube which protrudes from the underside of the helicopter. Each engine compartment is also fitted with two holes allowing the draining of the fluids which would otherwise collect on the compartment floor. These holes are inter connected by a tube which protrudes from the underside of the helicopter.

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AIR

(Chapter 75)

GENERAL

The air system includes the Engine Air Particle Separator system (EAPS).

ENGINE AIR PARTICLE SEPARATOR SYSTEM

The EAPS system provides clean air to the engines, by separating the particles of sand and dust present in the air, from the air flow sucked through the air-intakes.

The EAPS system consists of a right and a left particle separator installed symmetrically on the right and left engine air intake.

The primary components of the EAPS systems are:

– The two particle separators
– The two particle separator fairings
– The two shut-off valves
– The two pressure switches
– The two bleed lines.

The main components of the engine air particle separator system are shown in Figure 7-149A.

Each particle separator consists of a group of tube assemblies arranged in parallel and fixed between two aluminium sheets. Each tube consists of a vortex tube, vortex generator and outlet tube.

The function of the EAPS is to separate the particles of sand, dust, snow, ice, and other particles present in the air, from the air flow sucked by the engines through the helicopter air intakes, by means of the scrolls that are part of the filter itself. This function is needed in order to prevent erosion of engine compressor blades and unwanted deposits on engine combustion chamber and other engine parts.

The particles, heavier than air, are moved towards the external walls from the scrolls centrifugal effect and are carried inside the separator.

From here, they are discharged outboard through the ejectors, activated by pressurized air bled from engine ports.

When the ejectors are not activated, the scrolls are however able to partially separate the sand (efficiency drops to approximately 50%), which accumulates inside the separator plenums without being released outside.

When the entire plenums of the separators are filled, sand is not separated anymore. In this case, the sand will enter the engine through the inlet, but there is no possibility that the airflow blocks completely, because the particle separator scrolls are never totally occluded.

The pressure switch senses the bleed air pressure in the relative bleed line and allows monitoring of the pressure downstream of the related shut-off valve before it reaches the particle exhaust system in the particle separator.

The separator cannot be operated during engine ignition, because the engine, in this phase, does not allow bleeding. Therefore, it is necessary that the engine reaches the IDLE condition, before operating the relevant EAPS system.

When the particle separator is activated, the copilot's EDU displays the green advisory EAPS ON.
When the particle separator is activated and the pressure switch gives low pressure, the pilot's EDU displays the yellow caution EAPS PRESS and the MASTER CAUTION illuminates.
When the particle separator is not activated and the pressure switch gives high pressure, the pilot's EDU displays the yellow caution EAPS PRESS and the MASTER CAUTION illuminates.

The system is powered through the switches that follows:

– EAPS 1 (MAIN BUS #1)
– EAPS 2 (MAIN BUS #2).

ENGINE AIR PARTICLE SEPARATOR CONTROLS AND DISPLAYS

Refer to Figure 7-149B.

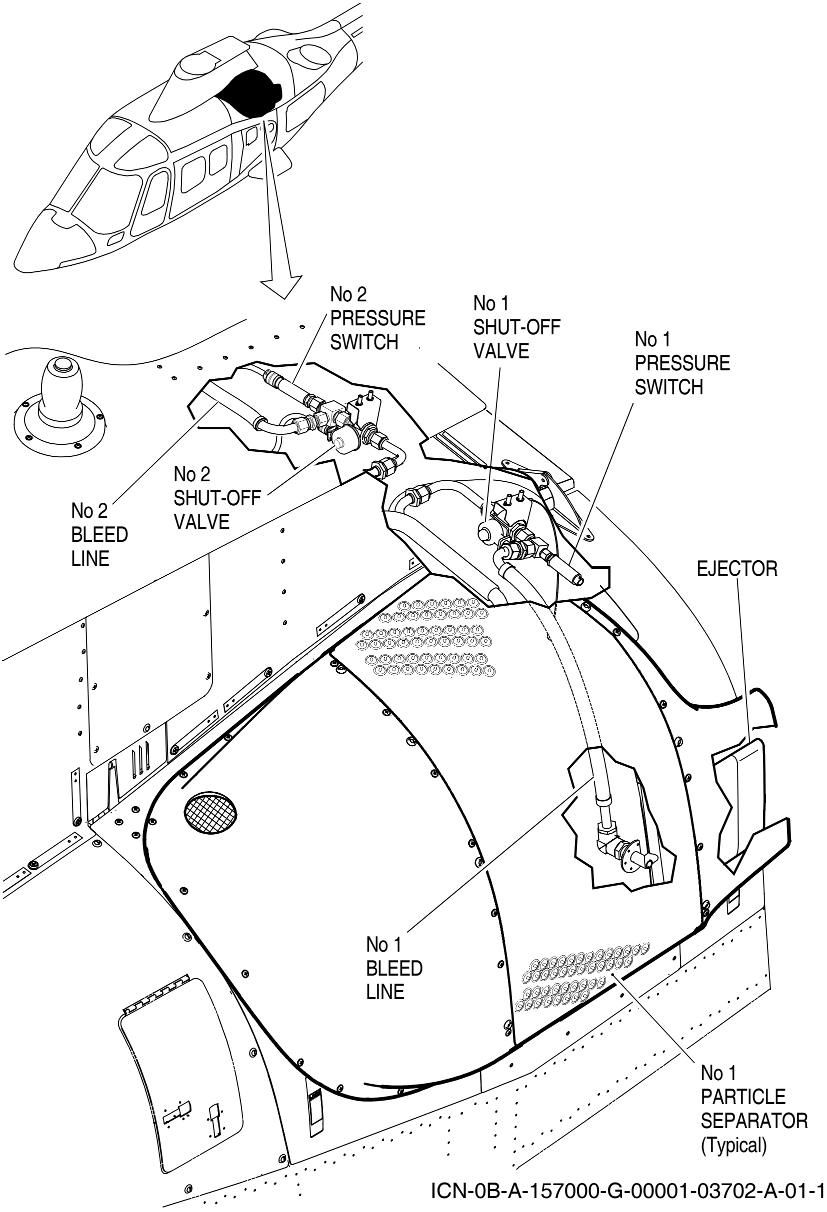

Figure 7-149A Engine Air Particle Separator - Component Location

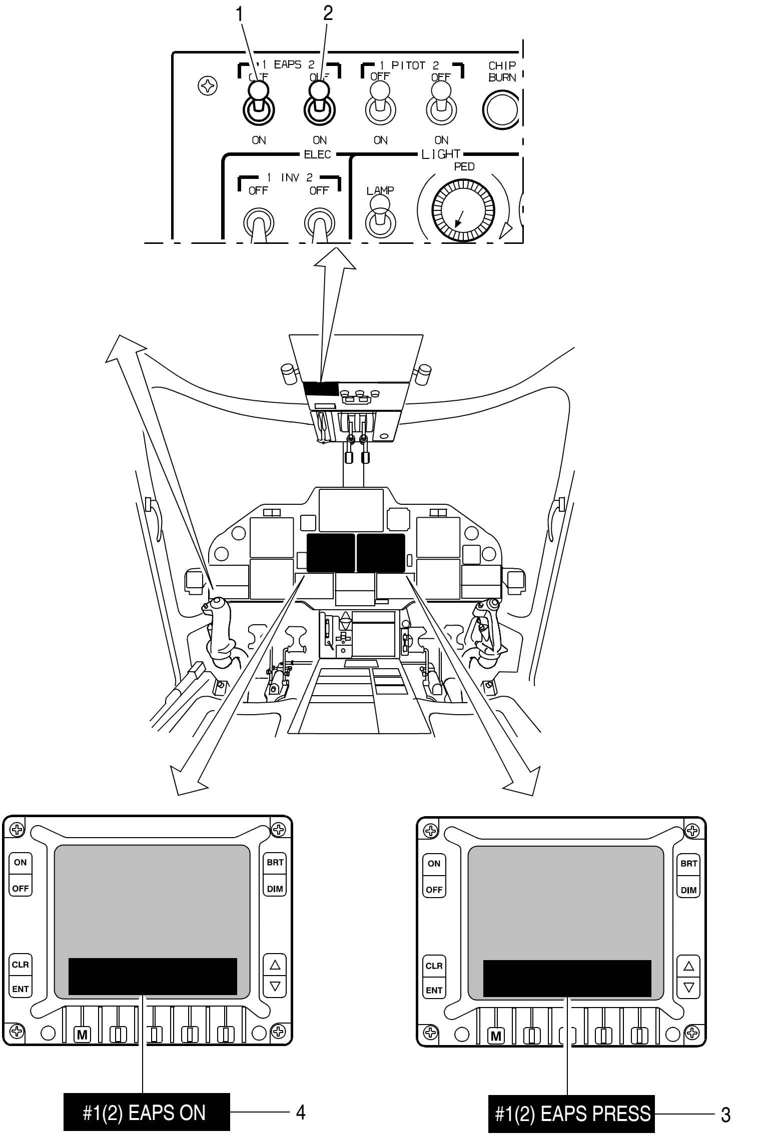

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Figure 7-149B Engine Air Particle Separator - Controls and Displays

Key to Figure 7-149B

Ref.	Controls/displays	Function
1	EAPS 1 switch	ON - The No 1 shut-off valve is open. When open, the shut-off valve allow the pressurized engine bleed air pass through the bleed line and supply the particle separation operation. OFF - The No 1 shut-off valve is closed.
2	EAPS 2 switch	ON - The No 2 shut-off valve is open. When open, the shut-off valve allow the pressurized engine bleed air pass through the bleed line and supply the particle separation operation. OFF - The No 2 shut-off valve is closed.
3	1 (#2) EAPS PRESS caution message	When the EAPS 1 (or 2) switch is set to ON, shown a condition of non-opening of the No 1 (or the No 2) shut-off valve (low pressure in the No 1 (or No 2) pressure switch). When the EAPS 1 (or 2) switch is set to OFF, shown a condition of non-closure of the No 1 (or the No 2) shut-off valve (high pressure in the No 1 (or No 2) pressure switch).
4	#1 (#2) EAPS ON advisory message	Shown when the No 1 (or the No 2) EAPS switch is set to ON. No 1 (or No 2) particle separator activated.

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ENGINE CONTROLS

(Chapter 76)

GENERAL

The engines are equipped with a single channel Full Authority Digital Electronic Control (FADEC) system. This system incorporates all the units that provide for complete automatic and manual control of the engines.

ENGINE POWER CONTROL SYSTEM

The engine power control system includes:

– The engine power lever quadrant
– The two Engine Control Units (ECU)
– The two Fuel Management Modules (FMM)
– The control panel with the Power Management Switches (PMS).

The system is powered through the circuit breakers that follows:

– GOV CONTR (ESS BUS #2)
– ECU 1 (ESS BUS #1)
– FIRE DET 1 (ESS BUS #1)
– ECU 2 (ESS BUS #2)
– FIRE DET 2 (ESS BUS #2).

The block diagram of the engine power control system is shown in Figure 7-150. Refer to Figure 7-151 for the schematic diagram of the engine power control system.

ENGINE POWER LEVER QUADRANT

The engine power lever quadrant comprises the two power levers, controlled either manually or electrically through actuators that are in turn controlled by the Linear Variable Differential Transducers (LVDT). The transducers are connected to the collective levers.

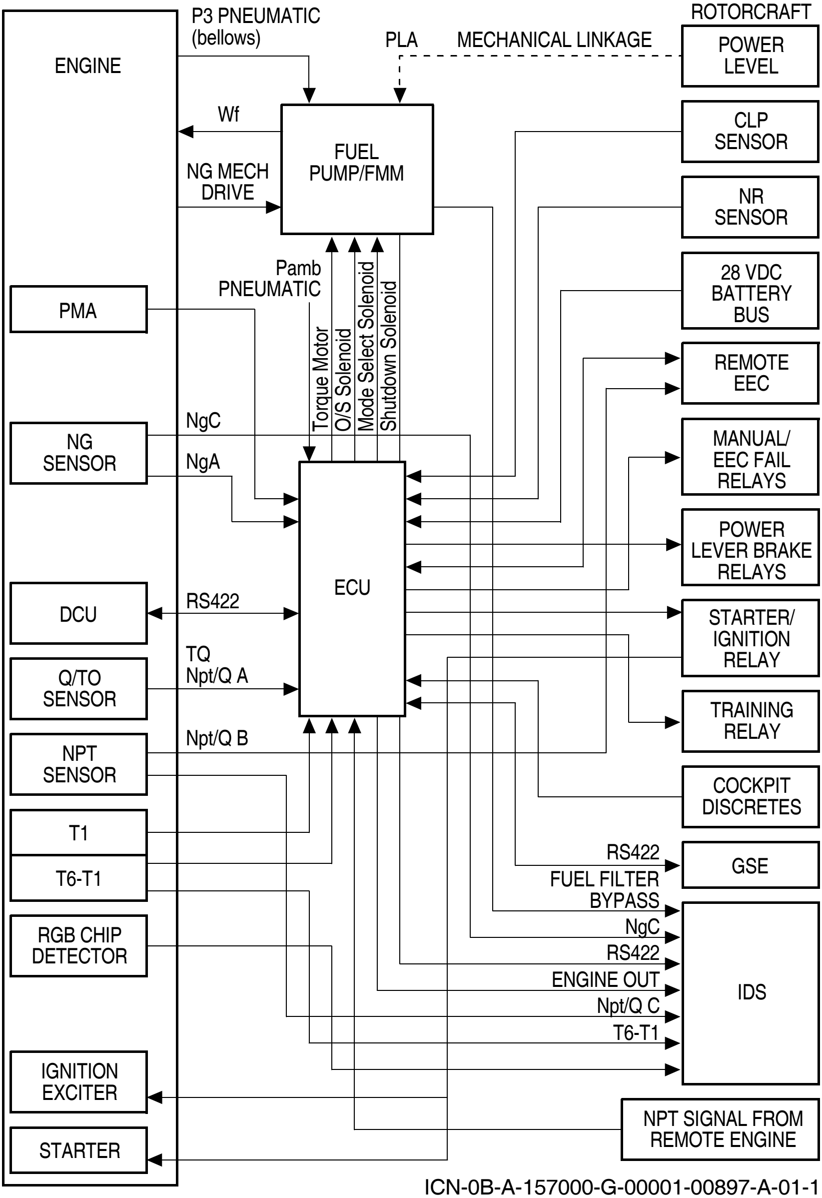

Figure 7-150 Engine Controls System - FADEC Functional Interface Diagram

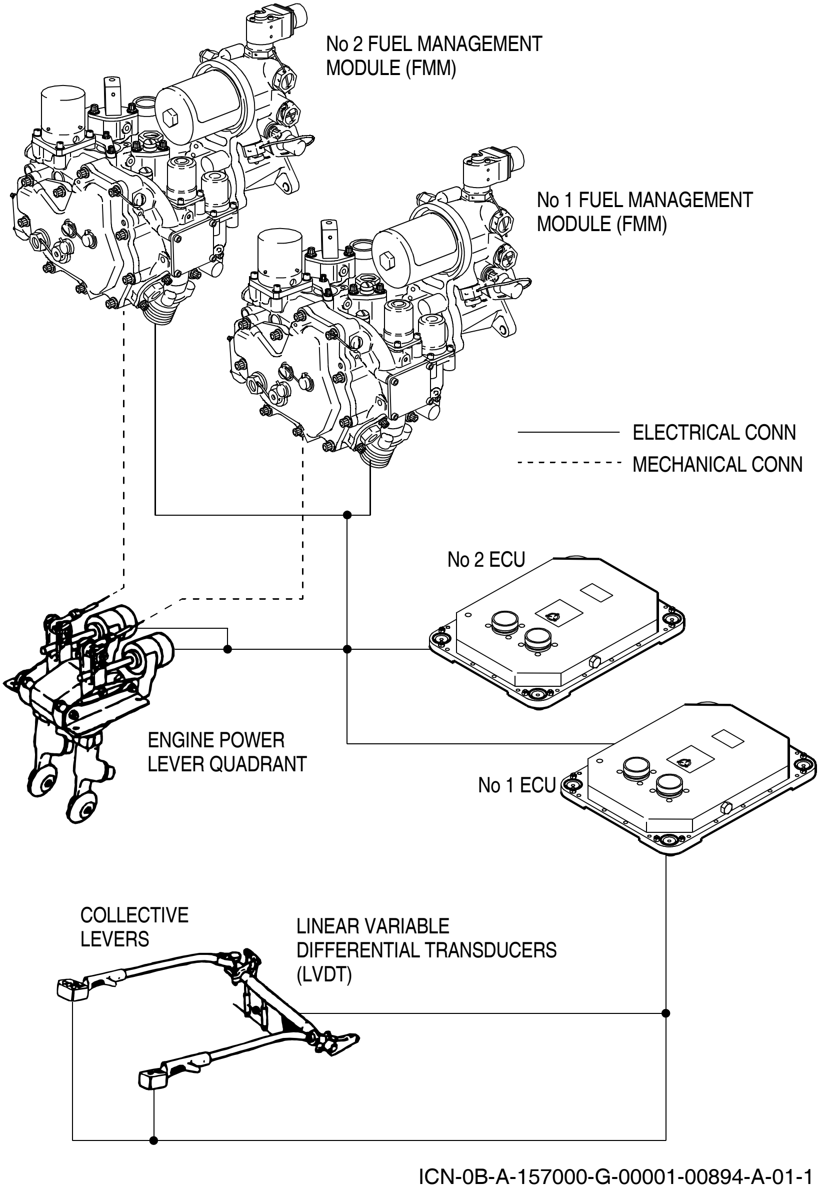

Figure 7-151 Engine Controls System - Schematic Diagram

The engine power levers, connected to each engine FMM through a cable, are provided with an engine starter push button that is used for engine starting in the backup mode. A red fire warning light is installed in the lever handgrip.

The power lever operates as follows:

– When the engine is controlled by the FADEC, the engine power is modulated by the FADEC independent of the lever position that is always the FLIGHT position.
– When the engine is not controlled by the FADEC (manual control), the engine lever controls the engine power directly, and the pilot can set it between IDLE and MAX. Under these conditions the FLIGHT position is a setting near “maximum continuos power”.

ENGINE CONTROL UNIT (ECU)

There are two ECUs in the helicopter.

Each ECU is a single-channel digital electronic unit that, together with the related FMM and a network of sensors, controls the engine gas generator and power turbine speeds in response to the load demanded by the rotor system.

The ECU also cuts off the fuel to the related engine when the main transmission torque exceeds 110% with the two engines operative, or 156% with one engine inoperative. In addition it measures the time the helicopter stays in the range of “6 seconds” operation, and after 4 seconds supplies the instant signal that causes the XMS OVRTRQ caution message to start flashing.

FUEL MANAGEMENT MODULE (FMM)

The FMMs is installed in the accessory gearbox of each engine. The FMM is an electro-mechanical unit which governs the fuel flow through the entire operational envelope of the engine. The input parameters of the FMM are the related ECU signals, the power lever angle and the ambient pressure.
The FMM has a fuel pump, which delivers high pressure fuel to the metering portion of the unit.

ENGINE POWER CONTROLS SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-152.

The several warning and caution messages inform the pilots about specific conditions and malfunctions of the engine power controls system (refer to Chapter 77 Engine indicating).

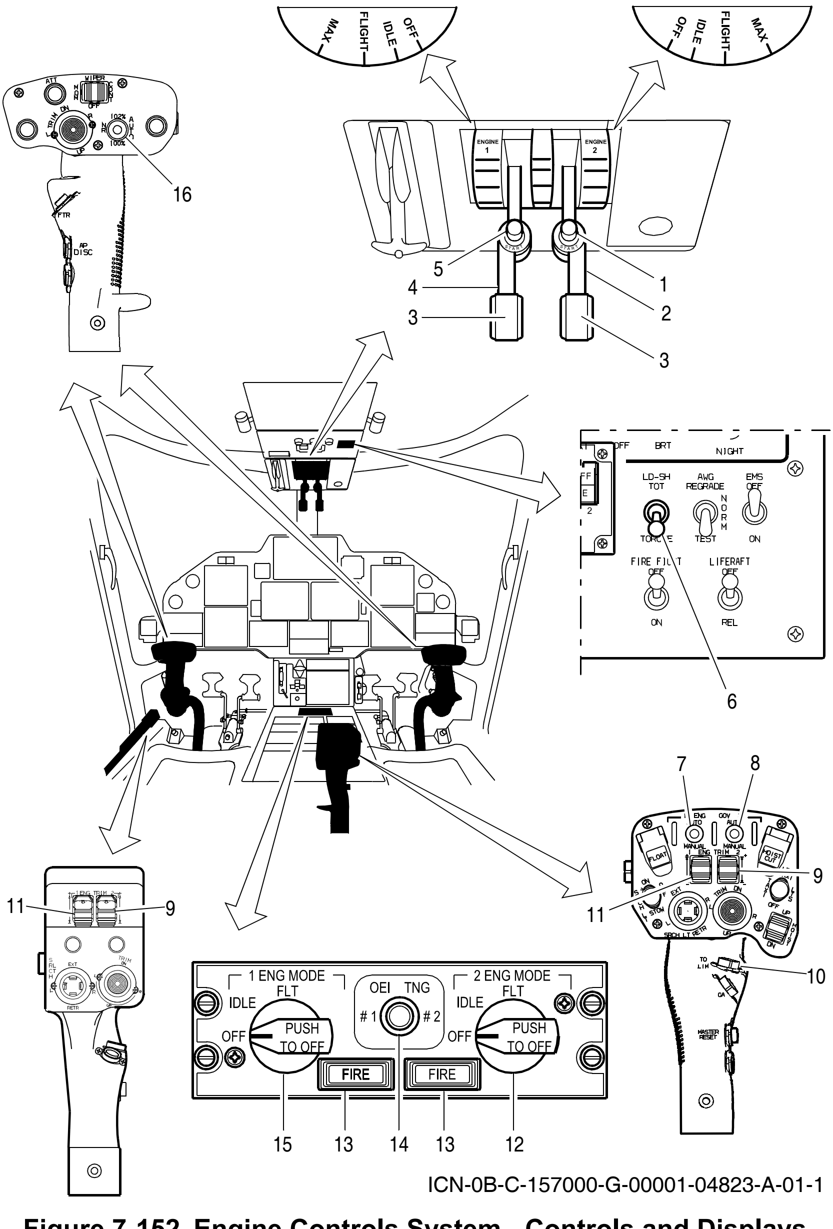

Figure 7-152 Engine Controls System - Controls and Displays

Key to Figure 7-152

Ref.	Control/Display	Function
1	STARTER push-button (No 2 engine)	Pushed - starts the No 2 engine in manual backup mode.
2	No 2 engine power lever	OFF position - the No 2 engine is cut off. IDLE position - the No 2 engine operates at idle. FLIGHT position - the No 2 engine operates at maximum continuos power. MAX position - the No 2 engine operates at maximum power.
3	Fire warning light	Comes on (red light), when a fire condition is detected, in the No 1 or No 2 engine compartment.
4	No 1 engine power lever	OFF position - the engine is cut oft. IDLE position - the No 1 engine operates at idle. FLIGHT position - the No 1 engine operates at maximum continuos power. MAX position - the No 1 engine operates at maximum power.
5	STARTER push-button (No 1 engine)	Pushed - starts the No 1 engine in manual backup mode.
6	LD-SH switch	TORQUE - The ECUs modulate the engine power to keep the engines torqued matched. TOT - The ECUs modulate the engine power to keep the Turbine Outlet Temperature (TOT) matched.
7	ENG 1 GOV switch	AUTO - the No 1 engine is automatically controlled by the FADEC system. MANUAL - the No 1 engine must be controlled manually through the power lever.
8	ENG 2 GOV switch	AUTO - the No 2 engine is automatically controlled by the FADEC system. MANUAL - the No 2 engine must be controlled manually through the power lever.

Key to Figure 7-152 (Continued)

Ref.	Control/Display	Function
9	ENG 2 trim switch	Pushed forward - controls the No 2 engine power lever to increase engine power. Pushed aft - controls the No 2 engine power lever to decrease engine power. Note 1: It is not possible to move electrically the engine power lever from IDLE to OFF. Note 2: It is possible to move electrically the engine power lever from FLIGHT to MAX only if the ENG 2 CONT switch is in MANUAL position.
10	TQ LIM push-button switch	Pushed and released - the ECUs activate the torque limiter function. When this function is activated, the EDU2 shows the LIMITER ON advisory message. Note: To de-activate the torque limiter function, the push-button must be pushed and released again.
11	ENG 1 trim switch	Pushed forward - controls the No 1 engine power lever to increase engine power. Pushed aft - controls the No 1 engine power lever to decrease engine power. Note 1: It is not possible to move electrically the engine power lever from IDLE to OFF. Note 2: It is possible to move electrically the engine power lever from FLIGHT to MAX only if the ENG 1 CONT switch is in MANUAL position.
12	2 ENG MODE selector knob	Selects the operating condition of the No 2 engine in the automatic mode (OFF, IDLE and FLIGHT position). To select the OFF position from IDLE, the knob must be pushed.
13	FIRE warning light	Comes on (red light), when a fire condition is detected, in the No 1 or No 2 engine compartment.

Key to Figure 7-152 (Continued)

Ref.	Control/Display	Function
14	OEI TNG selector switch	Allows OEI (One Engine Inoperative) operation for training purposes. Central position - the OEI function is not selected. # 1 - the No 1 engine is inoperative. # 2 - the No 2 engine is inoperative.
15	1 ENG MODE selector knob	Selects the operating condition of No 1 engine in the automatic mode (OFF, IDLE and FLIGHT position). To select the OFF position from IDLE, the knob must be pushed.
16	NR switch	100% - the rotor operates at 100% RPM. 102% - the rotor operates at 102% RPM. AUTO - the rotor automatically operates at 100% RPM or at 102% RPM during take-off or landing, depending on the landing gear position and helicopter velocity.

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ENGINE INDICATING

(Chapter 77)

GENERAL

The engine indicating system includes:

– The engine power monitoring system
– The engine temperature monitoring system
– The engine data display system.

The engines supply signals to the ECU which sends the engine data to the Data Acquisition Unit (DAU).
The DAU supplies engine data to the pilot’s EDU that displays the primary engine parameters:

– N1 (Gas generator speed)
– TOT (Turbine Outlet Temperature)
– TRQ (Torque)
– N2 (Power turbine speed)
– NR (Rotor speed).

The rotor speed data (NR) are sent to the DAU and ECU by a pick-up located on the main transmission base.
The pilot’s EDU also displays the engine warning message ENG 1 OUT and ENG 2 OUT when N1 of the related engine is abnormally low (below 35%). In addition the aural messages “ENGINE ONE OUT” and “ENGINE TWO OUT” are transmitted to the crew through the ICS system.

ENGINE INDICATING SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-153.

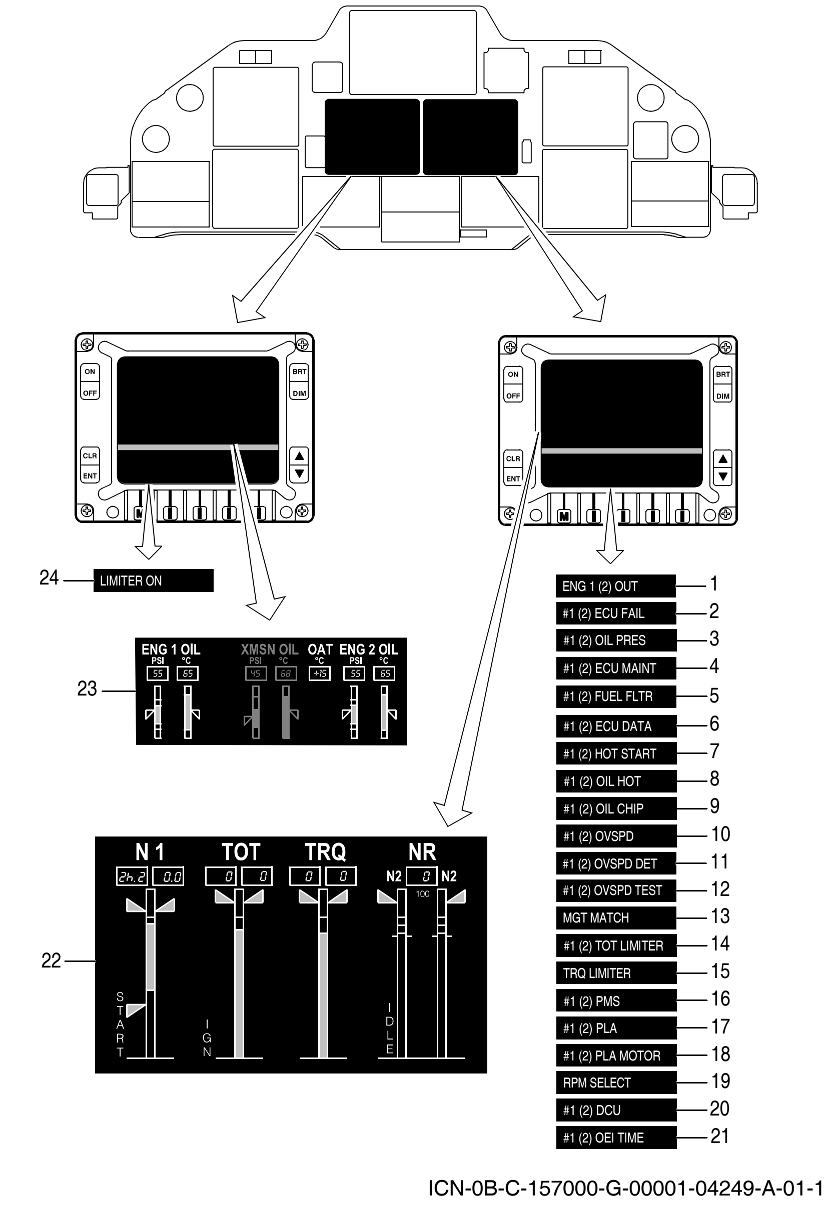

Figure 7-153 Engine Indicating System - Controls and Displays

Key to Figure 7-153

Ref.	Controls/displays	Function
1	ENG1 (2) OUT warning message	Displayed in case of engine failure (N1 <35%).
2	#1 (#2) ECU FAIL warning message	Displayed when a critical failure is detected in the ECU 1 (2).
3	#1 (2) OIL PRES warning message	Displayed when the No 1 (2) engine oil pressure is lower or higher than the limits stored in the DAU.
4	#1 (2) ECU MAINT caution message	Displayed when a non critical failure is detected in the ECU 1 (2).
5	#1 (2) FUEL FLTR caution message	Displayed when the No.1 (2) engine fuel filter is clogged.
6	#1 (2) ECU DATA caution message	Displayed when some of the ECU data are lost in the ECU 1 (2).
7	#1 (2) HOT START caution message	Displayed when during starting the TOT is high.
8	#1 (2) OIL HOT caution message	Displayed when the No.1 (2) engine oil temperature is high.
9	#1 (2) OIL CHIP caution message	Displayed when chips are detected in the No.1 (2) engine oil.
10	#1 (2) OVSPD caution message	Displayed when the overspeed protection system cuts-in.
11	#1 (2) OVSPD DET caution message	Displayed when an overspeed protection system failure is detected.
12	#1 (2) OVSPD TEST caution message	Displayed when the test of the overspeed protection system detects a failure. This message appears only on ground before engine starting.
13	MGT MATCH caution message	Displayed when the engine matching is not possible due to ECU failure or to TOT.
14	#1 (2) TOT LIMITER caution message	Displayed when the TOT matching is not possible.
15	#1 (2) TQR LIMITER caution message	Displayed when the TQR matching is not possible.
16	#1 (2) PMS caution message	Displayed when a PMS failure is detected

Key to Figure 7-153 (Continued)

Ref.	Controls/displays	Function
17	#1 (2) PLA caution message	Displayed when the PLA (Power Lever Angle) is out of the FLIGHT position.
18	#1 (2) PLA MOTOR caution message	Displayed when a failure of the remote control on the collective stick is detected.
19	RPM SELECT caution message	Displayed when the NR selector switch is inoperative or the setting of NR is impossible.
20	#1 (2) DCU caution message	Displayed when a failure of the No 1 (2) DCU is detected. This message appears only on ground before engine starting.
21	#1 (2) OEI TIME caution message	Displayed when the time of 2.5 minutes OEI for the No 1 (2) engine is reached.
22	N1, TOT, TRQ and NR indications	Displayed in digital and analog form on the pilot's EDU as primary data.
23	ENG 1 (2) OIL pressure and temperature indications	Displayed when pressure and temperature indications in digital and analog form on the copilot's EDU as secondary data (main page).
24	LIMITER ON advisory message	Displayed when the ECUs activate the engine torque limiter function.

ENGINE POWER MONITORING SYSTEM

The engine power monitoring system includes (Figure 7-154):

– The torque monitoring system
– The power turbine RPM monitoring system
– The gas generator RPM monitoring system.

The sensors installed on the engines transmit the sensed values to the ECU. The primary data (N1, N2, TOT and TRQ) are then transmitted directly to the EDUs (in normal operating conditions the data are displayed on the pilot's EDU only). A secondary signal sends the data to the DAU to permit the operation of the backup channel in case of primary signal failure.

The backup channel receives the primary data signal (N1, N2 and TOT) from the sensors. If the primary signal fails the DAU processes the data and sends them to the EDU. The backup TRQ is calculated using N2, some other engine parameters and a special algorithm.
All secondary engine data are sent to the DAU for processing to that they become available to the related EDU.

TORQUE MONITORING SYSTEM

The torque monitoring system displays, for each engine, the power turbine torque in percent referred to half the transmission maximum rating (100% engine torque corresponds to 450 Hp). The system consists of two torque sensors, one for each engine, located on the accessory gearbox. The torque sensors are electrically connected to the two ECUs which interfaced with the DAU. In normal operating conditions the torque signal is sent to the EDU and also to the DAU for backup data storage. In case of an ECU failure (ECU FAIL caution message in view) the signal is not sent from the ECU and is calculated from the DAU analog backup of the N2 sensor and by applying the algorithm. In this case the supplied value can differ up to 7% from the value displayed when the EDU is operative. Besides when the engine is at IDLE or when the collective lever is completely down (MPOG), there can be torque fluctuations as large as 20% of the actual value. When the engine operates in the normal range, the torque fluctuactions are only a small percent. The large fluctuations are due to the algorithm inaccuracy at low power range.

POWER TURBINE RPM MONITORING SYSTEM

The power turbine RPM monitoring system permits the rotational speed of the power turbine (N2) of each engine to be displayed in percent of the maximum rating.
The system consists of two speed sensors, one for each engine, located in the accessory gearbox.
The speed sensor senses the passage of spur gear tooth form to generate output signal for conversion into power turbine speed (N2).
The speed sensor is a triple coil magnetic sensor and is electrically connected to the ECU and DAU.
The first coil of speed sensor supplies the power turbine speed data, the second coil provides the overspeed protection, the third is a stand-by coil.
The pilot’s EDU displays the power turbine and rotor speed (N1 and NR/N2) in related screen field. Two analog indicators, one for each engine, show N1, while NR/N2 is displayed on a central digital readout and two analog indicators.

GAS GENERATOR SPEED MONITORING SYSTEM

The gas generator RPM monitoring system permits the rotational speed of the gas generator (N1) of each engine to be displayed.
The system consists of a speed sensor located on of the upper side of the accessory gearbox.

The speed sensor senses the gear teeth passage of the gear wheels in the accessory gearbox as its speed reference. It generates an output signal for conversion into the gas generator speed (N1).

The speed sensor is a triple coil magnetic sensor and is electrically connected to the ECU and DAU. One coil of the speed sensor supplies the speed data for the gas generator speed, the other two coils provide the speed signals for the N1 speed governor and overspeed protection respectively.
The pilot’s EDU displays the gas generator speed (N1) in the related screen field. Two analog indicators and digital readouts display the gas generator speed in percent of the maximum rating.

ENGINE TEMPERATURE MONITORING SYSTEM

The TOT monitoring system permits the gas temperature to be measured at the power turbine outlet of each engine.
The system consists of two identical and interchangeable harnesses, each having four insulated thermocouples (T6) connected in parallel in an engine-mounted junction-box (Figure 7-155).
Flexible wires from the two junction boxes carry the signals to the inlet temperature sensor box where they are integrated with the signal from the inlet thermocouple (T1) which is located near the inlet of each compressor. The T1 and T6 signals are sent to the ECU to be processed then they are sent to the EDU. The EDU displays the TOT in Celsius degrees on two indicators and digital readouts in the related screen field. In normal conditions, the EDU receives the signal directly from the ECU. In emergency conditions (ECU FAIL caution message in view) the signal sent to EDU is that processed by the DAU. The analog signal is received by the DAU directly from the sensors. In this case the displayed value can differ by up to 40° C from the indication given by the ECU when it is supplied directly by the EDU.

ENGINE DATA DISPLAY SYSTEM

The engines supply life data signals to the Data Collection Unit (DCU). The Data Collection Unit (DCU) through ECU and the DAU stores the engine-specific information supplied by the FADEC for subsequent recall. These data can be displayed when the MAINT page mode is selected on the EDU.

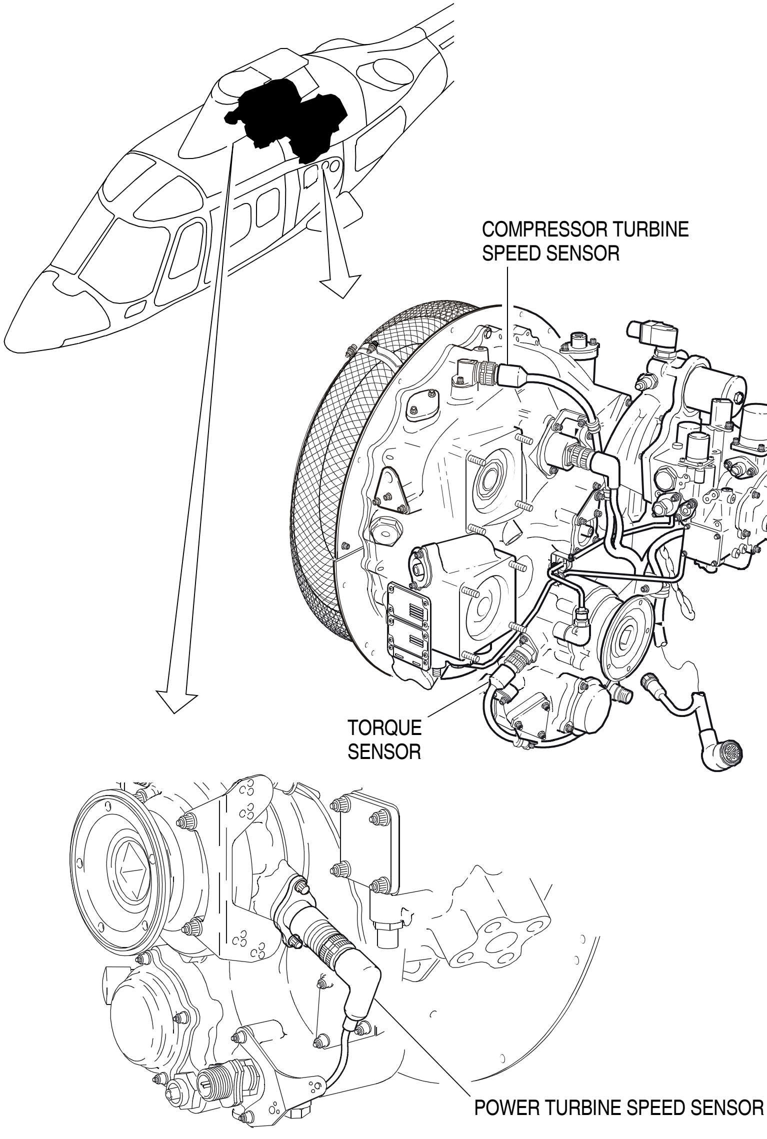

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Figure 7-154 Engine Power Monitoring System - Component Location

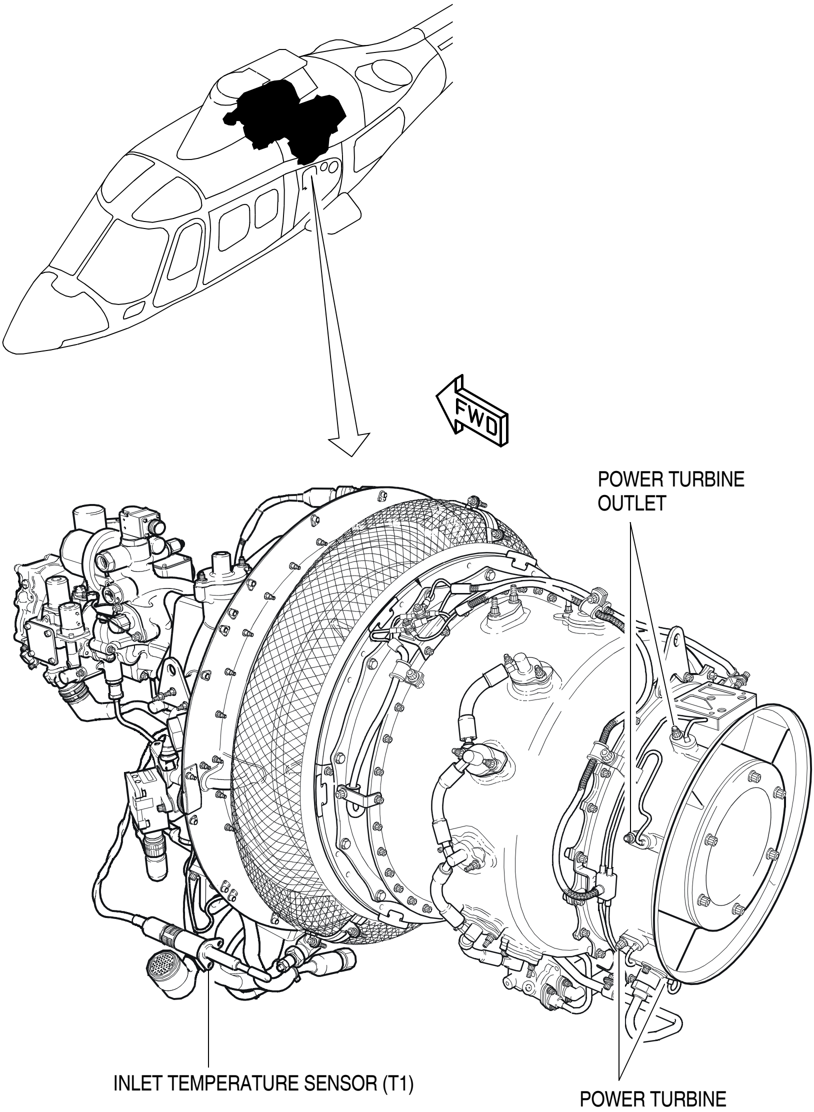

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Figure 7-155 Engine Temperature Monitoring System - Component Location

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IMAGE RECORDING

(Chapter 97)

GENERAL

The optional systems that follow can be installed on the helicopter:

– The external video camera EVS-1000
– The external video camera EVS-1500.

EXTERNAL VIDEO CAMERA (EVS-1000) SYSTEM

The external video camera system is a MAX VIZ EVS-1000 Enhanced Vision System (EVS). This system gives to the flight crew an enhanced visual awareness of terrain and possible obstacles in the forward field of view.

The EVS can operate at night, in smoke, haze and smog. It can also operate in many conditions of rain, snow and fog. The EVS lets the flight crew see in a better way:

– The ground vehicles and other ground-based equipment/ obstacles
– The aircraft on taxi-ways and runways
– The other traffic during takeoff, approach and landing
– The runway and terrain features during climb, descent and low altitude maneuvering.

The system includes a video camera and a power supply (Figure 7-156).

The video camera is installed below the helicopter nose with a protection fairing. The video camera is a long-wave infrared (8 to 14 μm) camera assembly that has a field of view of 53° horizontal by 40° vertical. The video camera uses heaters to prevent the ice formation on the lens. The video camera gives a RS-170 output video signal that is got through advanced Automatic Gain Control (AGC) algorithms. The video camera sends the RS-170 signal to the power supply.

The power supply, installed in the cockpit, gives the electrical power and all electrical interfaces to the video camera. The power supply sends the RS-170 video signal to the cockpit monitor through a coaxial cable. The cockpit monitor shows thermal scenes of the area in front of the helicopter.

The system is supplied with electrical power from the 28 V dc Number 2 main bus through the EVS circuit breaker.

OPERATION OF THE EXTERNAL VIDEO CAMERA SYSTEM

The external video camera system operates when the PWR and VID SEL buttons on the bezel of the cockpit monitor are pushed. After the power-up, approximately 20 s are necessary to have a usable image on the monitor. During this time a video test pattern is shown for approximately 5 s followed by the message OK. After approximately 20 s the monitor becomes black momentarily for a NUC cycle and then a usable image comes in the view.

To see the IR image in the better way the system does an automatic recalibration at intervals of 4 min. The recalibration process is known as Non-Uniformity Correction” (NUC). The NUC can be seen on the monitor as a momentary image interruption for approximately 1 s, while the CAL message comes in the view.

EXTERNAL VIDEO CAMERA SYSTEM CONTROLS AND DISPLAYS

Refer to description of the cockpit monitor (Chapter 31).

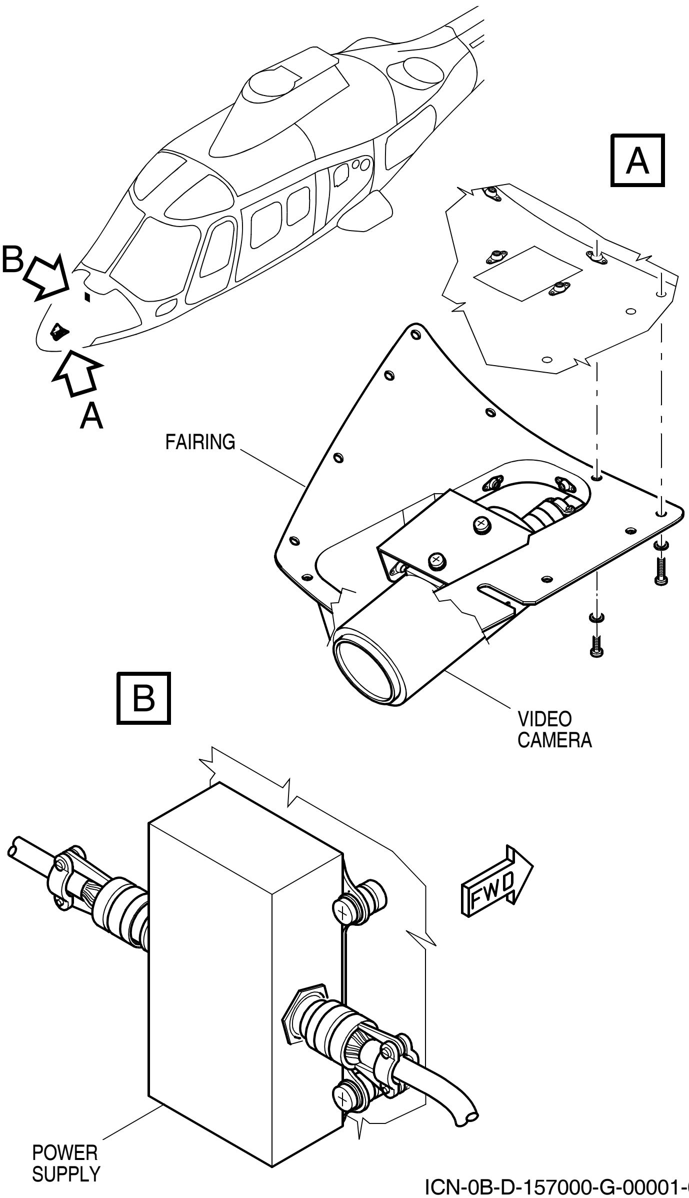

Figure 7-156 External Video Camera System - Components Location

EXTERNAL VIDEO CAMERA (EVS-1500) SYSTEM

The external video camera system is a MAX VIZ EVS-1500 Enhanced Vision System (EVS). This system gives to the flight crew an enhanced visual awareness of terrain and possible obstacles in the forward field of view.

The EVS can operate at night, in smoke, haze and smog. It can also operate in many conditions of rain, snow and fog. The EVS lets the flight crew see in a better way:

– The ground vehicles and other ground-based equipment/ obstacles
– The aircraft on taxi-ways and runways
– The other traffic during takeoff, approach and landing
– The runway and terrain features during climb, descent and low altitude maneuvering.

The system includes a video camera, a power supply and a control panel (Figure 7-156).

The video camera is installed below the helicopter nose with a protection fairing. The video camera is a long-wave infrared (8 to 14 μm) camera assembly. Its normal field of view is 53° horizontal by 40° vertical. The field of view can change to 30° horizontal by 22,5° vertical when the ZOOM switch on the control panel is set to N. The video camera uses heaters to prevent the ice formation on the lens. The video camera gives a RS-170 output video signal that is got through advanced Automatic Gain Control (AGC) algorithms. The video camera sends the RS-170 signal to the power supply.

The power supply, installed in the cockpit, gives the electrical power and all electrical interfaces to the video camera. The power supply sends the RS-170 video signal to the cockpit monitor through a coaxial cable. The cockpit monitor shows thermal scenes of the area in front of the helicopter.

The system is supplied with electrical power from the 28 V dc Number 2 main bus through the EVS circuit breaker.

OPERATION OF THE EXTERNAL VIDEO CAMERA SYSTEM

The external video camera system operates when the PWR and VID SEL buttons on the bezel of the cockpit monitor are pushed. After the

power-up, approximately 20 s are necessary to have a usable image on the cockpit monitor. During this time a video test pattern is shown for approximately 5 s followed by the message OK. After approximately 20 s the monitor becomes black momentarily for a recalibration known as NON-Uniformity Correction (NUC) and then a usable image comes in the view.

To see the IR image in the better way the system does an automatic recalibration at intervals of 4 min. The NUC can be seen on the cockpit monitor as a momentary image interruption for approximately 1 s, while the CAL message comes in the view.

EXTERNAL VIDEO CAMERA SYSTEM CONTROLS AND DISPLAYS

Refer to Figure 7-157.
For the controls on the cockpit monitor, refer to Chapter 31.

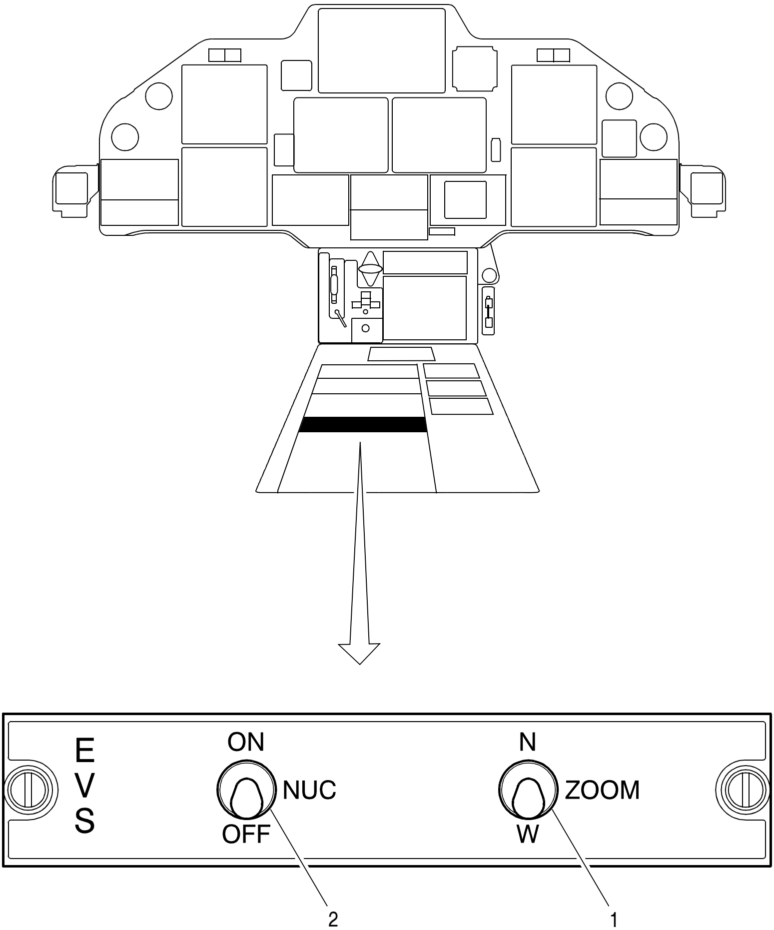

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Figure 7-157 External Video Camera System - Controls and Displays

Key to Figure 7-157

Ref	Control/Display	Function
1	ZOOM switch	N - Sets the camera field-of-view to 30° horizontal by 22,5° vertical. W - Sets the camera field-of-view to 53° horizontal by 40° vertical.
2	NUC switch	ON - The recalibration process of the IR image, known as Non-Uniformity Correction (NUC), is done automatically at intervals of 4 min. OFF - When the NUC switch is moved to OFF, the NUC is done one last time, then the NUC is prevented to operate. The NUC switch must be moved to OFF when the image loss for 1 s is not permitted, but a decreased quality of image is permitted.

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SECTION 8

HANDLING, SERVICING AND MAINTENANCE

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SECTION 8

HANDLING, SERVICING AND MAINTENANCE

TOWING

The helicopter can be manoeuvred on the ground manually, or by a suitable vehicle, using the towing bar which can be secured to the nose wheel bar. The wheel can be steered 40° each side from the center. A red mark on the left and right sides of the nose fairing indicate the maximum towing angle.

To tow the helicopter proceed as follows:

1. Clear the departure area of support equipment such as work stands, power units, etc.
2. Disengage the nose gear centering lock by setting the NOSE WHEEL LOCK lever on the right side of the front console to ‘OFF’.
3. Connect the tow bar to the holes of the nose wheel fork.
4. If applicable, remove the wheel chocks.
5. Ensure that the rotor brake is applied.
6. Unlock the main landing gear wheels by releasing the PARK AND EMER BRAKE lever on the front console.
7. Connect the tow bar to the towing vehicle and tow the helicopter slowly. Maximum towing speed is limited to 10 km/h, avoid sudden starts and stops during the towing process.

TOWING LIMITATIONS

Towing must be carried out using towing bar P/N 109-3900-02-123.

Maximum steering angle .... 40 degrees

Maximum towing speed .... 10 km/h

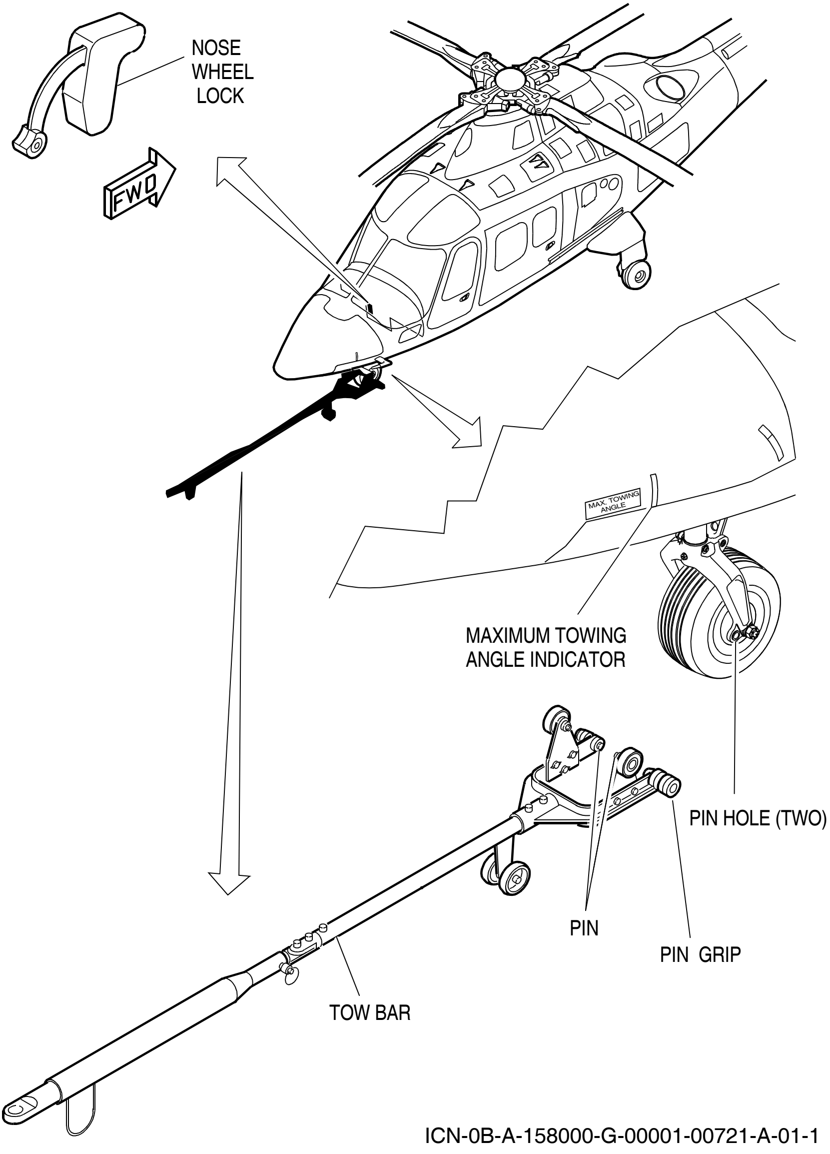

Figure 8-1 Towing

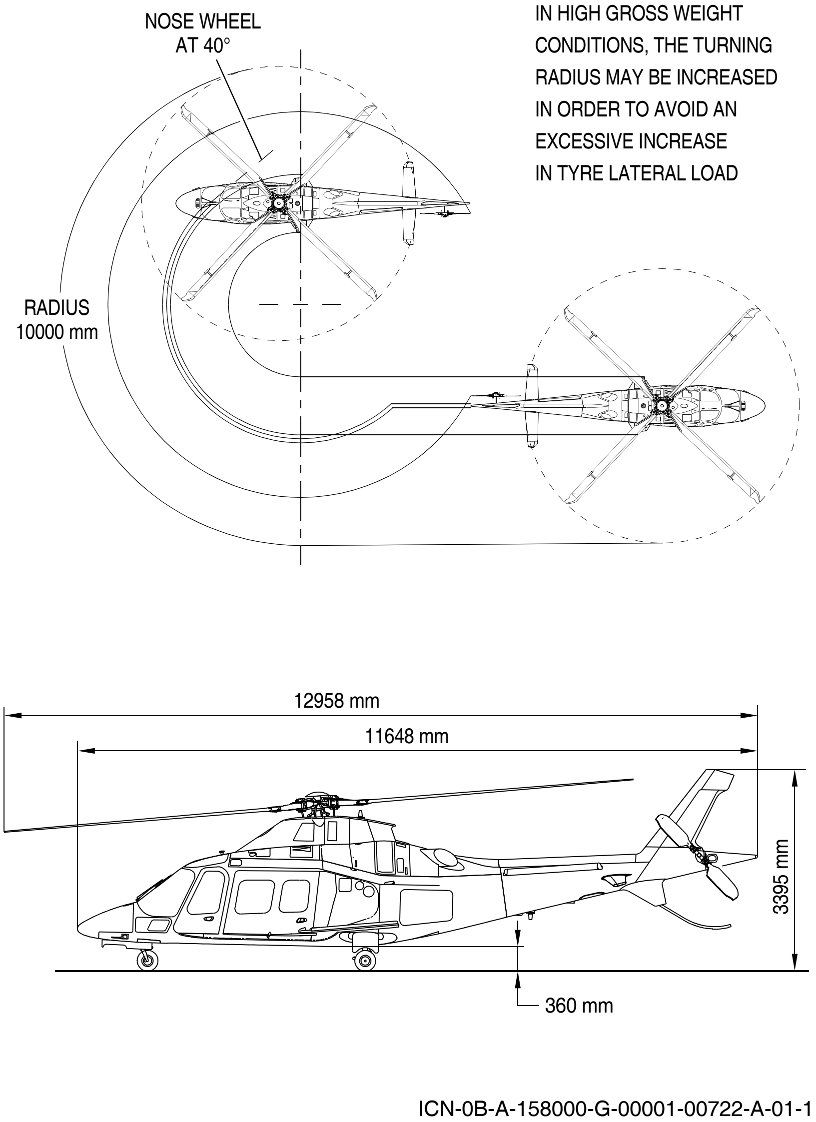

Figure 8-2 Turning Radius and Ground Clearance

TAXIING

Taxiing must only be performed by qualified pilots. Assistance of ground personnel may be necessary when taxiing in obstructed areas to aid the pilot by means of visual clearance hand signals.

PARKING

Refer to Figure 8-3.

CAUTION

Wait at least 5 minutes after pitot heat has been switched off before installing pitot-static tube covers. Wait at least 30 minutes after engine shut-down before installing engine exhaust duct covers.
Refer the Maintenance Manual for additional information.

Park the helicopter in desired parking area on a level surface, when possible. Attach static ground wire and check that all switches are in OFF or neutral position. During adverse weather conditions install the approved straps and covers, as required.
For extended parking, disconnect the battery, engage rotor brake, lock main landing gear wheels by operating PARK AND EMERG BRAKE handle located on front console. Centre and lock nose wheel by operating the NOSE WHEEL LOCK lever located on the front console. Close all access doors and panels and install protective covers as shown in Figure 8-4.

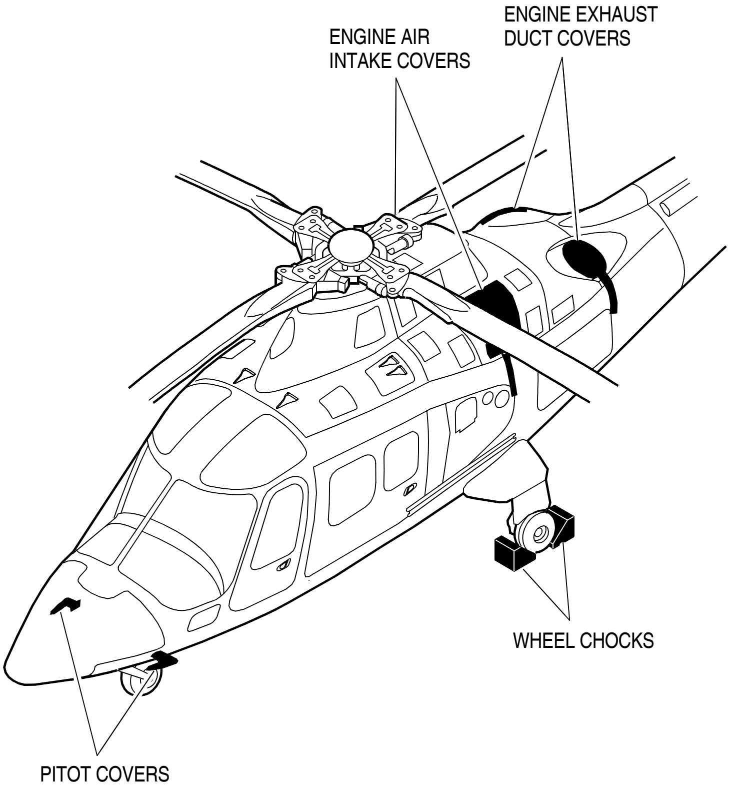

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Figure 8-3 Parking the Helicopter

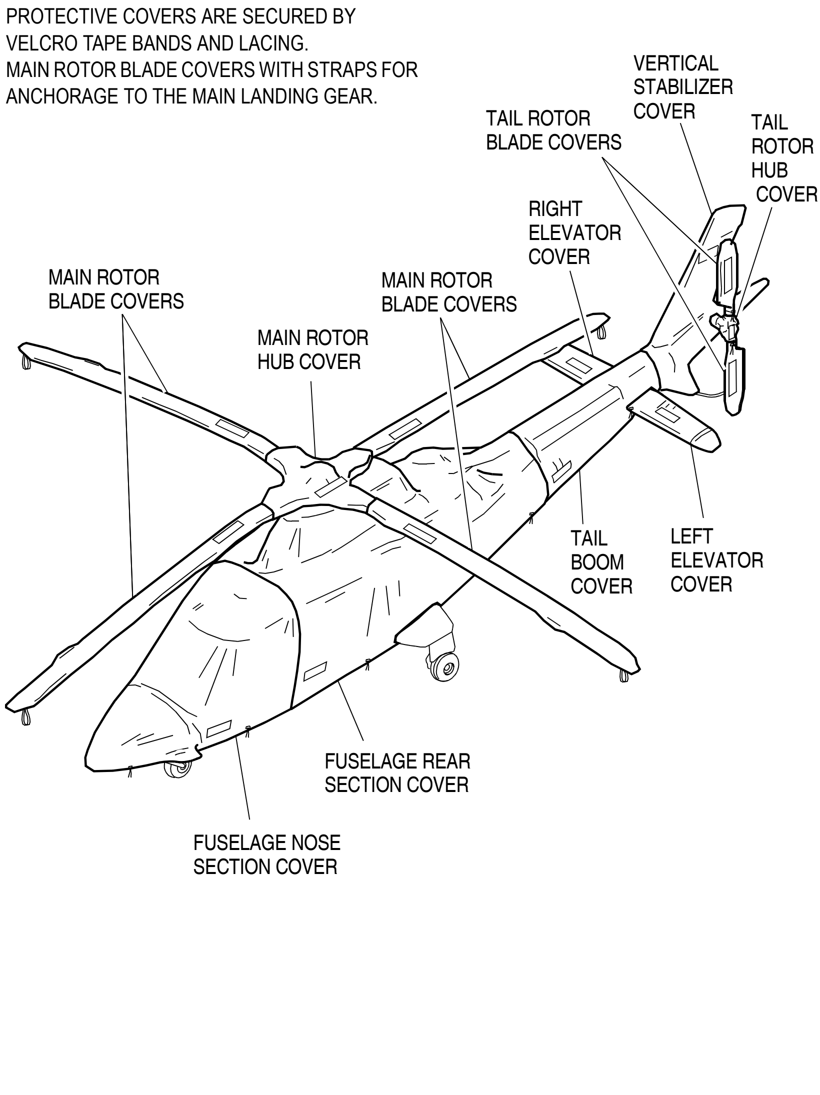

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Figure 8-4 All Weather Protective Covers

MOORING

The helicopter can be moored on a paved ramp, if available, with suitably spaced tie-down rings, and headed in direction from which forecast wind is expected.

CAUTION

Use this procedure only when the possible wind velocity is 40 thru 60 Kts.
If forecast wind velocity exceeds 60 Kts, put helicopter in a sheltered area or in a hangar.

If suitable paved ramp and tie-down rings are not available, park helicopter on an appropriate parking area, headed into wind and use appropriate mooring anchors or make “dead man” anchors.
Engage rotor brake, lock main landing gear wheels, center and lock nose wheel, close all access doors and panels. Remove from parking area all loose equipment that can be lifted by wind.

To moor the helicopter proceed as follows (Figure 8-5):

a) Fill fuel tanks to capacity.
b) Using screws and washer, install the mooring plate on both sides of the fuselage.
c) If a paved ramp with suitably spaced tie-down rings is available, park the helicopter headed in the direction from which the highest forecast winds are expected.
d) Engage rotor brake by operation ROTOR BRAKE lever located on overhead console.
e) Lock main landing gear wheels by operating PARK AND EMERG BRAKE handle located on front console.
f) Centre and lock nose wheel by operating NOSE WHEEL LOCK lever located on the front console.
g) Ensure that all electrical switches are ‘OFF’.
h) Close all access doors and access panels.
i) Place protective covers on pitot tubes both engine air intakes and engine exhaust ducts.

j) Install chocks fore and aft of each main landing gear wheel.

k) Using the main rotor blade straps, anchor the main rotor blades to the main landing gear.

l) Using appropriate straps anchor the tail rotor blades to the vertical fins.

m) Secure the tow bar pin holes, mooring plates and main landing gear legs to tie-down rings with Manila ropes of suitable length.

n) Remove all loose equipment that can be lifted by wind from the parking area. Secure all support equipment standing in the parking area.

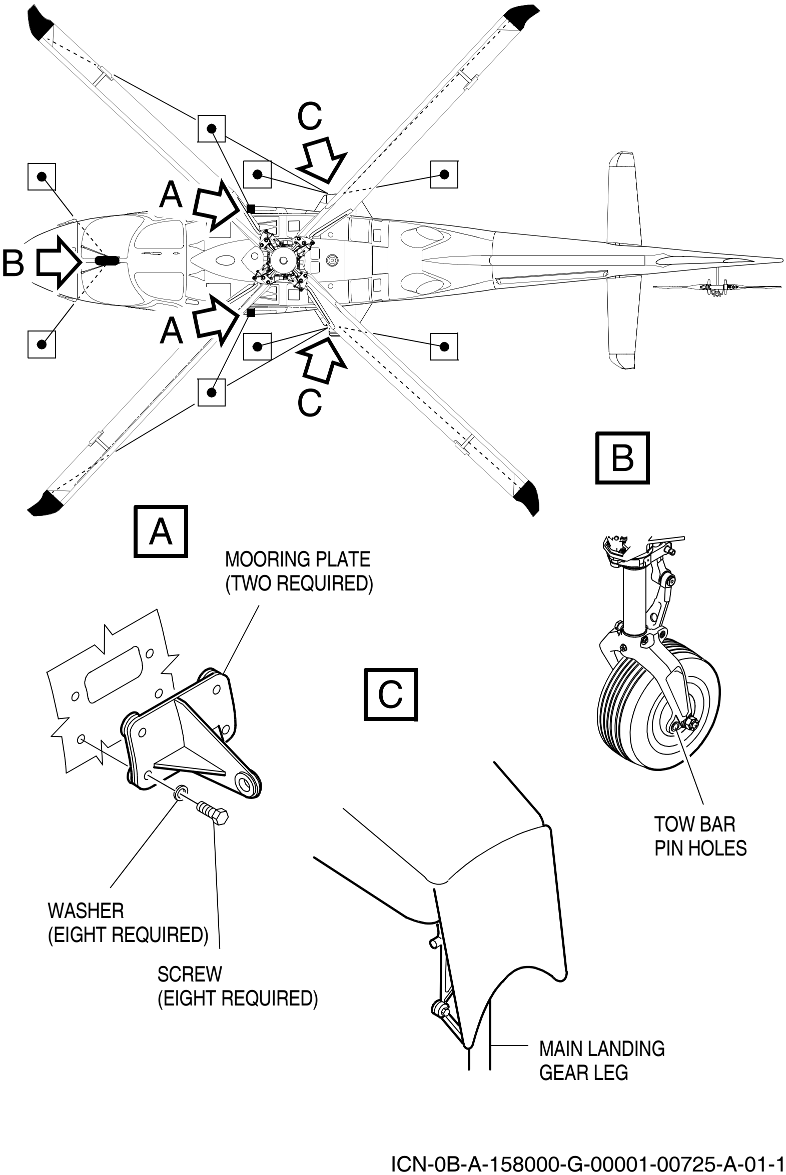

Figure 8-5 Mooring

SERVICING

Refer to Figure 8-6 for the servicing points on the helicopter.

REF.	ITEM	CAPACITY (l)	NOTE
1	Main transmission	11	
2	Fuel system: Main tanks	 575	Usable: 562 liters
3	90° gearbox	0.24	
4	Engine oil system	5.25	Each system
5	Landing gear strut		Nitrogen. Main LG: 210.3 psi Nose LG: 120.4 psi
6	Landing gear wheel tyres		Nitrogen: 85 psi
7	Accumulators (utility hydraulic system)		Nitrogen Charge to 427 psi
8	Flight controls accumulators		Nitrogen
9	Battery		
10	Hydraulic reservoir	1.350	N° 1 system
10	Hydraulic reservoir	3.032	N° 2 system
11	Dampers	0.028	Each damper [1]

[1] On the fluid level indicator (refer to detail A of Figure 8-6), make sure that the level of the fluid is between the green FULL line and the red REFILL line. If the level indicator position aligns with or on the red REFILL line, fill the damper with the applicable hydraulic fluid. Refer to Maintenance Manual.

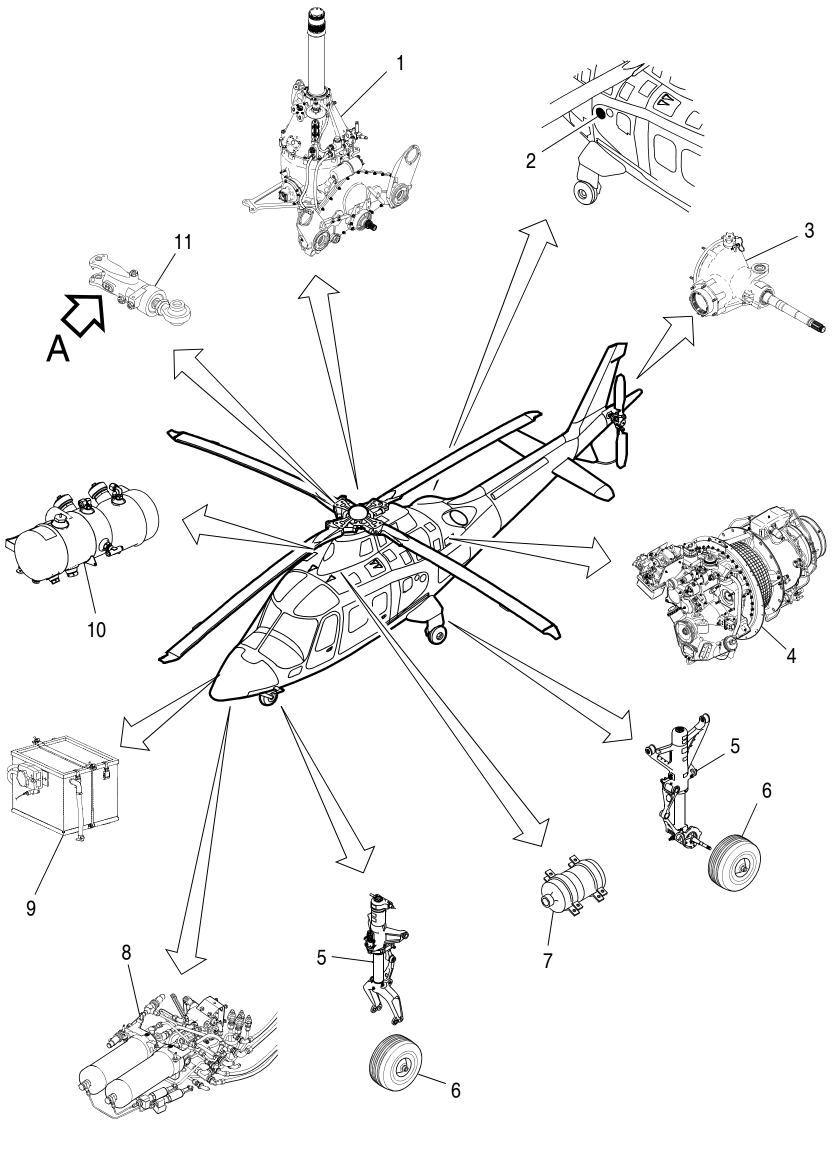

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Figure 8-6 Servicing (Sheet 1 of 2)

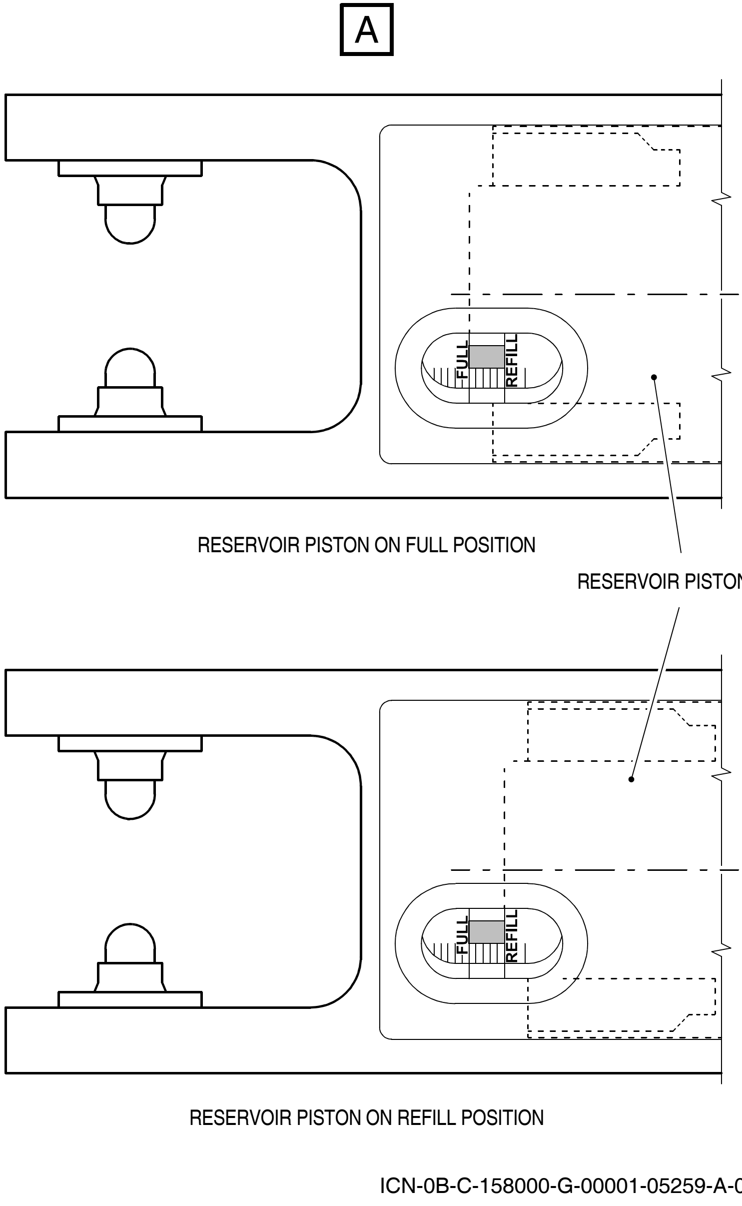

Figure 8-6 Servicing (Sheet 2 of 2)

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SECTION 9

SUPPLEMENTAL PERFORMANCE INFORMATION

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SECTION 9

SUPPLEMENTAL PERFORMANCE INFORMATION

GENERAL

The Supplemental Performance Information contained in this section is provided for use in conjunction with Section 4 and the Optional Equipment Supplements, as applicable.

This section contains useful cruise charts to determine:

- *The Maximum Endurance Cruise*: the airspeed required to achieve the maximum flight time.
- *The Maximum Range Cruise*: the airspeed required to achieve the maximum range.
- *The Recommended Cruise*: a reasonable increase in airspeed for a 1% increase in specific fuel consumption compared to the Maximum Range Cruise.

It also contains:

- Hover Ceiling OEI charts with headwind effect
- Torque required to hover IGE and OGE.

HELICOPTER CONFIGURATION

Clean configuration.

CRUISE CHARTS

The cruise charts are based on estimates and flight test data.

These data do not include the effect of bleed air on fuel consumption. The fuel flow data is applicable to the basic helicopter without the addition of any optional equipment which would appreciably affect lift, drag, or power available.

EXAMPLE FOR INTERPOLATION AMONG CHARTS

Wanted Indicated airspeed, torque, fuel consumption for True airspeed.

Known Pressure altitude: 5000 ft, OAT: 25 °C,
Gross weight: 2600 kg.

Method Select the suitable charts (Figures 9-16, 9-17, 9-22, 9-23) on each of the 4 charts intersect the gross weight 2600 kg line with the 135 kt TAS (maximum cruise range) to read, IAS, TRQ and fuel consumption. See the following table.

DESCRIPTION	DATA (GW=2600 kg)				INTERPOLATION		
	Fig. 9-16	Fig. 9-17	Fig. 9-22	Fig. 9-23			
Pressure Altitude (ft)	4000	4000	6000	6000	4000	6000	5000
OAT (°C)	20	40	20	40	25	25	25
TRQ (%)	60	60	60	60	60	60	60
IAS (kt)	128	124	122	119	127	121	124
TAS (kt)	135	135	135	135	135	135	135
Fuel Consumption (kg/h)	185	188	179	183	186	180	183

Note

To obtain data for a pressure altitude of 5000 ft and an OAT of 25 °C, interpolate linearly for OAT (constant pressure altitude) and obtain data for OAT 25 °C at pressure altitude of 4000 ft and of 6000 ft. Then, interpolate linearly for pressure altitude and obtain the final results at the pressure altitude of 5000 ft and OAT of 25 °C.

PERFORMANCE CORRECTION AFTER KIT INSTALLATION

Table 9-1 shows the correction to be applied to the Cruise performance when optional equipment are installed. The column Drag Factor give the effect in term of equivalent drag area on the helicopter performance, Level Flight column present the approximate speed-reduction, Fuel consumption the increase of fuel consumption (at same level of power) mean while Range and Endurance give the reduction effect on range/endurance due to the combination of fuel consumption increase and/or the speed reduction.

Drag reduction can be used to obtain a more precise evaluation on performance: entering with the desired speed, cross the dotted line on the left of chart the proceed up to read the value of DTQ on the secondary scale in the top left box.

Multiplying the value read for the double of Drag factor obtained for the optional equipment it is obtained the increase of TQ required to flight at the specified speed.

On the same chart entering with the total TQ (torque required to flightat specified speed without kit+DTQ obtained before) it is possible to read on the up scale the fuel consumption related.

Effect of optional equipments on level flight performance	Supplement	Drag factor	Level flight			Fuel cons	Range	End.
		F	V_H	V_{MR}	V_{RC}			
Equipment		[m^2]	kTAS	kTAS	kTAS	[%]	[%]	[%]
ECS PN 109-0813-50	1	n.i.	-10	n.i.	n.i.	+5%	-5%	-5%
Heater PN 109-B810-49	2	n.i.	-10	n.i.	n.i.	+5%	-5%	-5%
External hoist PN 109-B810-16	5	0.3	-9	-8	-8	n.i.	-7%	-2%
Cargo hook PN 109-0810-31	6	0.1	-3	-2	-2	n.i.	-2%	-1%
Snow Skis (including FLG effect) PN 109-0811-99	8	0.55	-17	-17	-17	n.i.	-10%	-3%
Slump protection pads (including FLG effect) PN 109-0811-73	9	0.55	-17	-17	-17	n.i.	-10%	-3%
Fixed wheeled L/G PN 109-B811-02	26	0.45	-15	-15	-15	n.i.	-10%	-2%

V_H = Maximum horizontal speed;
V_{MR} = Maximum range speed;
V_{RC} = Recommended cruise speed
n.i.: no impact on performance

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Table 9-1 Correction table after kit installation

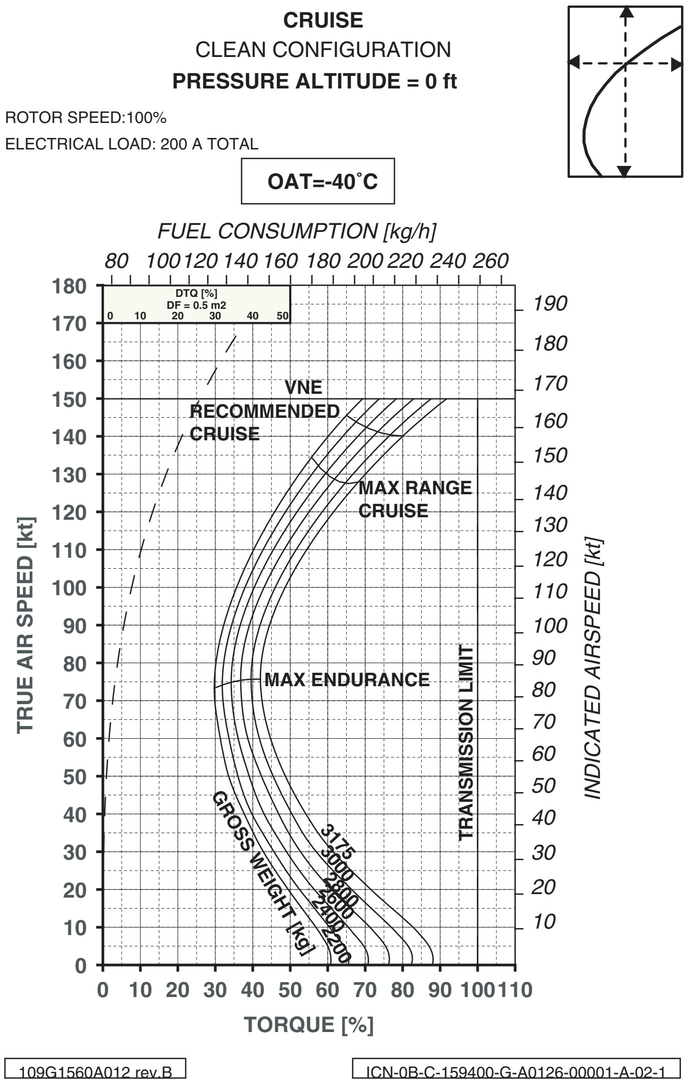

Figure 9-1 Cruise - Pressure Altitude 0 ft, OAT -40 °C

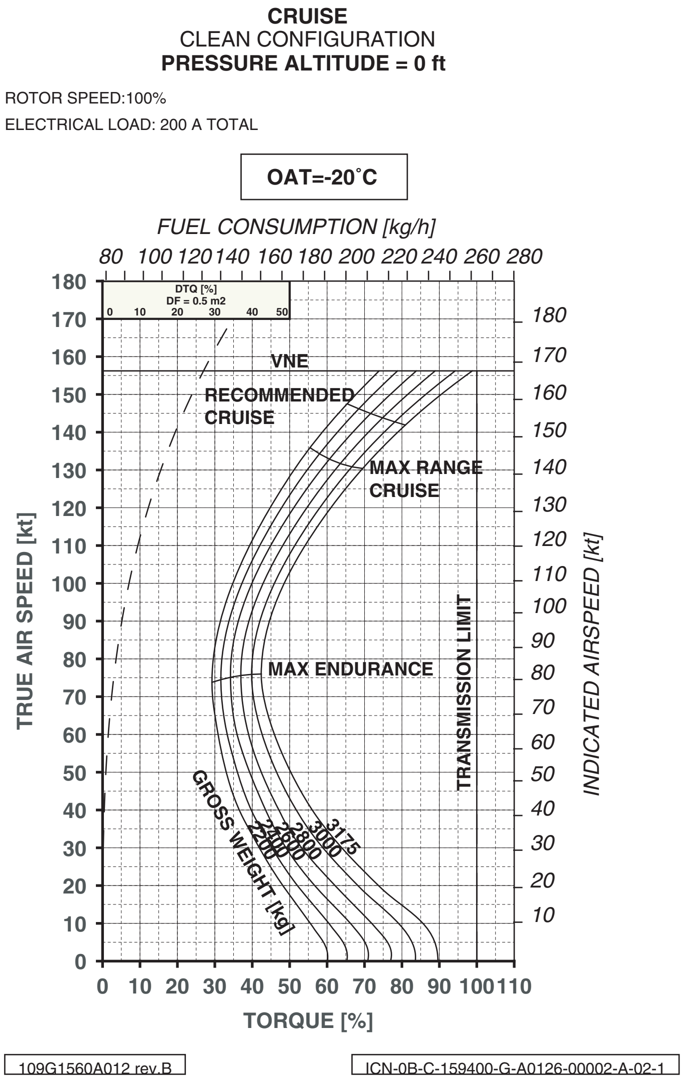

Figure 9-2 Cruise - Pressure Altitude 0 ft, OAT -20 °C

CRUISE
CLEAN CONFIGURATION
PRESSURE ALTITUDE = 0 ft

ROTOR SPEED:100%
ELECTRICAL LOAD: 200 A TOTAL

OAT=0˚C

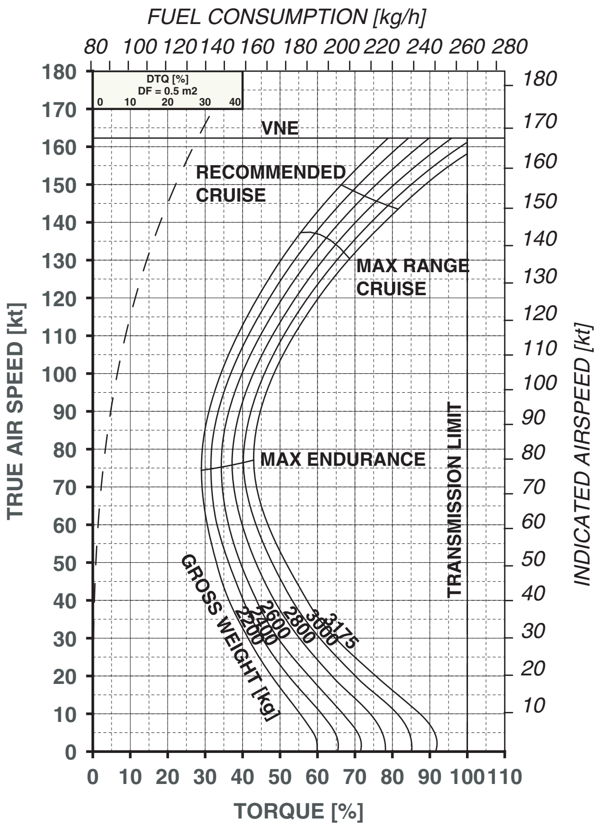

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Figure 9-3 Cruise - Pressure Altitude 0 ft, OAT 0 °C

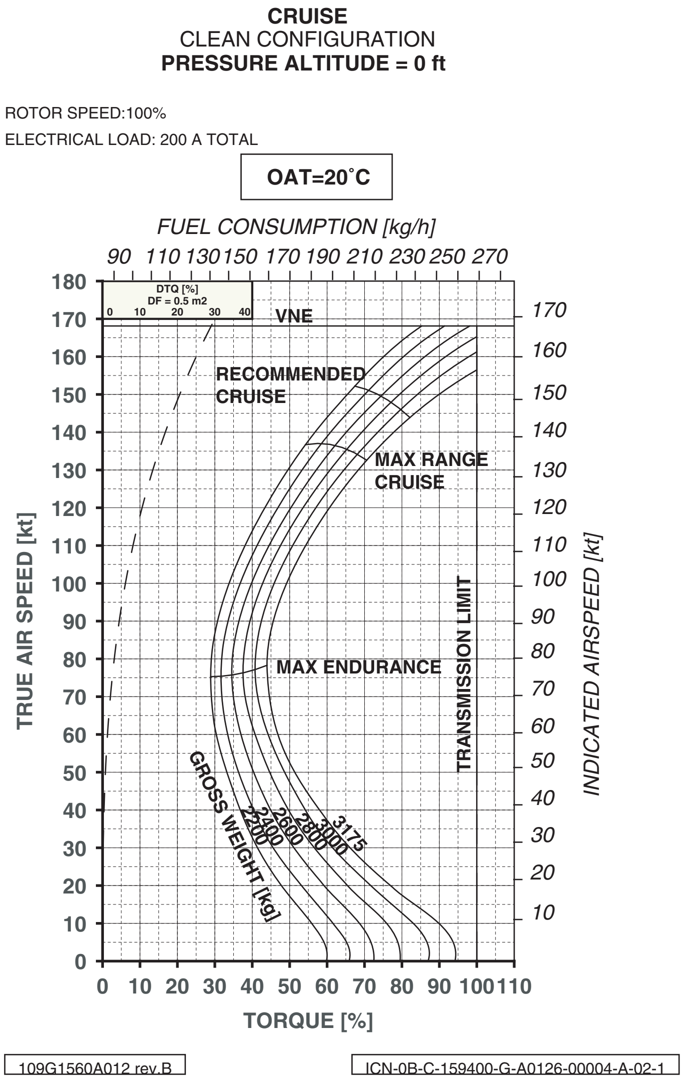

Figure 9-4 Cruise - Pressure Altitude 0 ft, OAT 20 °C

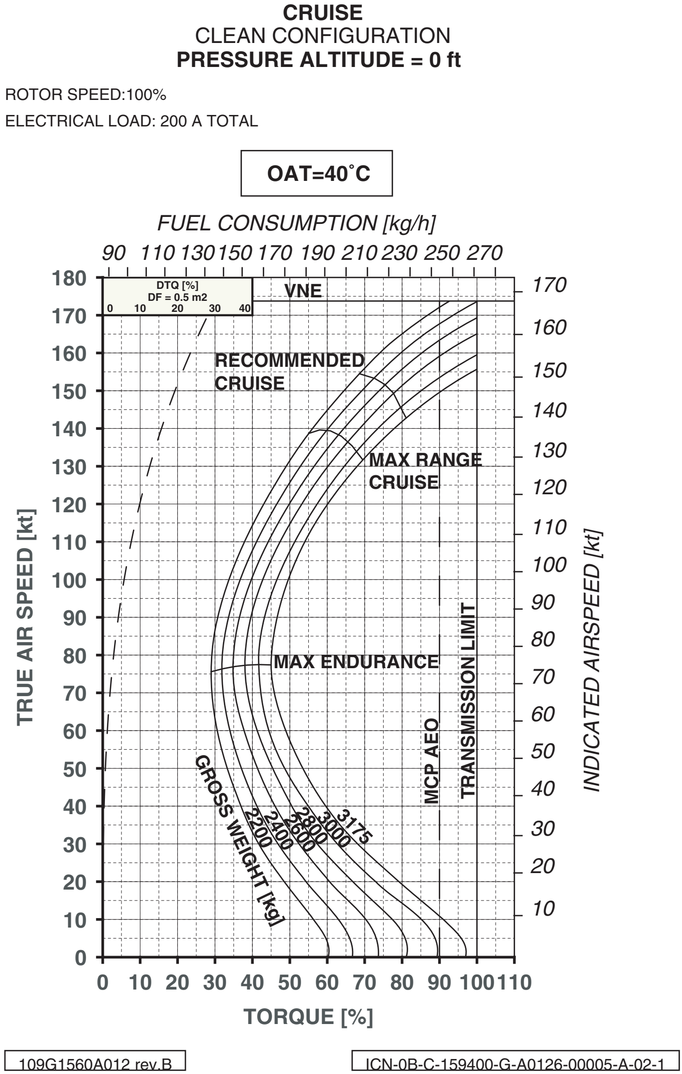

Figure 9-5 Cruise - Pressure Altitude 0 ft, OAT 40 °C

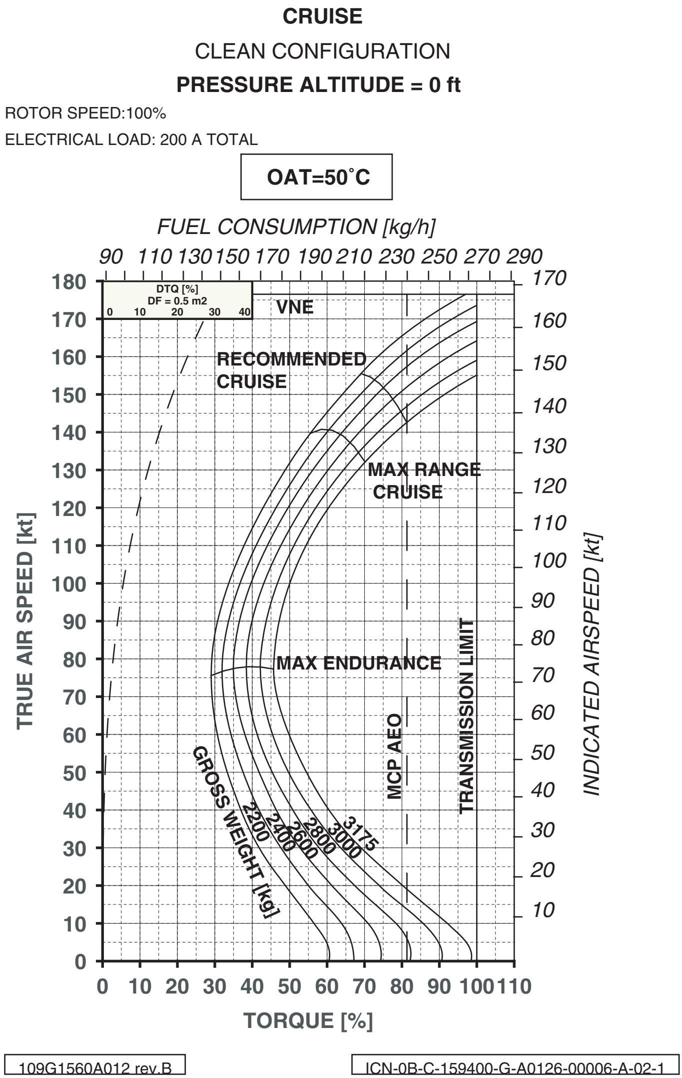

Figure 9-6 Cruise - Pressure Altitude 0 ft, OAT 50 °C

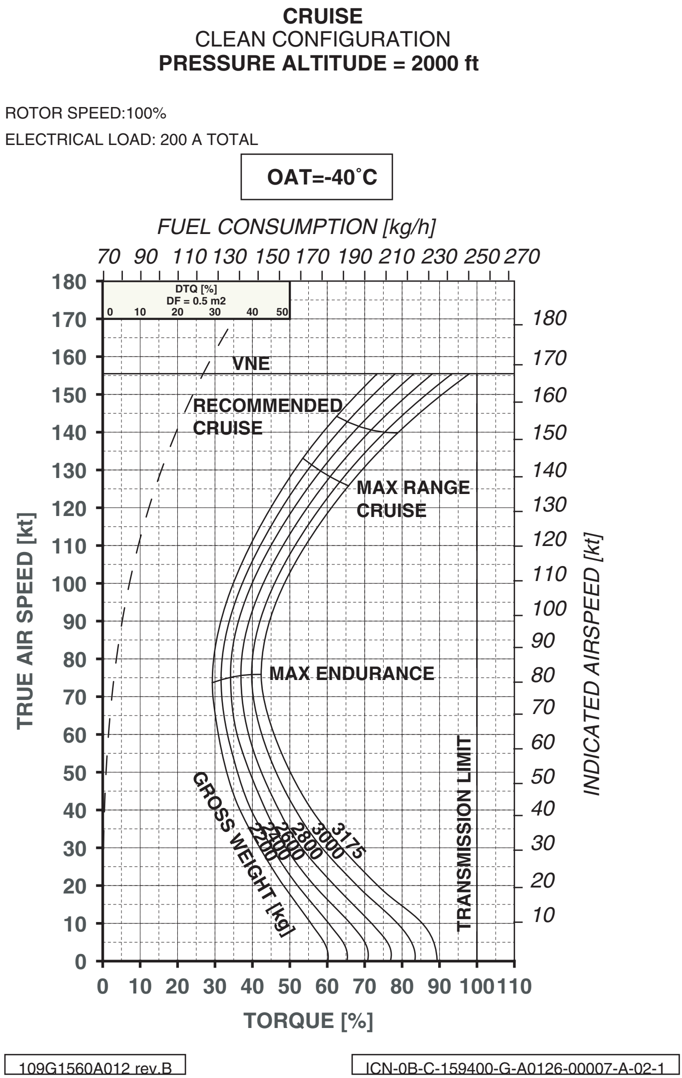

Figure 9-7 Cruise - Pressure Altitude 2000 ft, OAT -40 °C

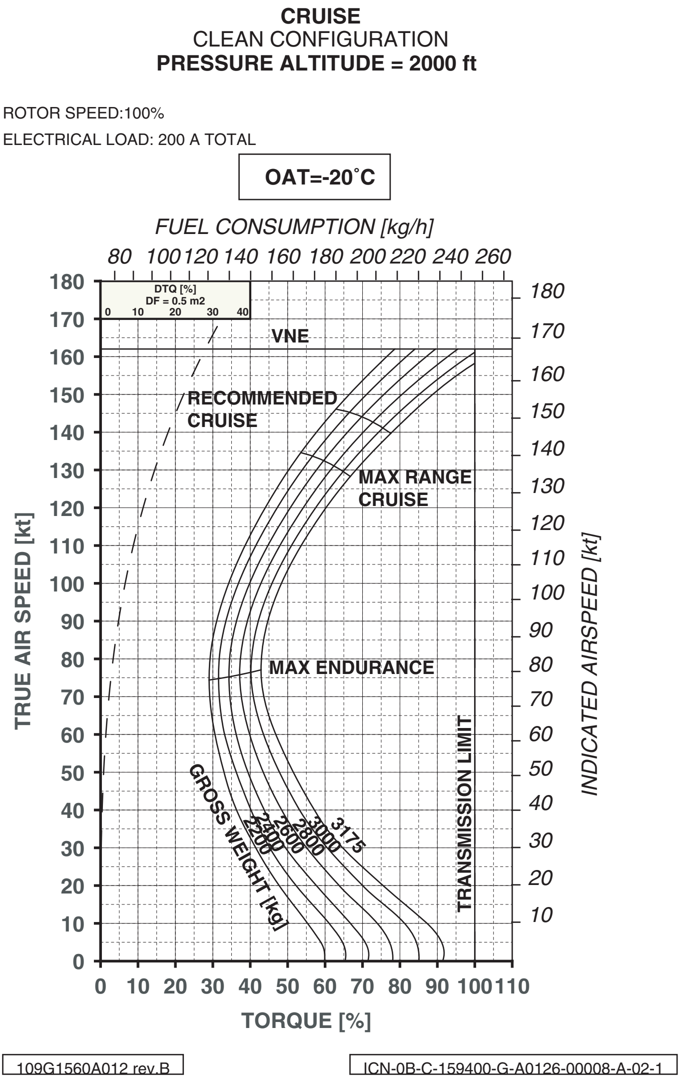

Figure 9-8 Cruise - Pressure Altitude 2000 ft, OAT -20 °C

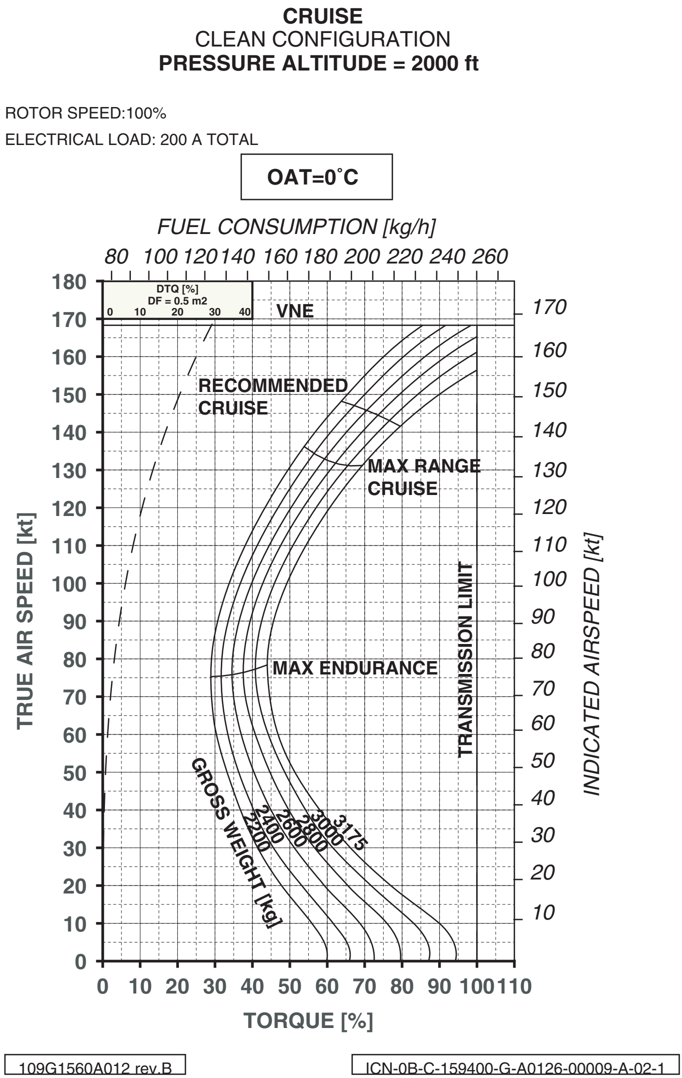

Figure 9-9 Cruise - Pressure Altitude 2000 ft, OAT 0 °C

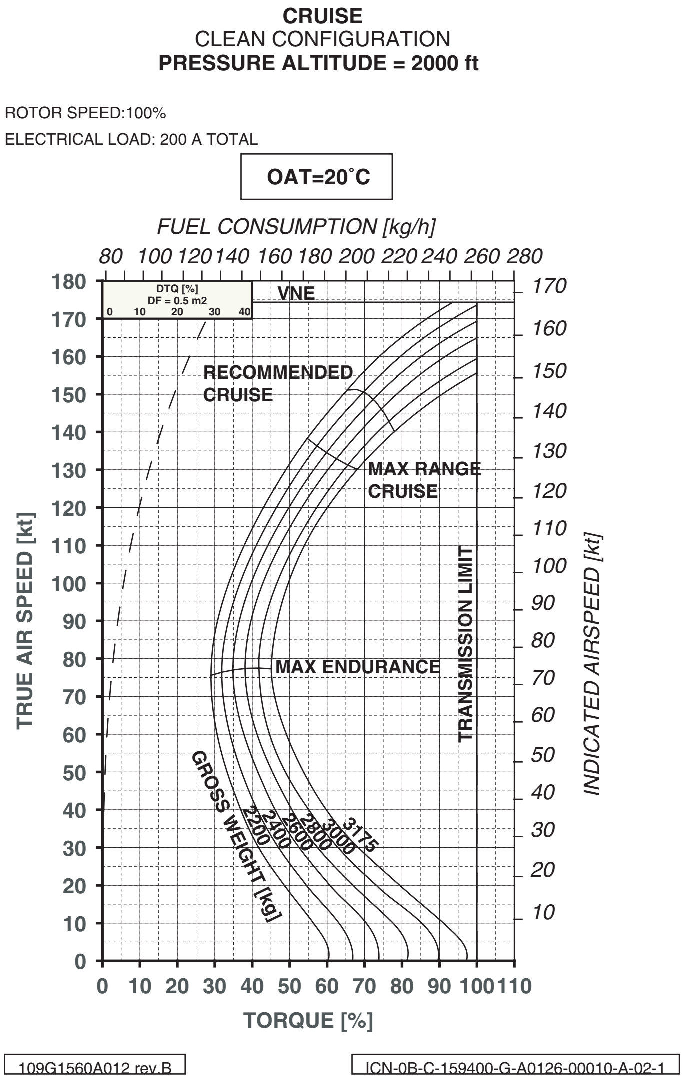

Figure 9-10 Cruise - Pressure Altitude 2000 ft, OAT 20 °C

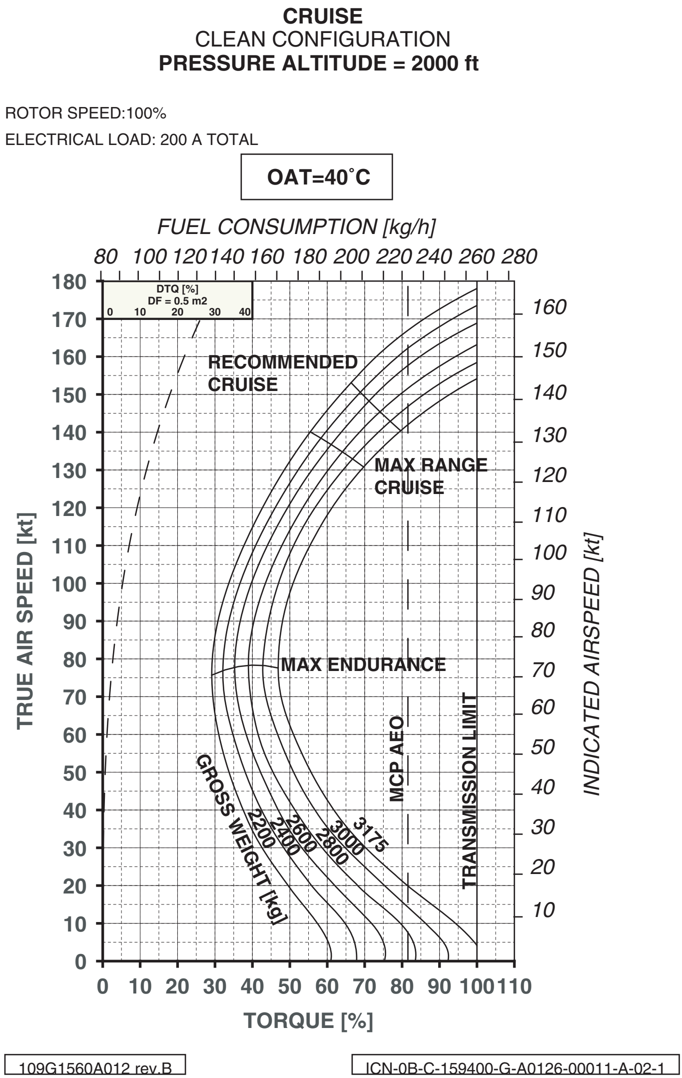

Figure 9-11 Cruise - Pressure Altitude 2000 ft, OAT 40 °C

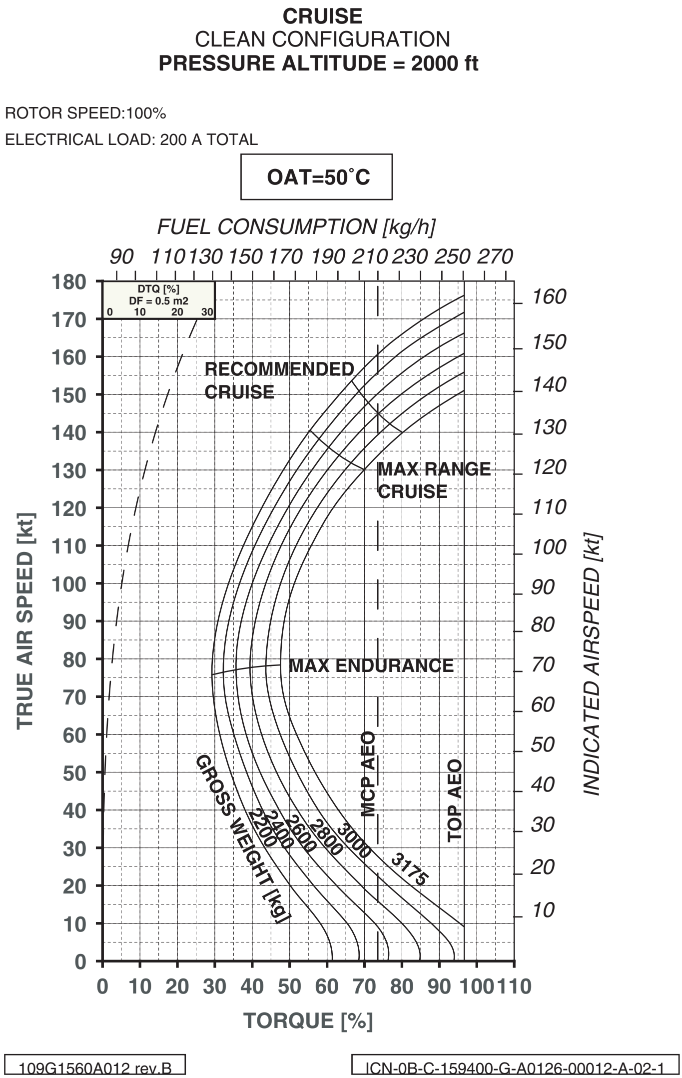

Figure 9-12 Cruise - Pressure Altitude 2000 ft, OAT 50 °C

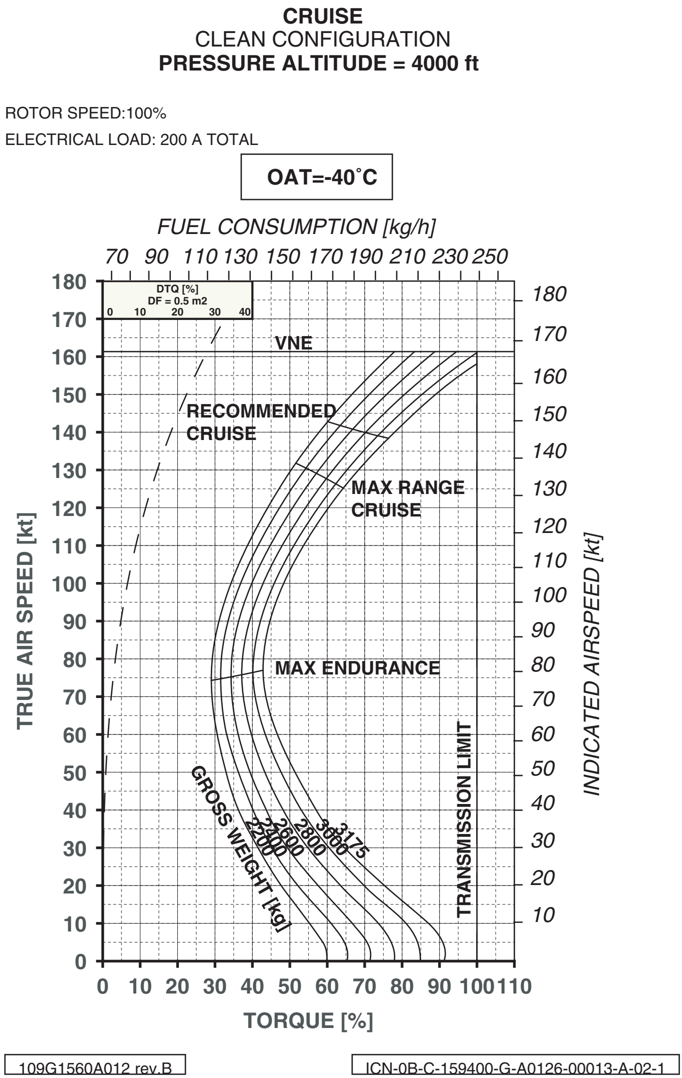

Figure 9-13 Cruise - Pressure Altitude 4000 ft, OAT -40 °C

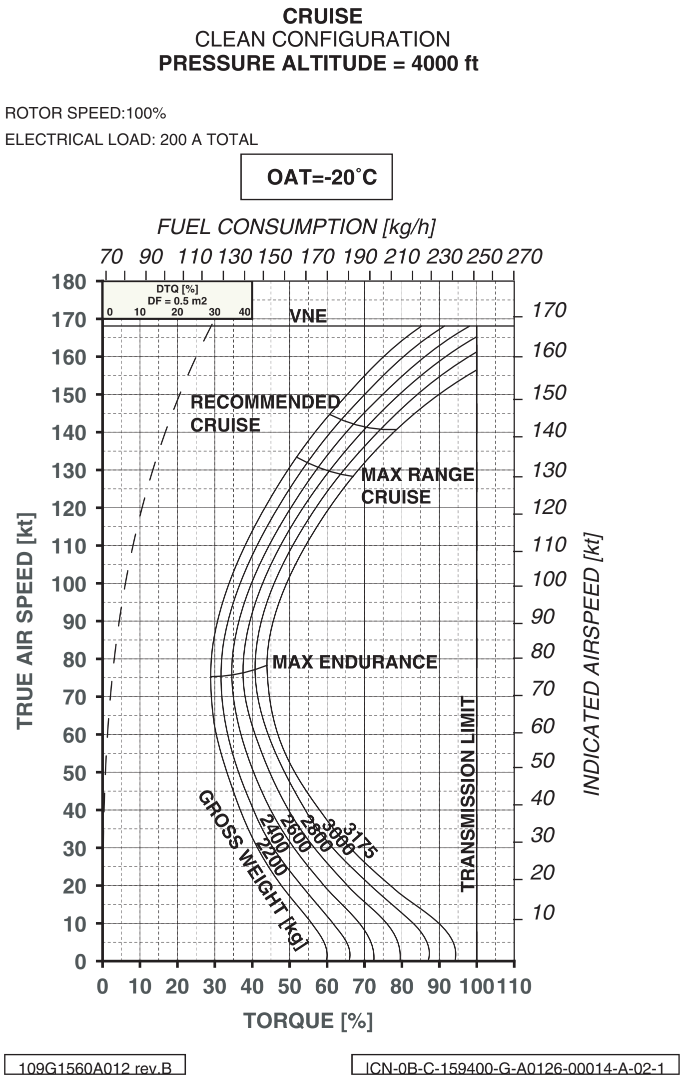

Figure 9-14 Cruise - Pressure Altitude 4000 ft, OAT -20 °C

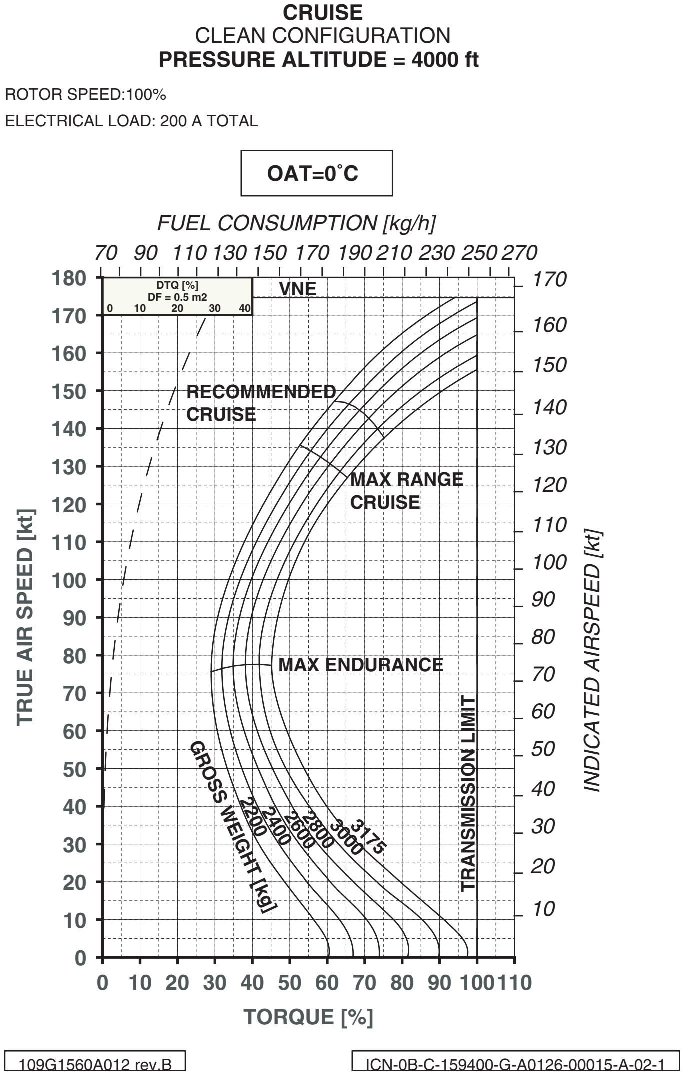

Figure 9-15 Cruise - Pressure Altitude 4000 ft, OAT 0 °C

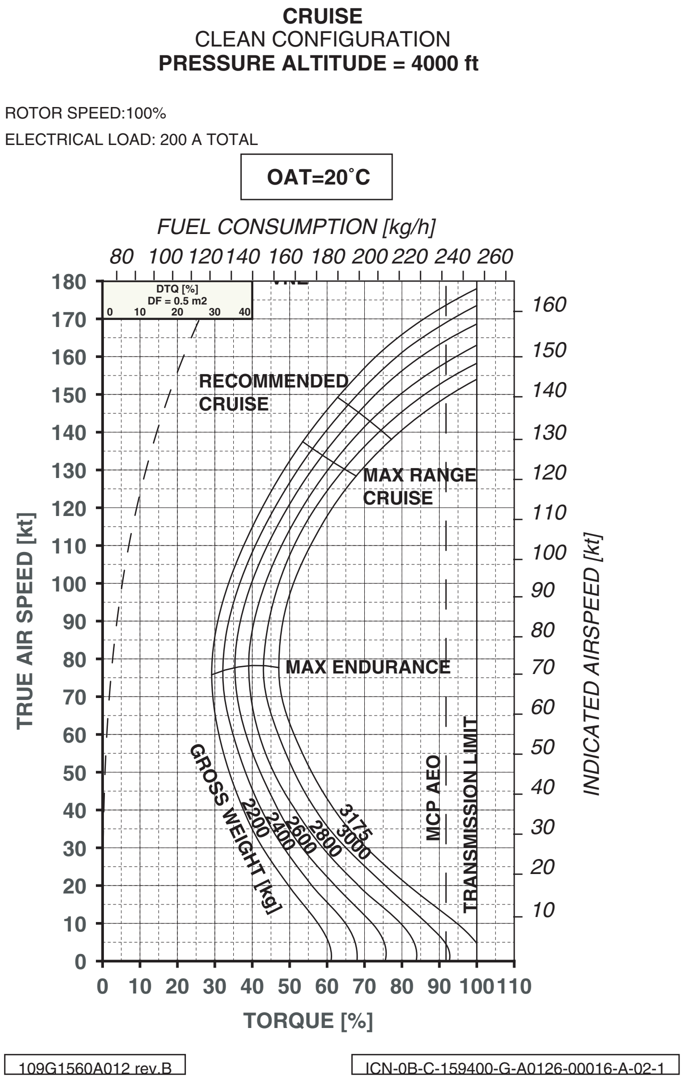

Figure 9-16 Cruise - Pressure Altitude 4000 ft, OAT 20 °C

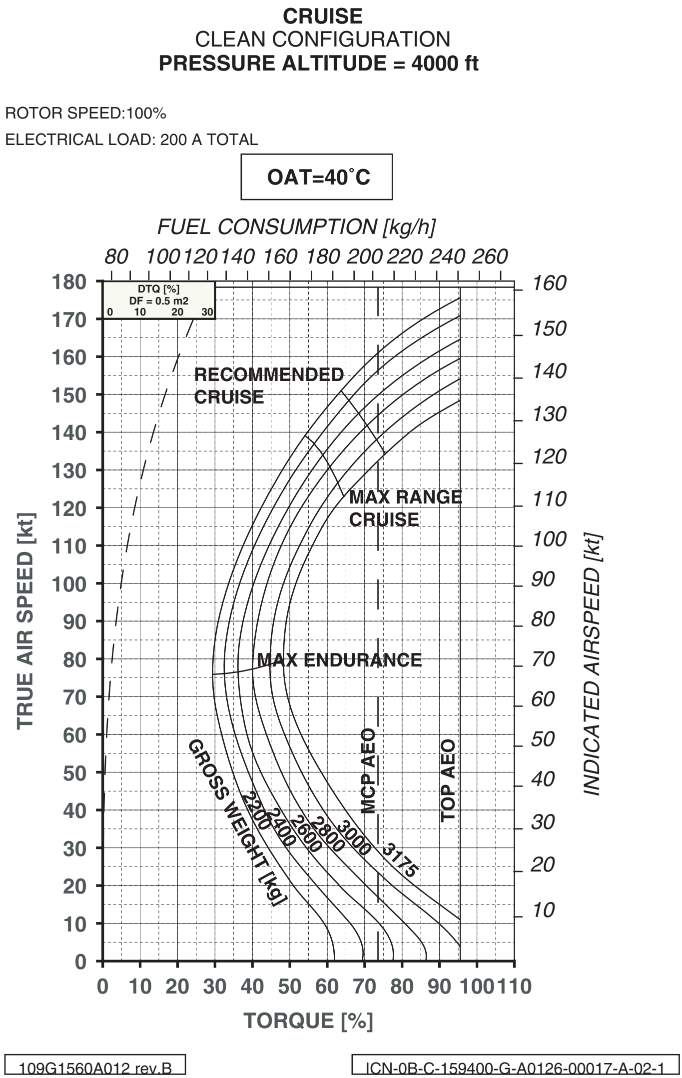

Figure 9-17 Cruise - Pressure Altitude 4000 ft, OAT 40 °C

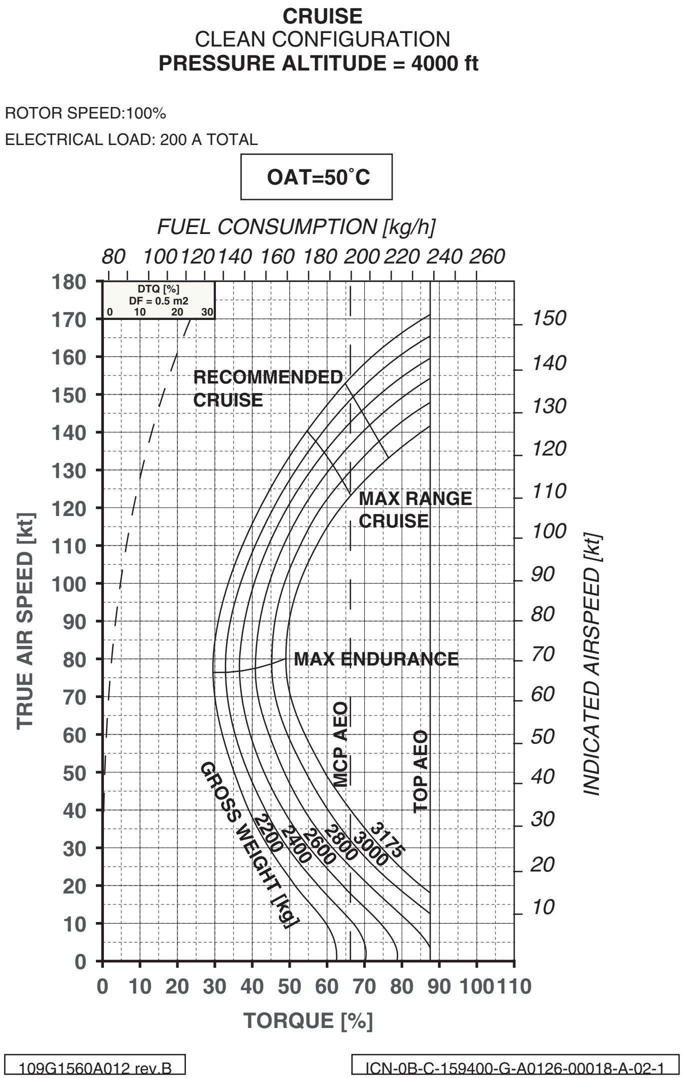

Figure 9-18 Cruise - Pressure Altitude 4000 ft, OAT 50 °C

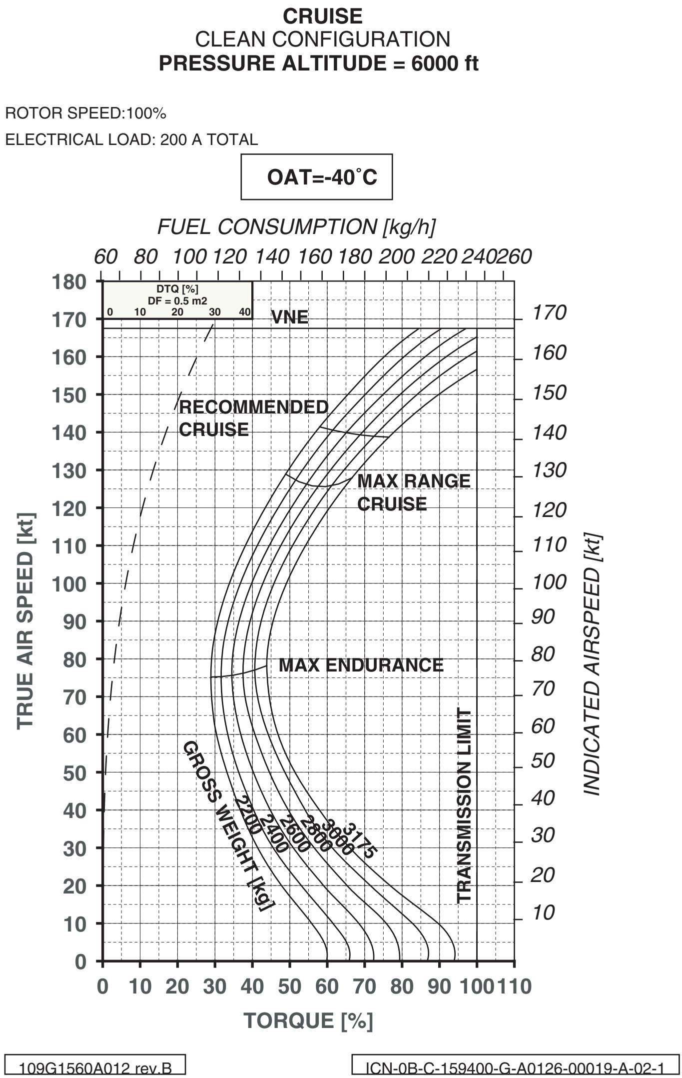

Figure 9-19 Cruise - Pressure Altitude 6000 ft, OAT -40 °C

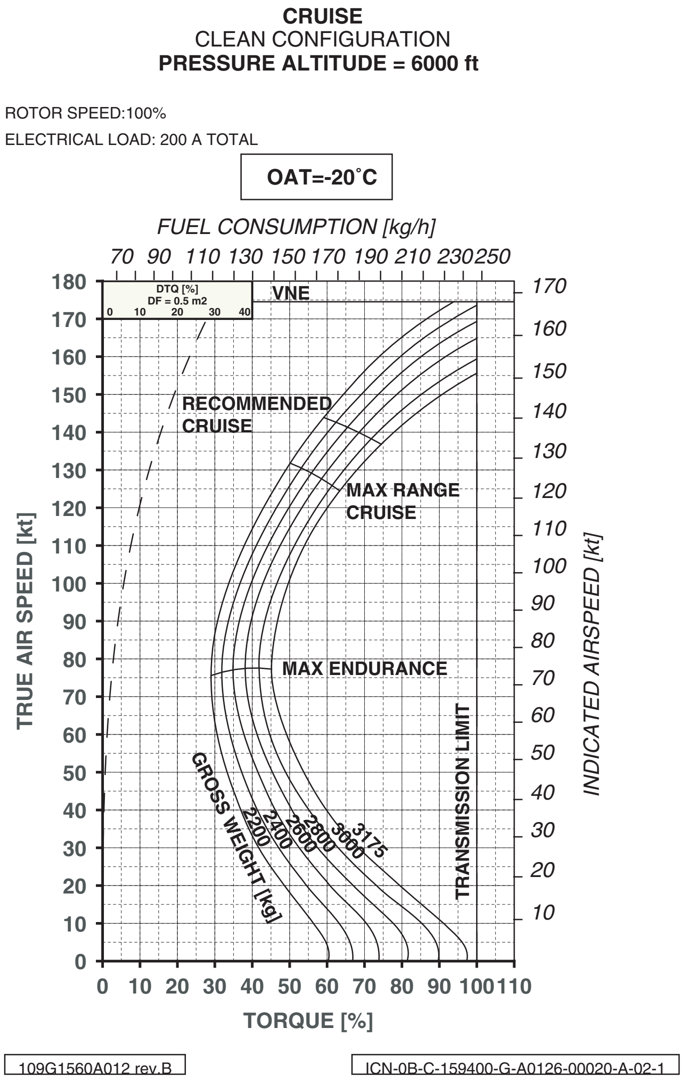

Figure 9-20 Cruise - Pressure Altitude 6000 ft, OAT -20 °C

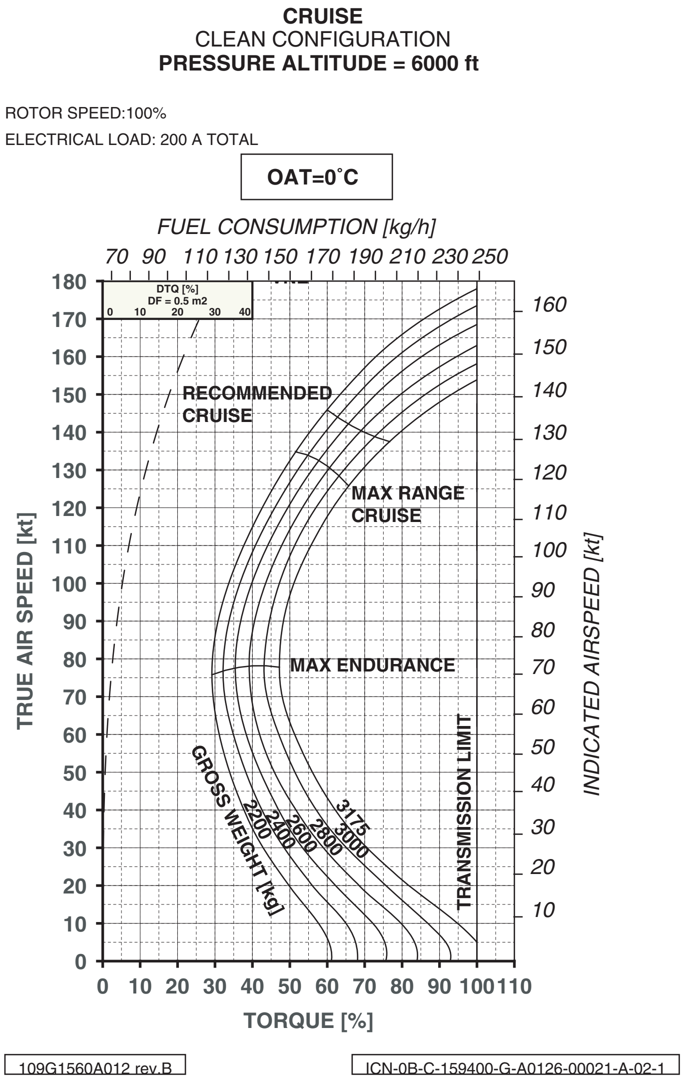

Figure 9-21 Cruise - Pressure Altitude 6000 ft, OAT 0 °C

CRUISE
CLEAN CONFIGURATION
PRESSURE ALTITUDE = 6000 ft

ROTOR SPEED:100%
ELECTRICAL LOAD: 200 A TOTAL

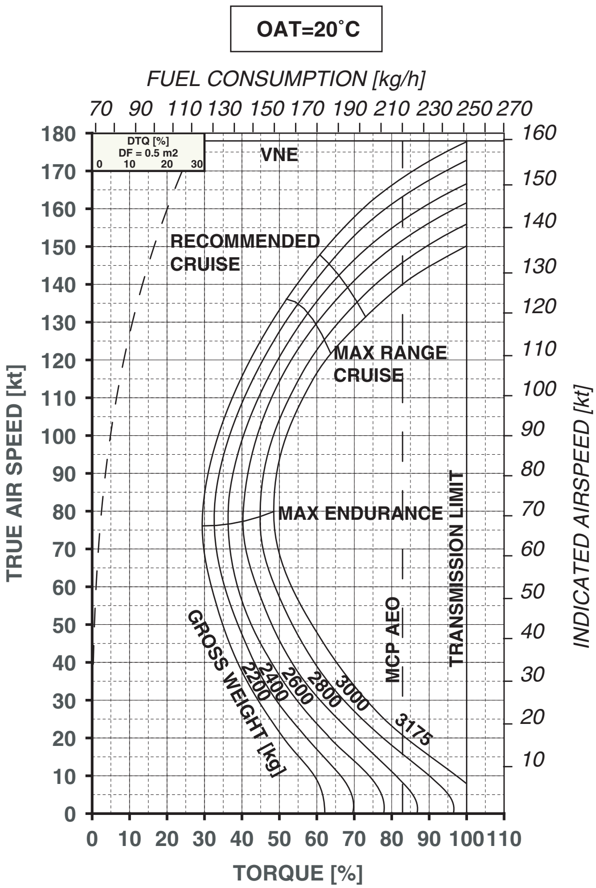

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Figure 9-22 Cruise - Pressure Altitude 6000 ft, OAT 20 °C

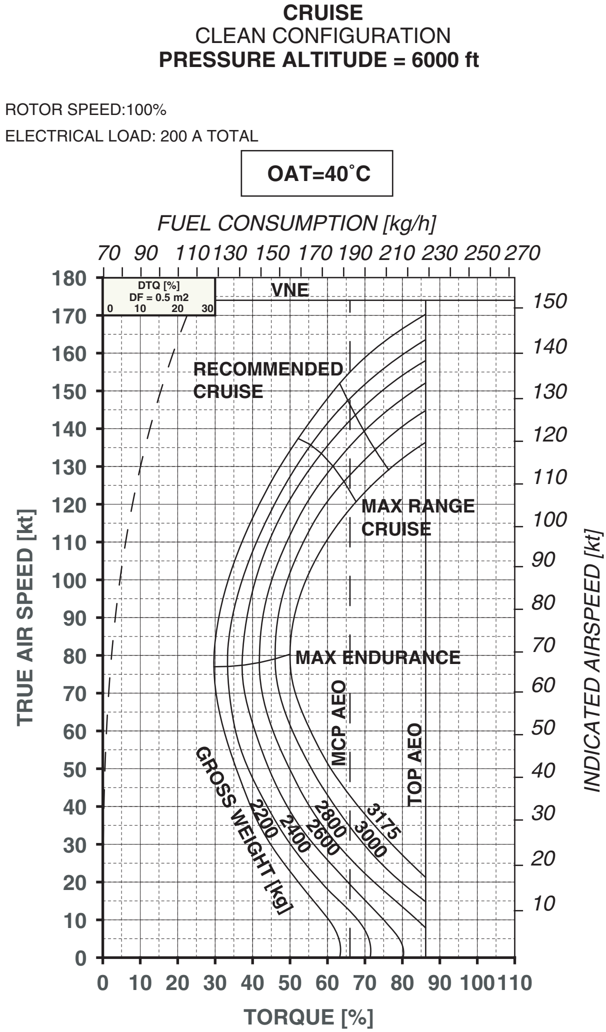

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Figure 9-23 Cruise - Pressure Altitude 6000 ft, OAT 40 °C

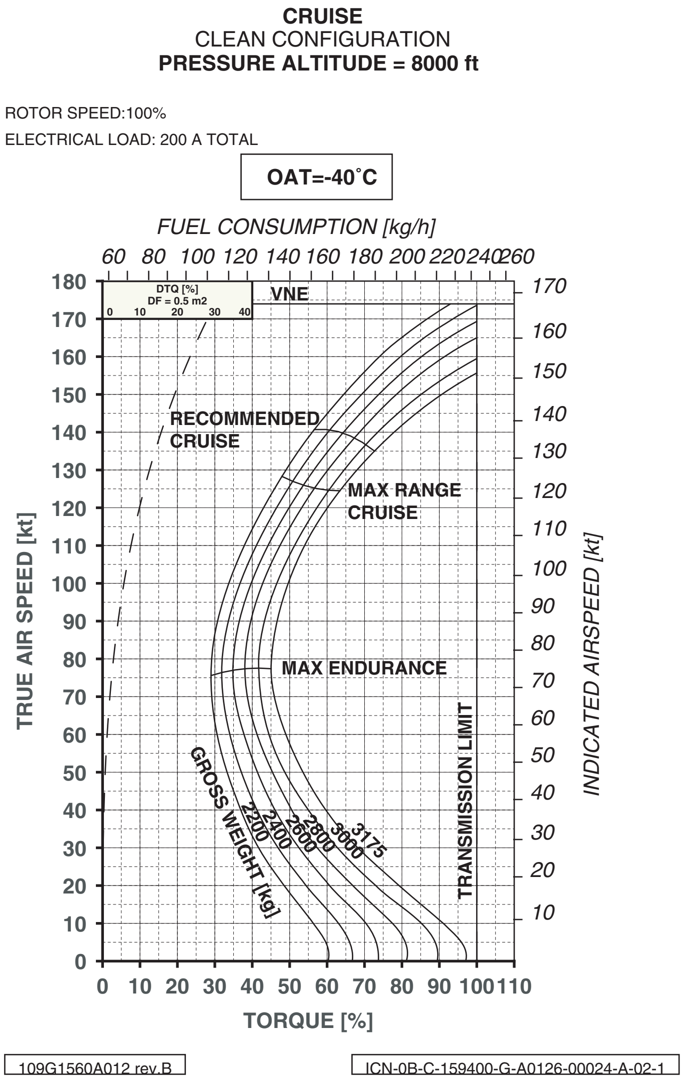

Figure 9-24 Cruise - Pressure Altitude 8000 ft, OAT -40 °C

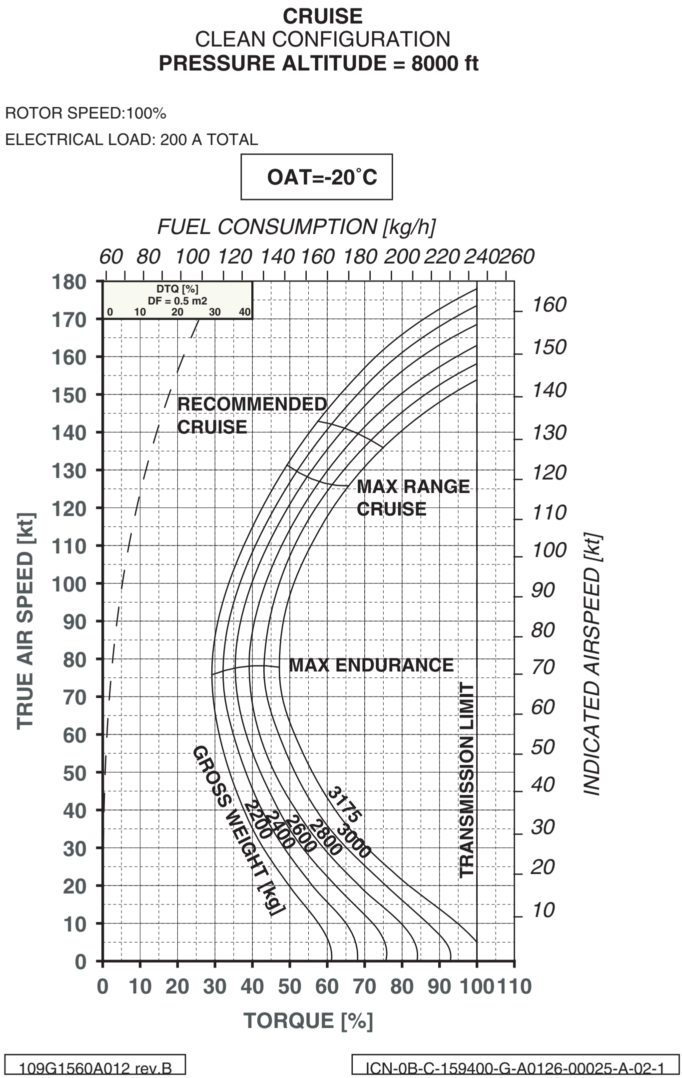

Figure 9-25 Cruise - Pressure Altitude 8000 ft, OAT -20 °C

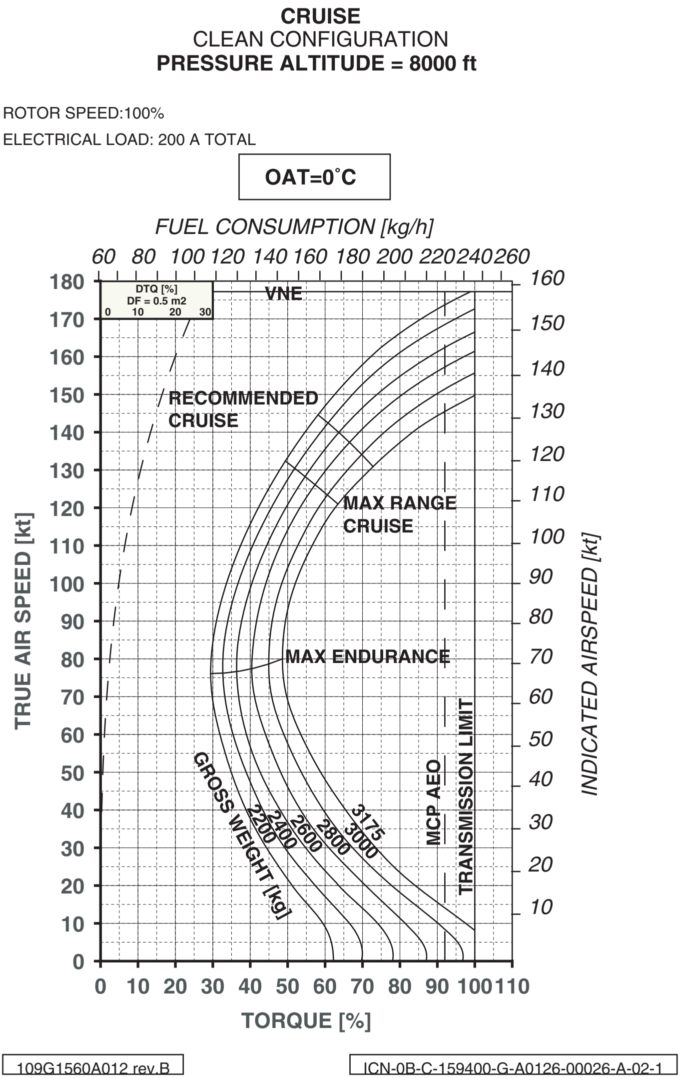

Figure 9-26 Cruise - Pressure Altitude 8000 ft, OAT 0 °C

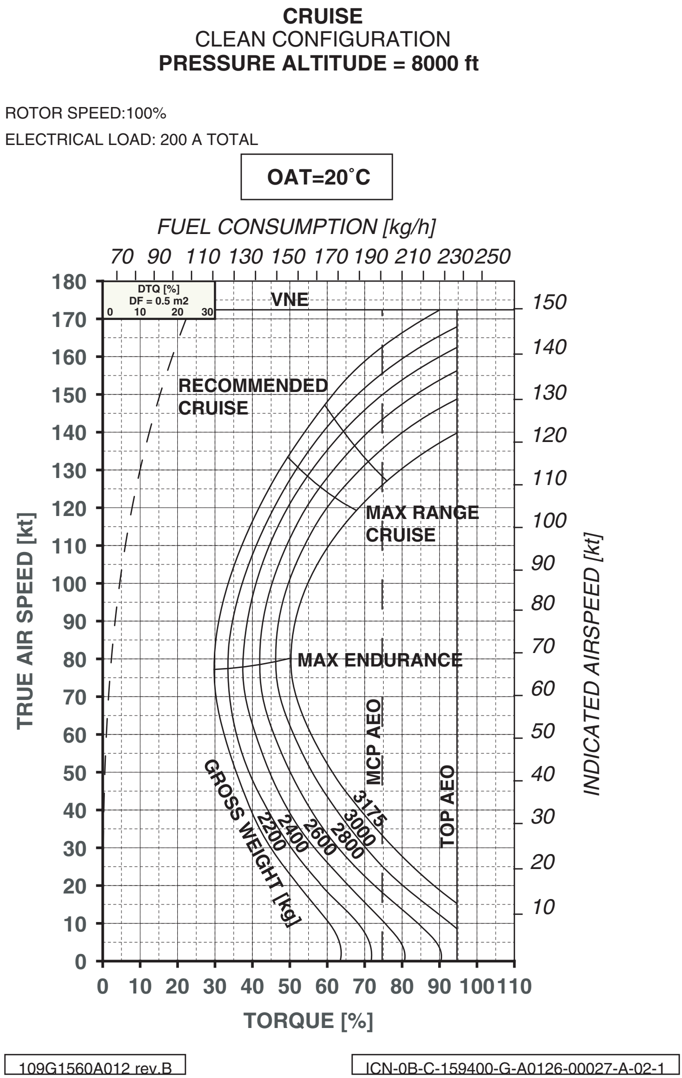

Figure 9-27 Cruise - Pressure Altitude 8000 ft, OAT 20 °C

CRUISE
CLEAN CONFIGURATION
PRESSURE ALTITUDE = 8000 ft

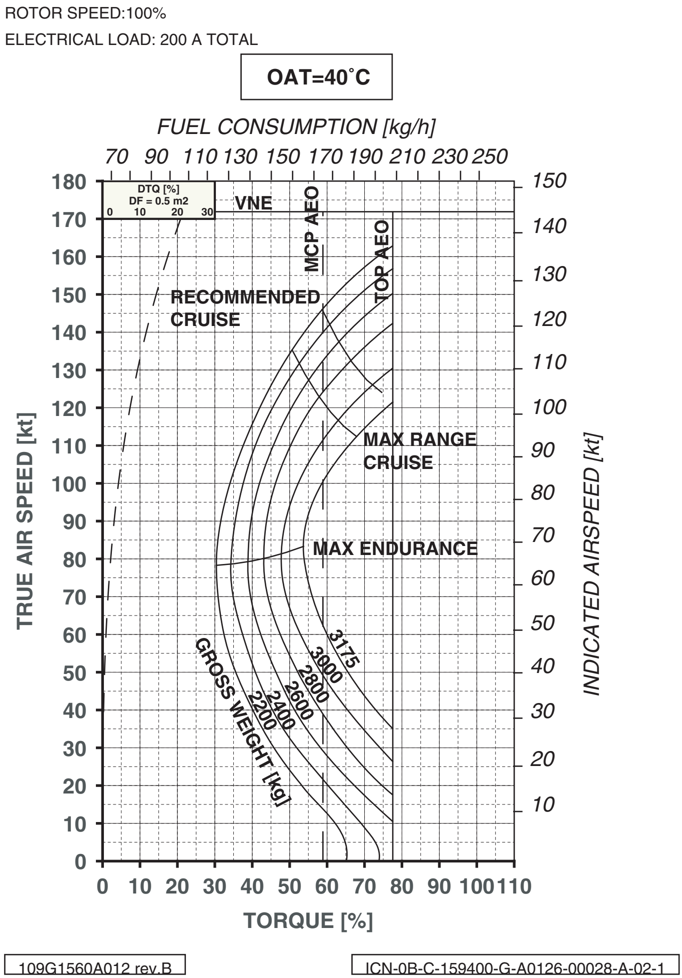

Figure 9-28 Cruise - Pressure Altitude 8000 ft, OAT 40 °C

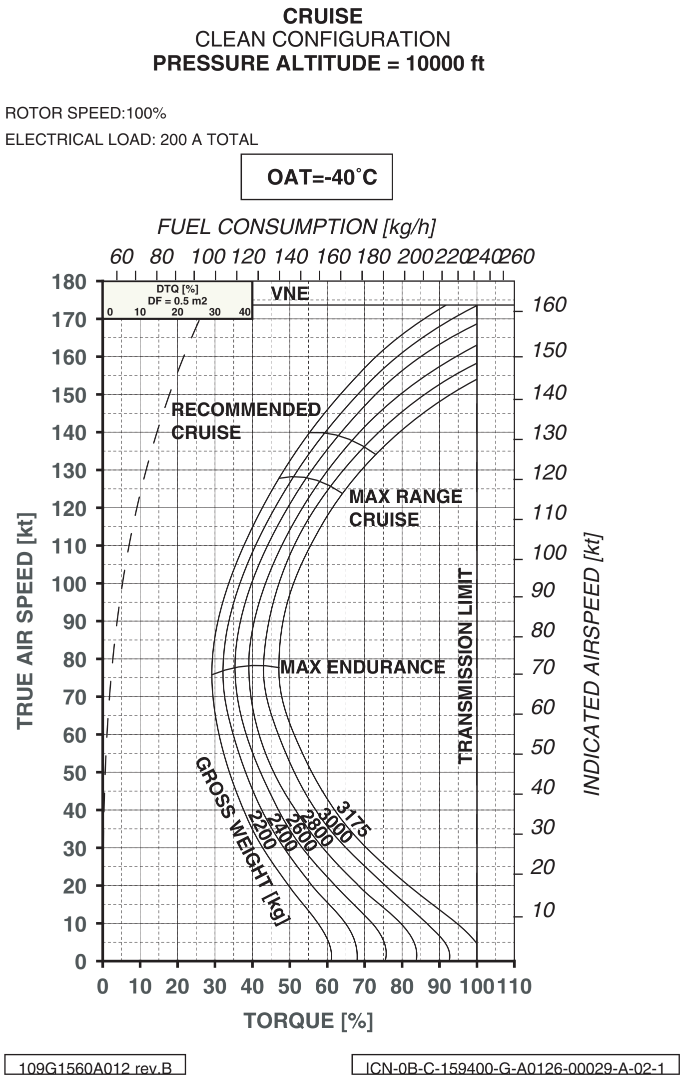

Figure 9-29 Cruise - Pressure Altitude 10000 ft, OAT -40 °C

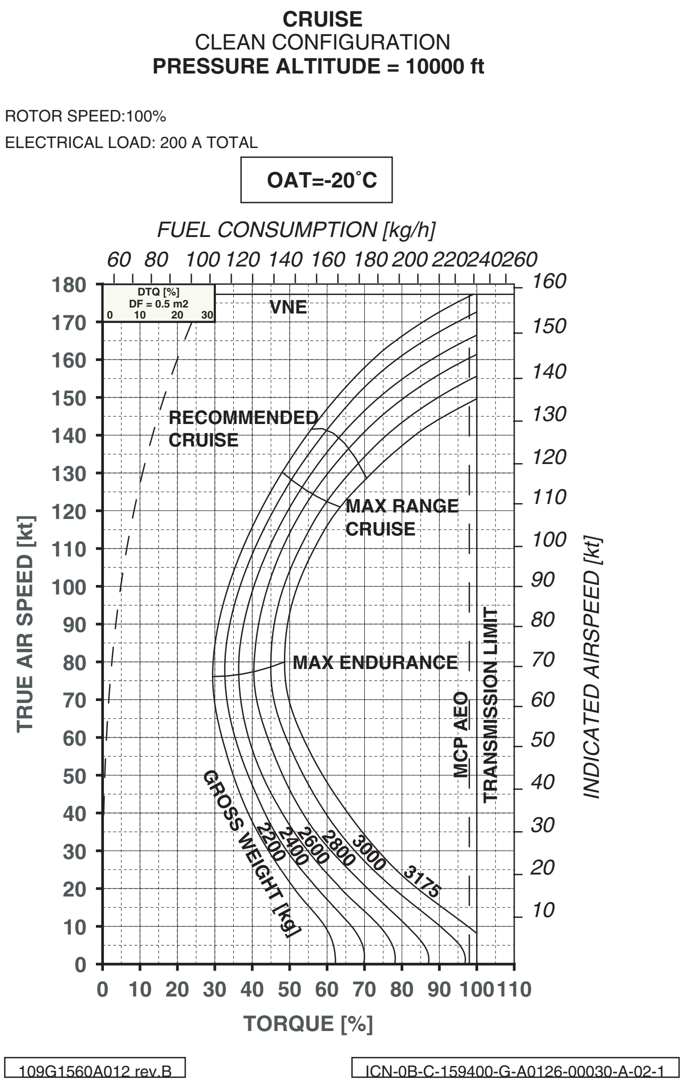

Figure 9-30 Cruise - Pressure Altitude 10000 ft, OAT -20 °C

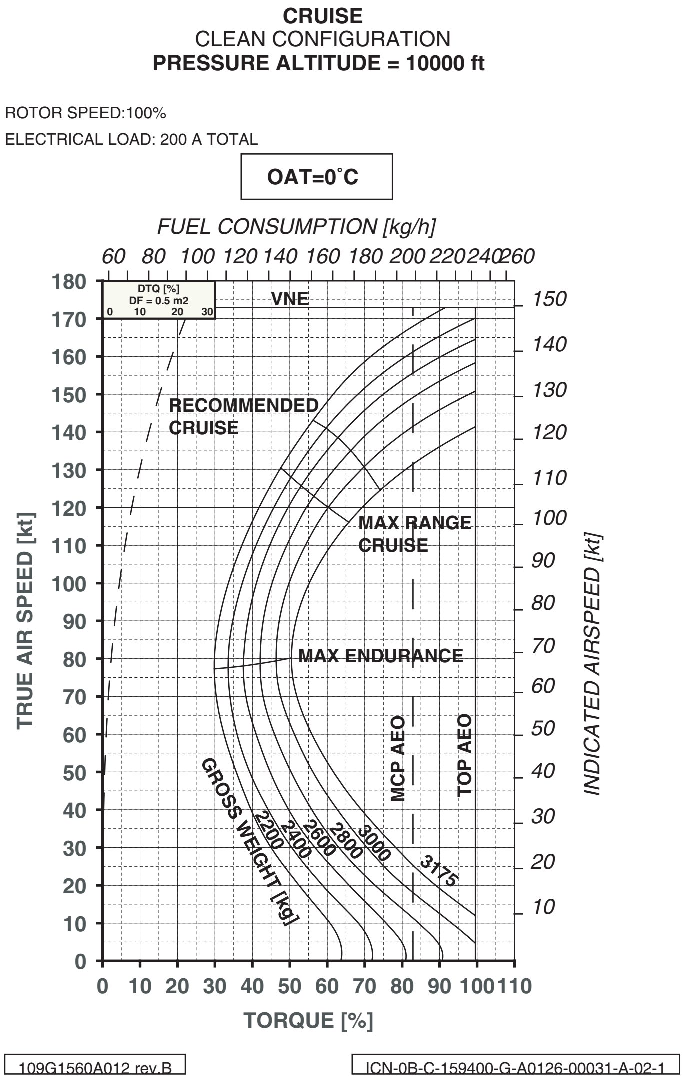

Figure 9-31 Cruise - Pressure Altitude 10000 ft, OAT 0 °C

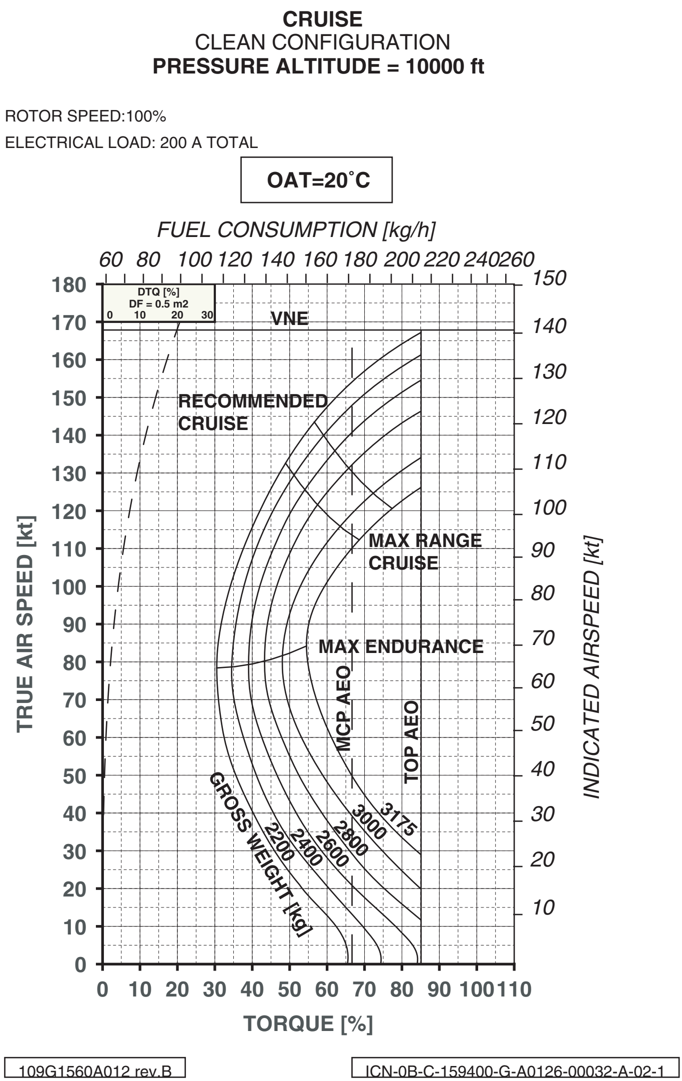

Figure 9-32 Cruise - Pressure Altitude 10000 ft, OAT 20 °C

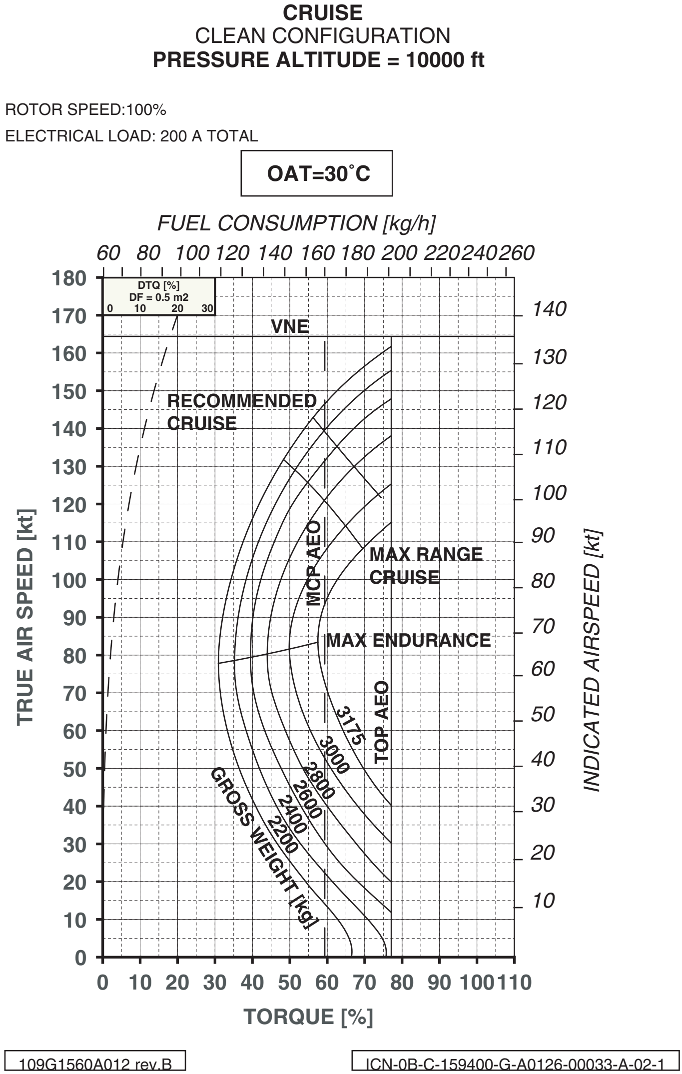

Figure 9-33 Cruise - Pressure Altitude 10000 ft, OAT 30 °C

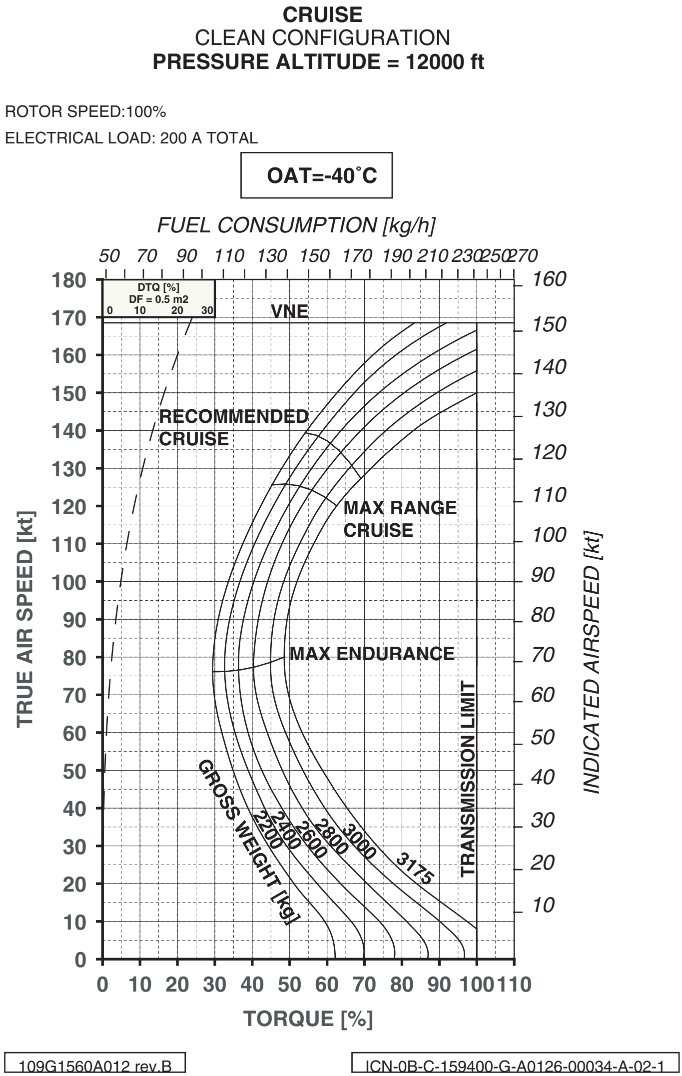

Figure 9-34 Cruise - Pressure Altitude 12000 ft, OAT -40 °C

CRUISE
CLEAN CONFIGURATION
PRESSURE ALTITUDE = 12000 ft

ROTOR SPEED:100%
ELECTRICAL LOAD: 200 A TOTAL

OAT=-20˚C

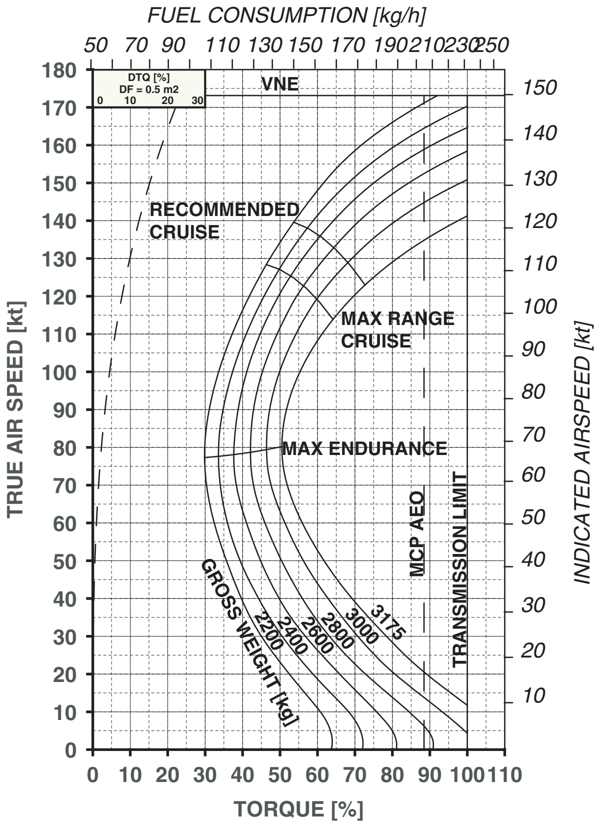

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Figure 9-35 Cruise - Pressure Altitude 12000 ft, OAT -20 °C

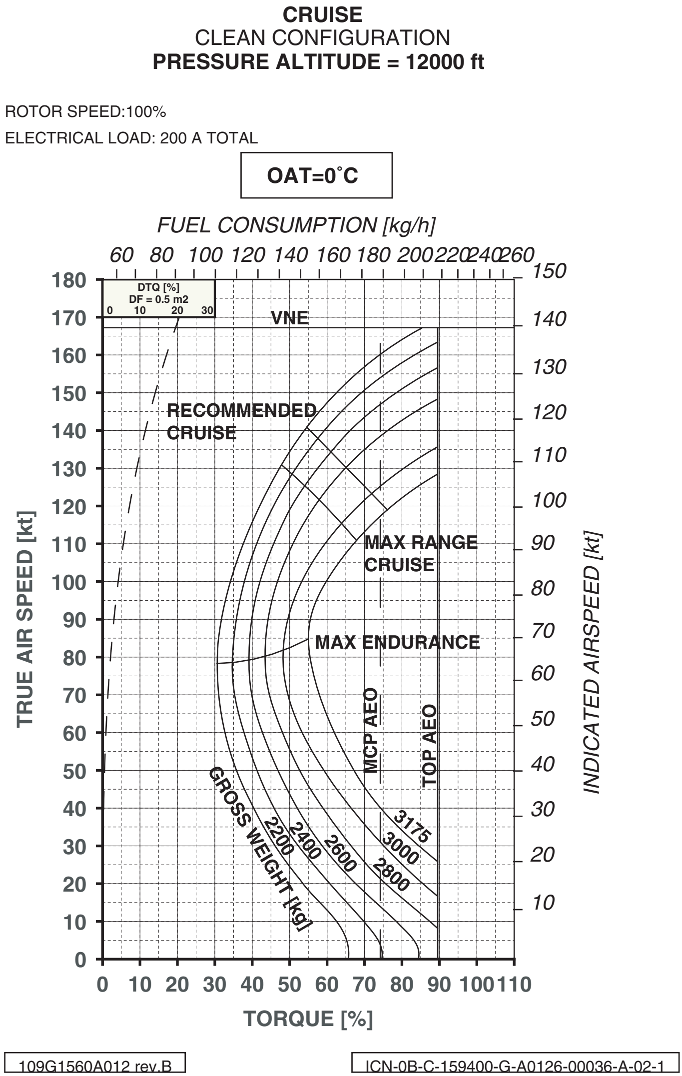

Figure 9-36 Cruise - Pressure Altitude 12000 ft, OAT 0 °C

CRUISE
CLEAN CONFIGURATION
PRESSURE ALTITUDE = 12000 ft

ROTOR SPEED:100%
ELECTRICAL LOAD: 200 A TOTAL

OAT=20˚C

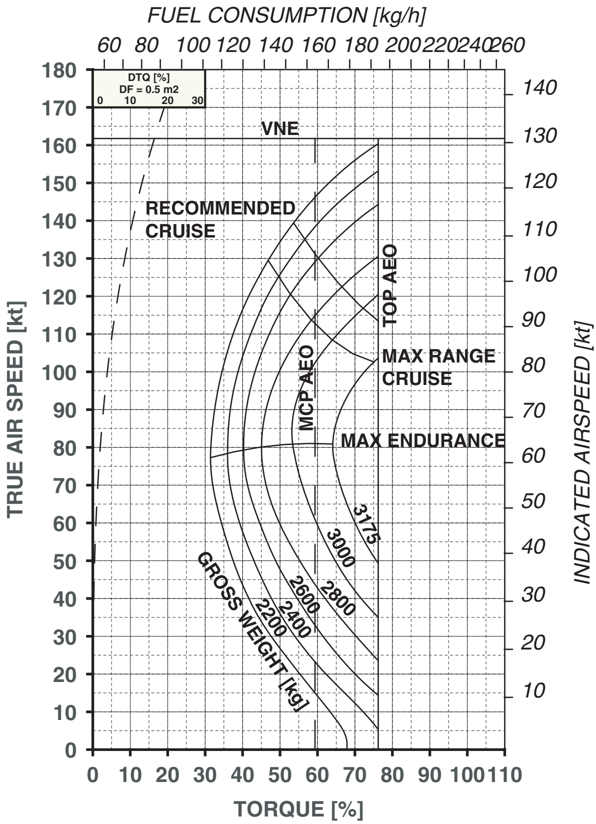

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Figure 9-37 Cruise - Pressure Altitude 12000 ft, OAT 20 °C

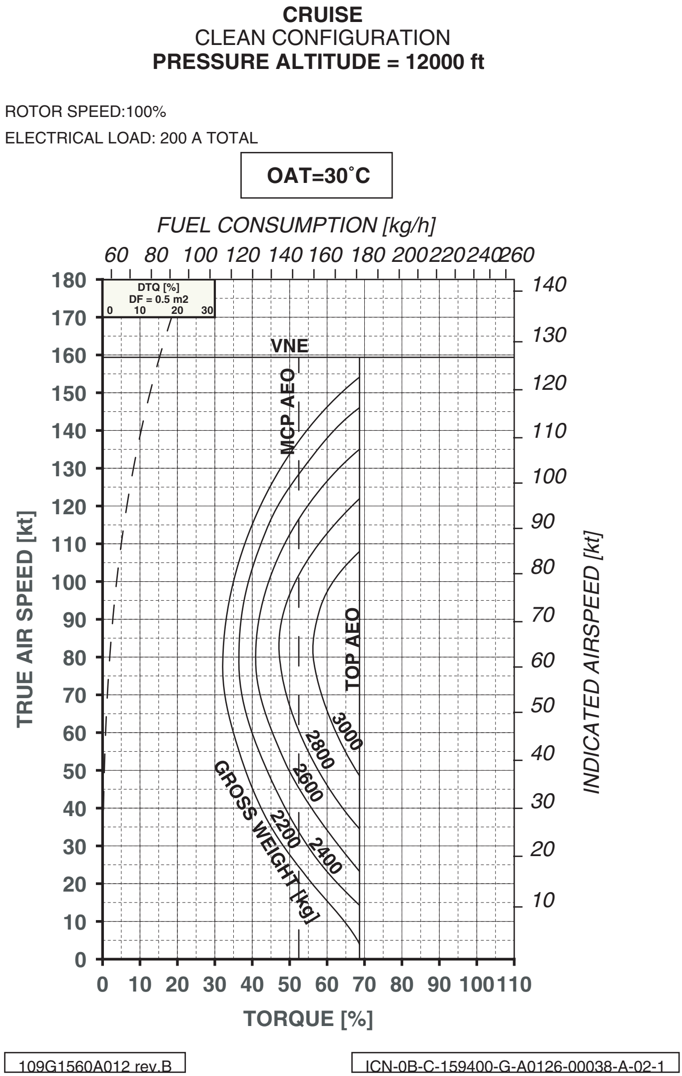

Figure 9-38 Cruise - Pressure Altitude 12000 ft, OAT 30 °C

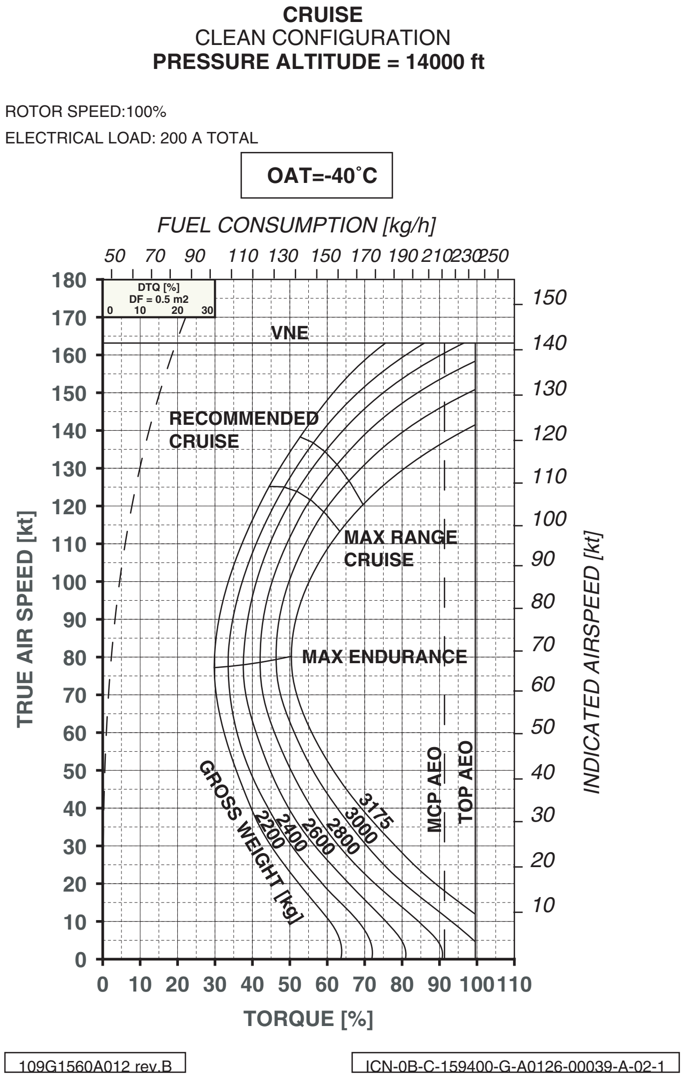

Figure 9-39 Cruise - Pressure Altitude 14000 ft, OAT -40 °C

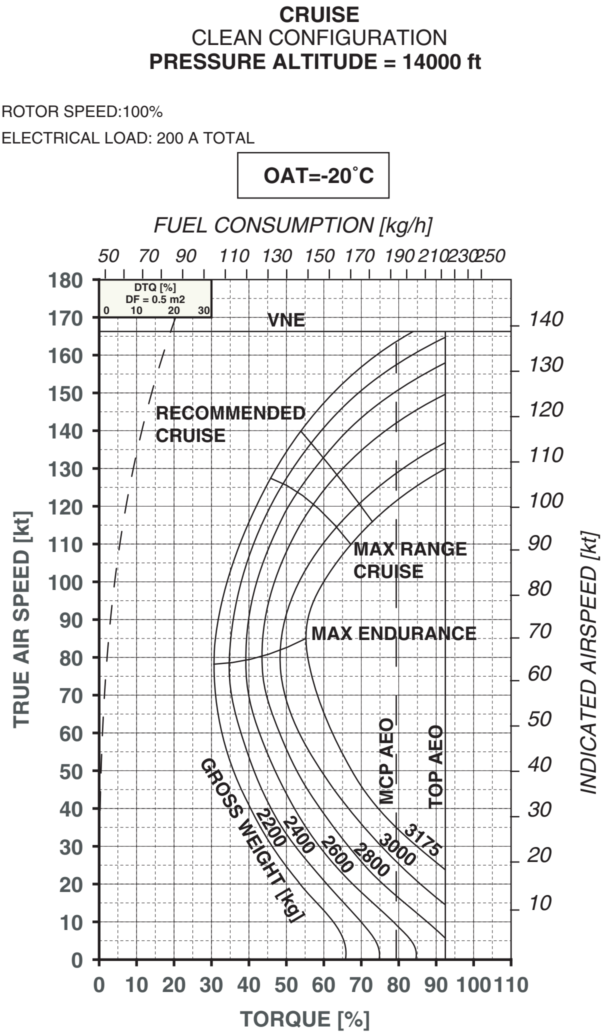

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Figure 9-40 Cruise - Pressure Altitude 14000 ft, OAT -20 °C

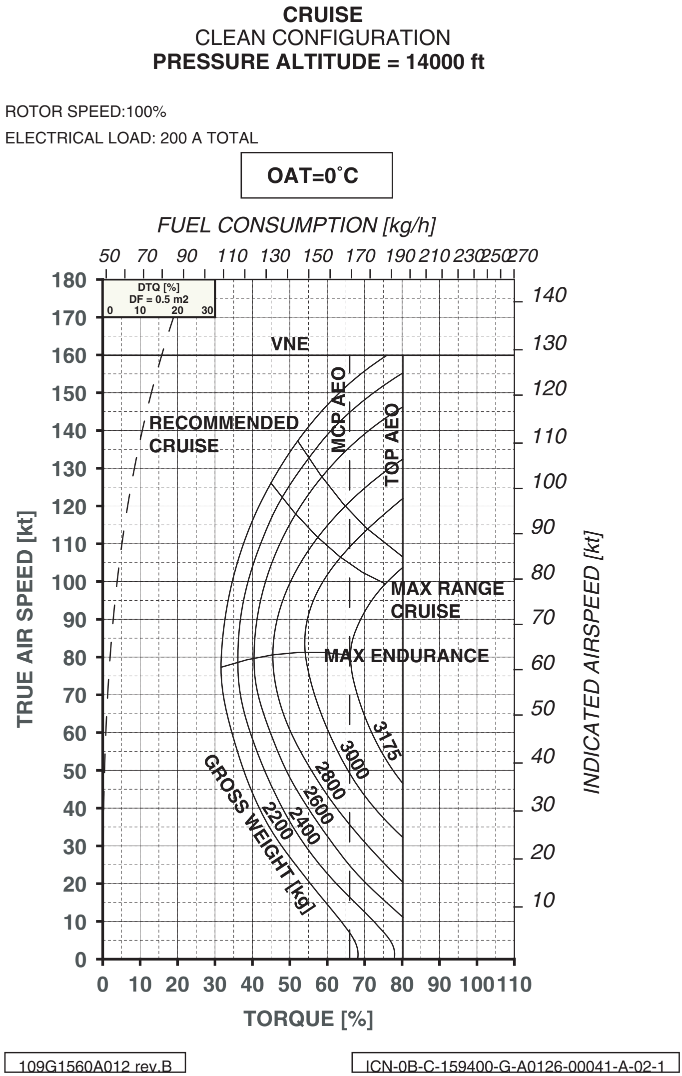

Figure 9-41 Cruise - Pressure Altitude 14000 ft, OAT 0 °C

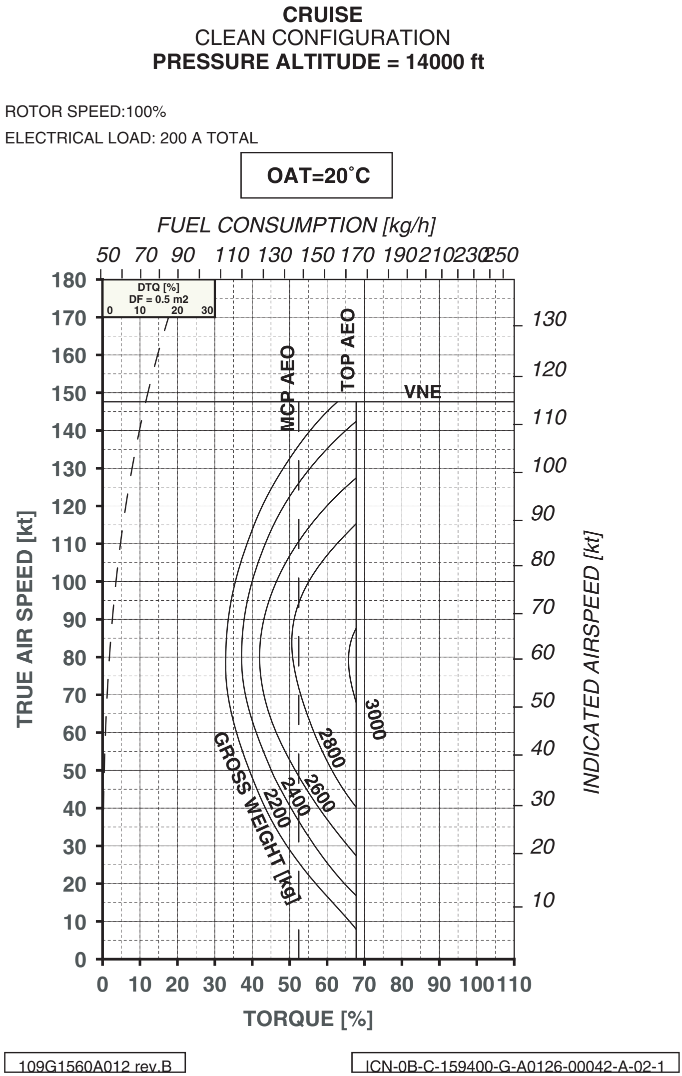

Figure 9-42 Cruise - Pressure Altitude 14000 ft, OAT 20 °C

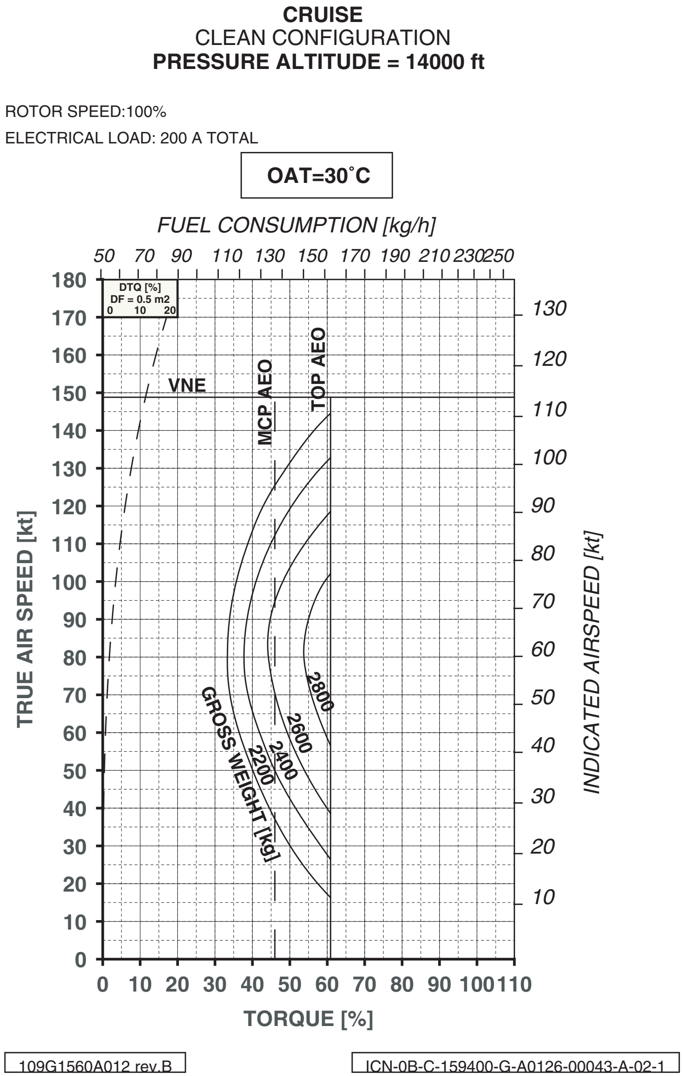

Figure 9-43 Cruise - Pressure Altitude 14000 ft, OAT 30 °C

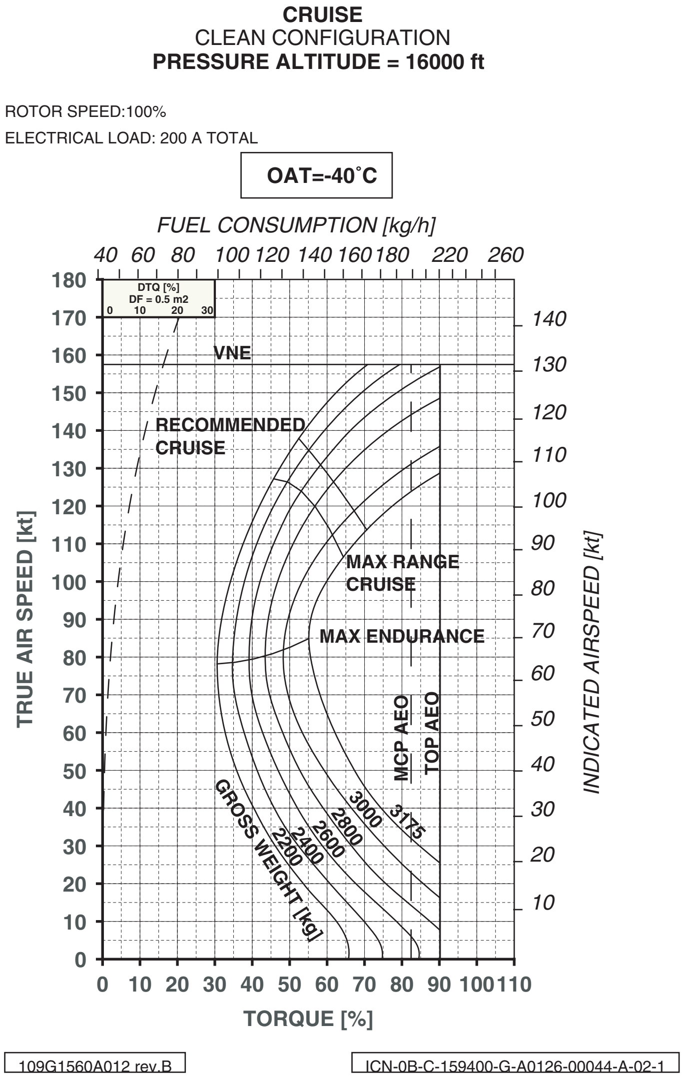

Figure 9-44 Cruise - Pressure Altitude 16000 ft, OAT -40 °C

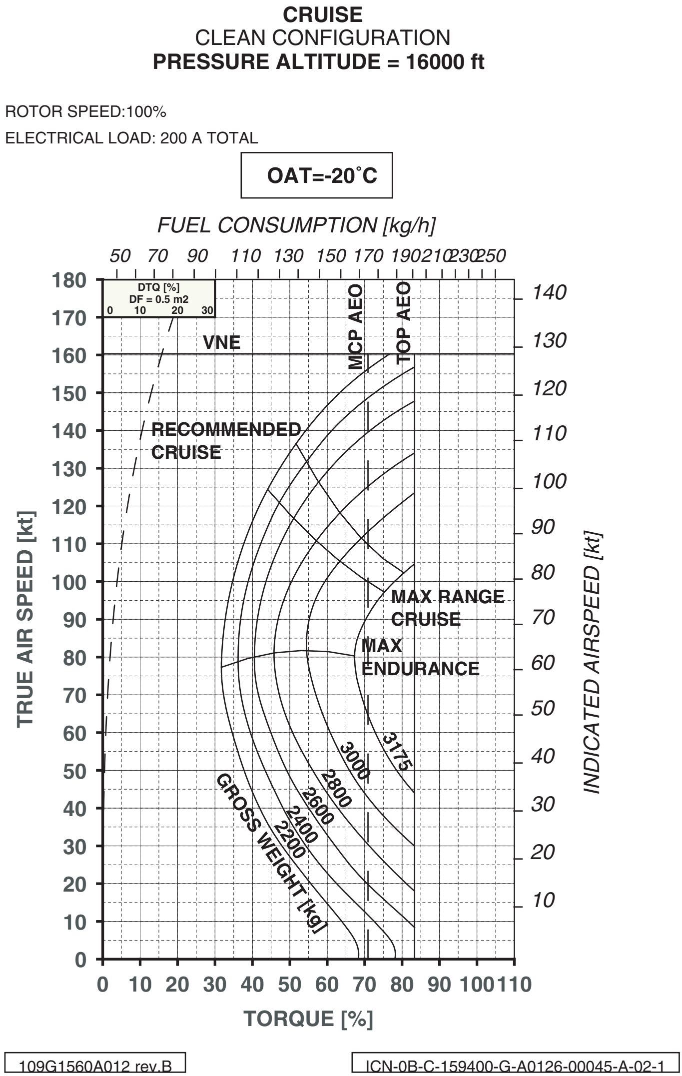

Figure 9-45 Cruise - Pressure Altitude 16000 ft, OAT -20 °C

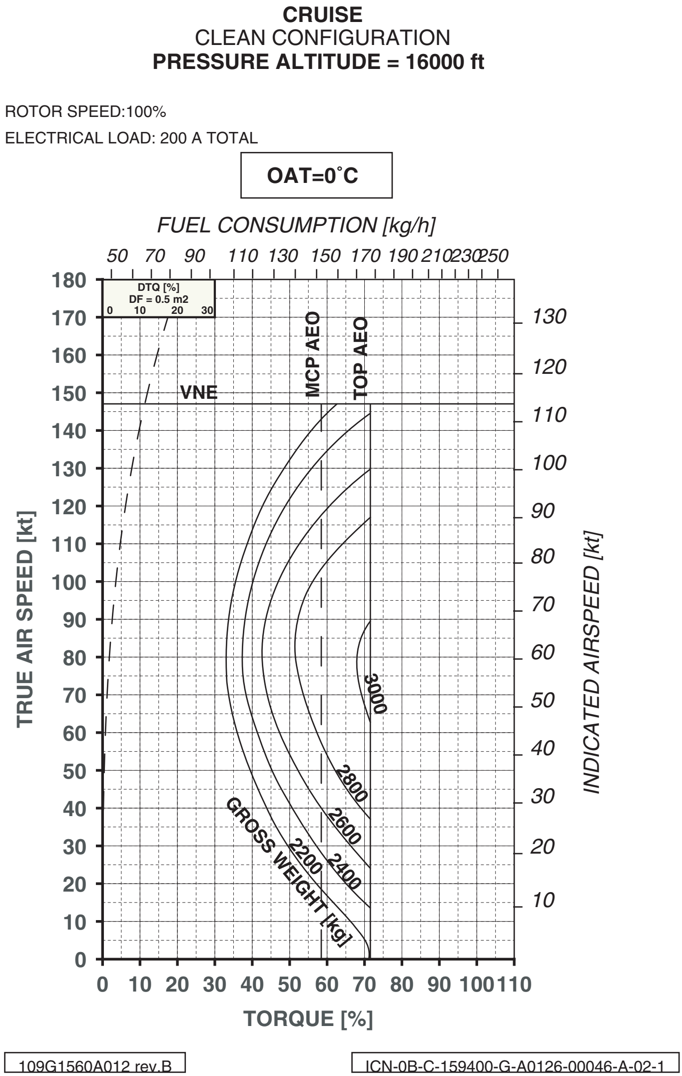

Figure 9-46 Cruise - Pressure Altitude 16000 ft, OAT 0 °C

CRUISE
CLEAN CONFIGURATION
PRESSURE ALTITUDE = 16000 ft

ROTOR SPEED:100%
ELECTRICAL LOAD: 200 A TOTAL

OAT=20˚C

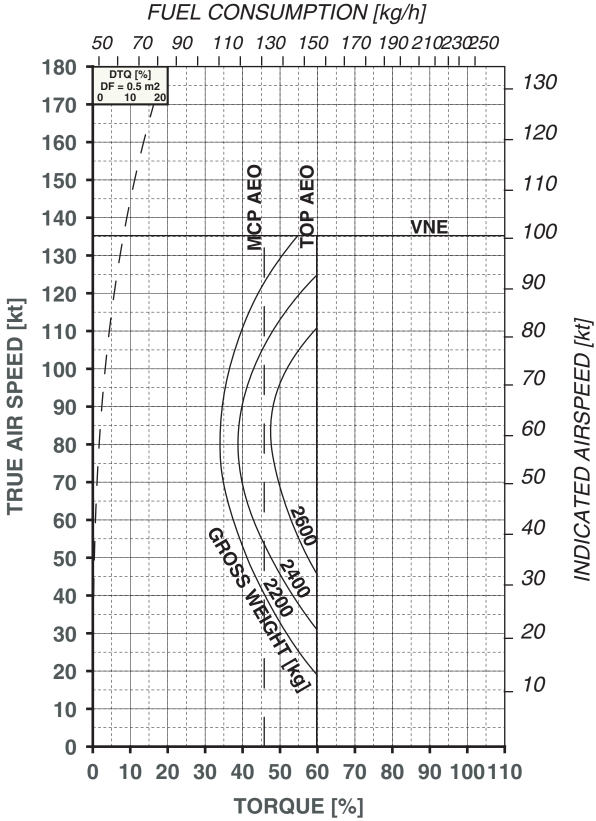

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Figure 9-47 Cruise - Pressure Altitude 16000 ft, OAT 20 °C

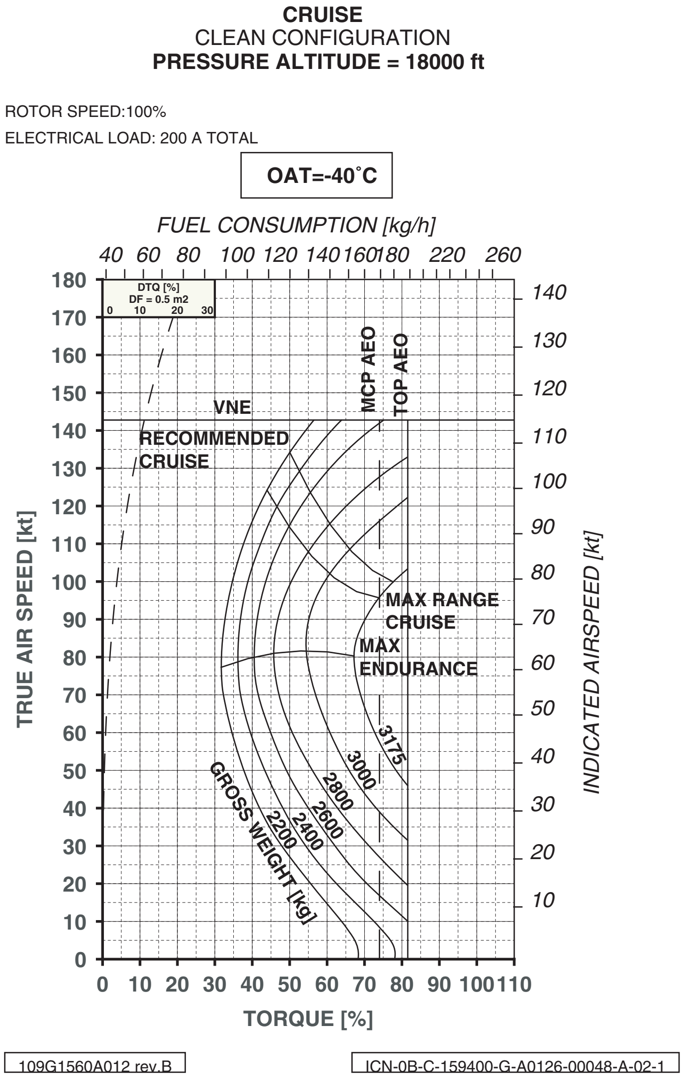

Figure 9-48 Cruise - Pressure Altitude 18000 ft, OAT -40 °C

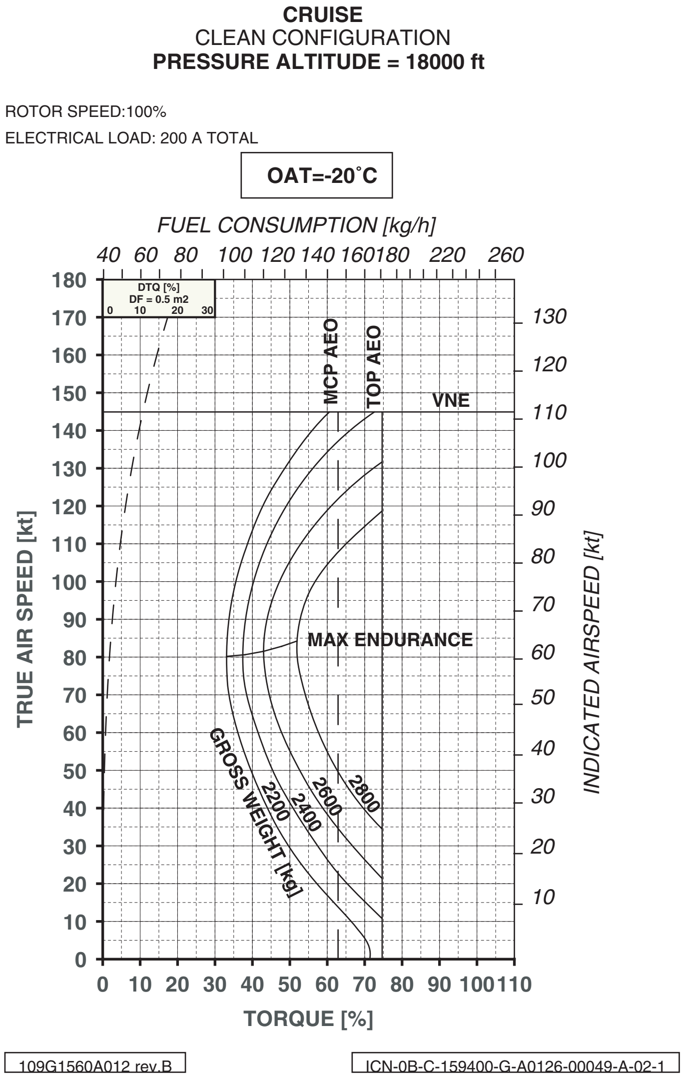

Figure 9-49 Cruise - Pressure Altitude 18000 ft, OAT -20 °C

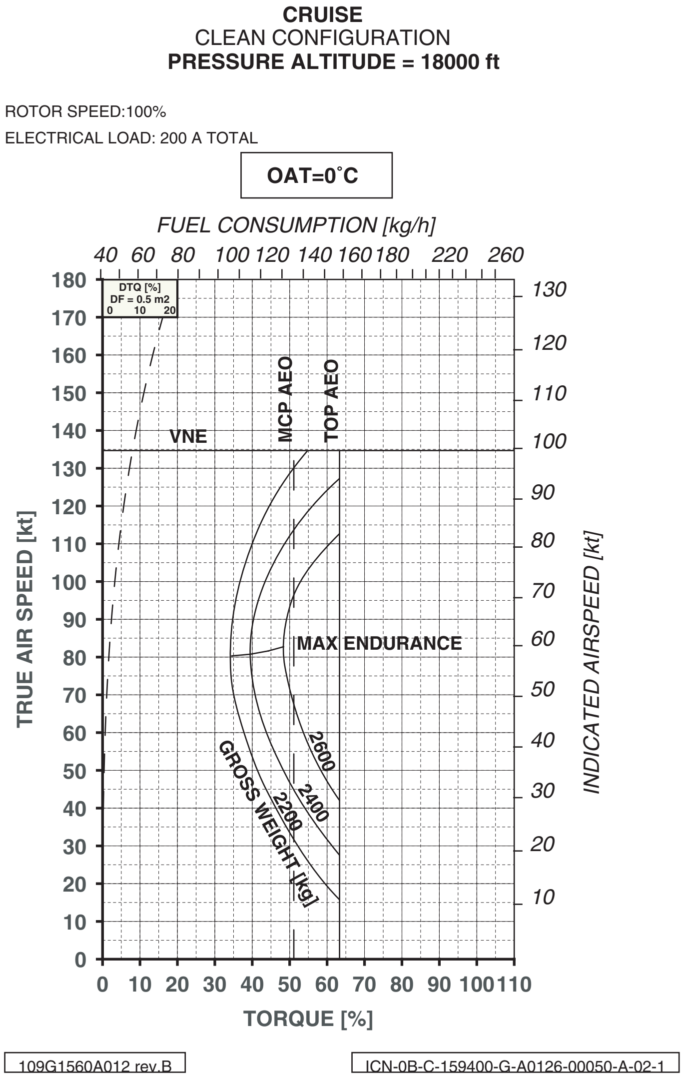

Figure 9-50 Cruise - Pressure Altitude 18000 ft, OAT 0 °C

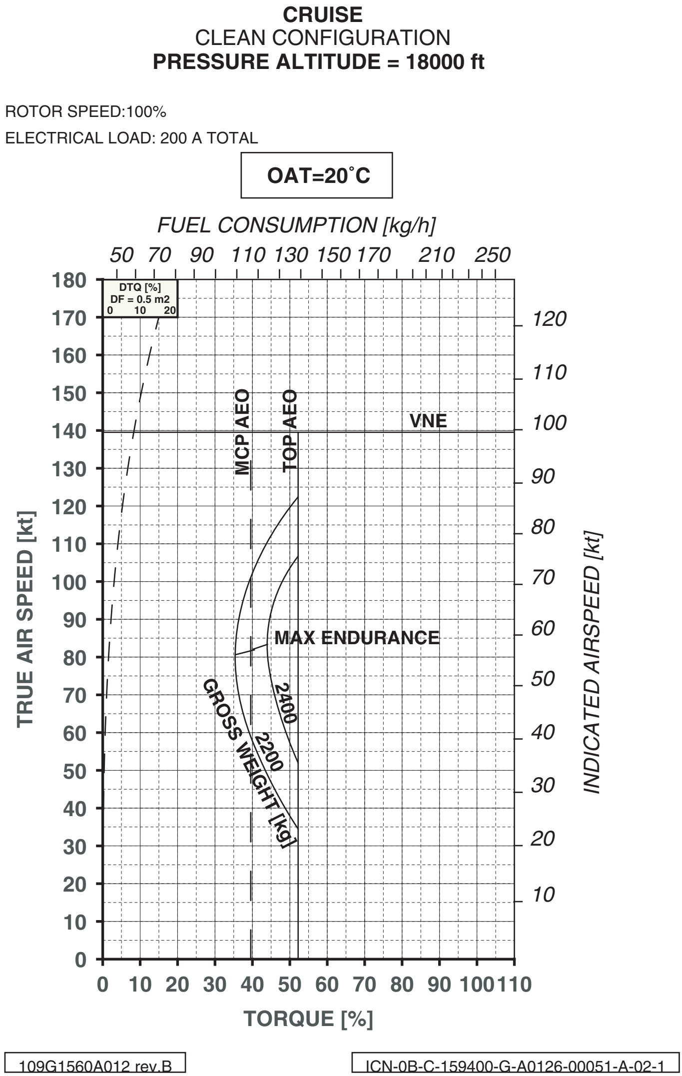

Figure 9-51 Cruise - Pressure Altitude 18000 ft, OAT 20 °C

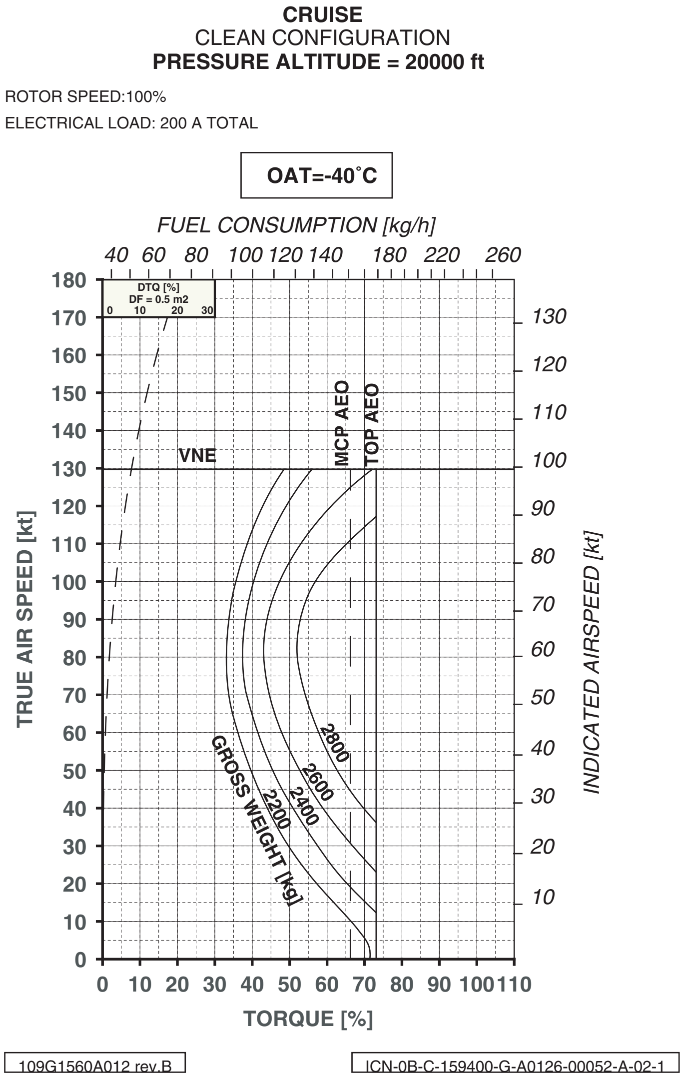

Figure 9-52 Cruise - Pressure Altitude 20000 ft, OAT -40 °C

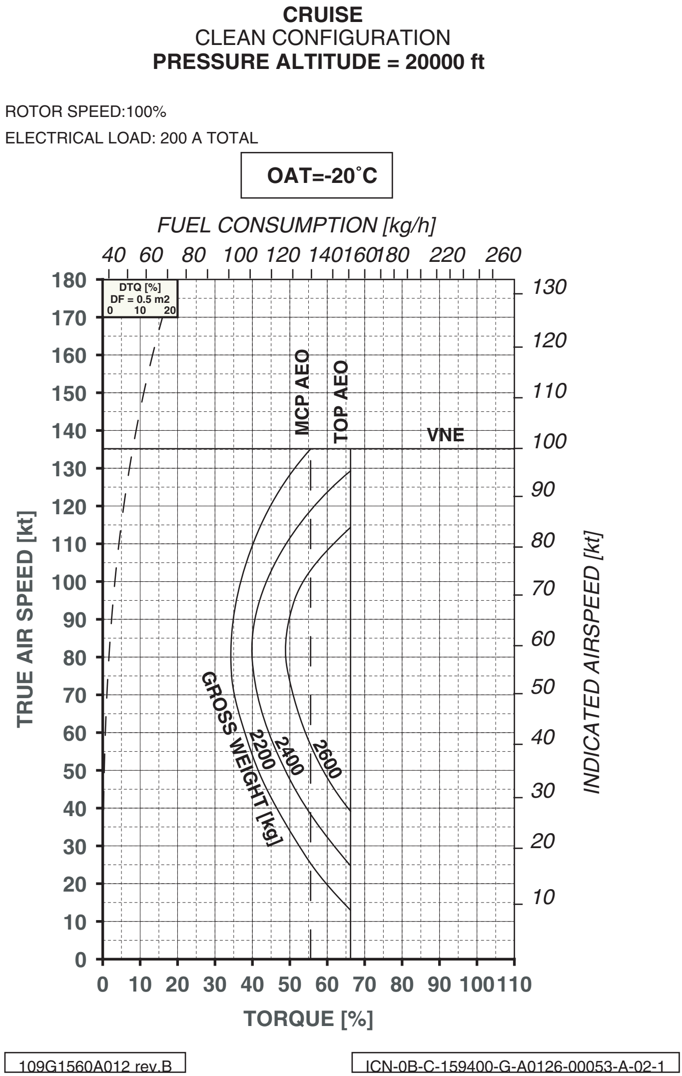

Figure 9-53 Cruise - Pressure Altitude 20000 ft, OAT -20 °C

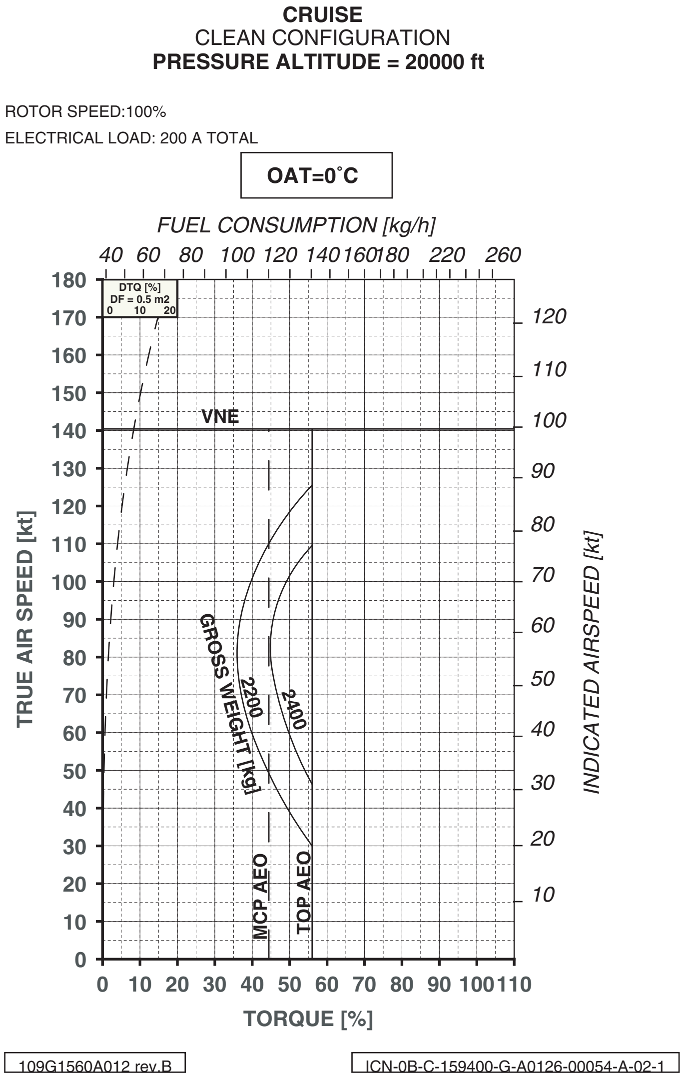

Figure 9-54 Cruise - Pressure Altitude 20000 ft, OAT 0 °C

RANGE AND ENDURANCE

Figure 9-55 shows the abacus to obtain endurance, distance and specific range in relation to fuel consumption.

Note

The range evaluated with the diagram corresponds to effective range on ground (wind=0).
To take into account the wind effect in the range calculation, use TAS reduced by the headwind component (ground speed) before entering the graph.

EXAMPLE WITHOUT WIND

Refer to Figure 9-56.

Wanted	Endurance, distance and specific range
Known	Fuel consumption: 166 kg/h Available fuel: 600 kg TAS: 130 kt
Method	Enter the graph at fuel consumption 166 kg/h. Move up and intersect available fuel 600 kg line. Move left and read 3.61h (3h:36':52"). Return to 166 kg/h, than move up to intersect True airspeed 130 kt line. Move left to intersect available fuel 600 kg line. Move left and read 0.78 nm/kg, or move down and read 470 nm, than move up and read 870 km.

EXAMPLE WITH WIND

Refer to Figures 9-55 and 9-56.

Wanted TAS corrected with headwind and respective range.

Known Wind: 25 kt from 35° respect to flight direction.
TAS: 150 kt.

Method Enter the chart (Figure 9-56) at 25 kt reported wind speed (point A), proceed right following shape of curved lines to wind relative angle 35° (point B). Then, proceed right horizontally to read 20 on the headwind component scale (point C).
Reduce TAS by 20 kt to obtain a ground speed of 130 kt.
Enter the graph (Figure 9-55) at fuel consumption of 166 kg/h, TAS corrected of 130 kt and available fuel of 600 kg to obtain a distance of 470 nm or 870 km with an endurance of 3.61 h.

RANGE AND ENDURANCE

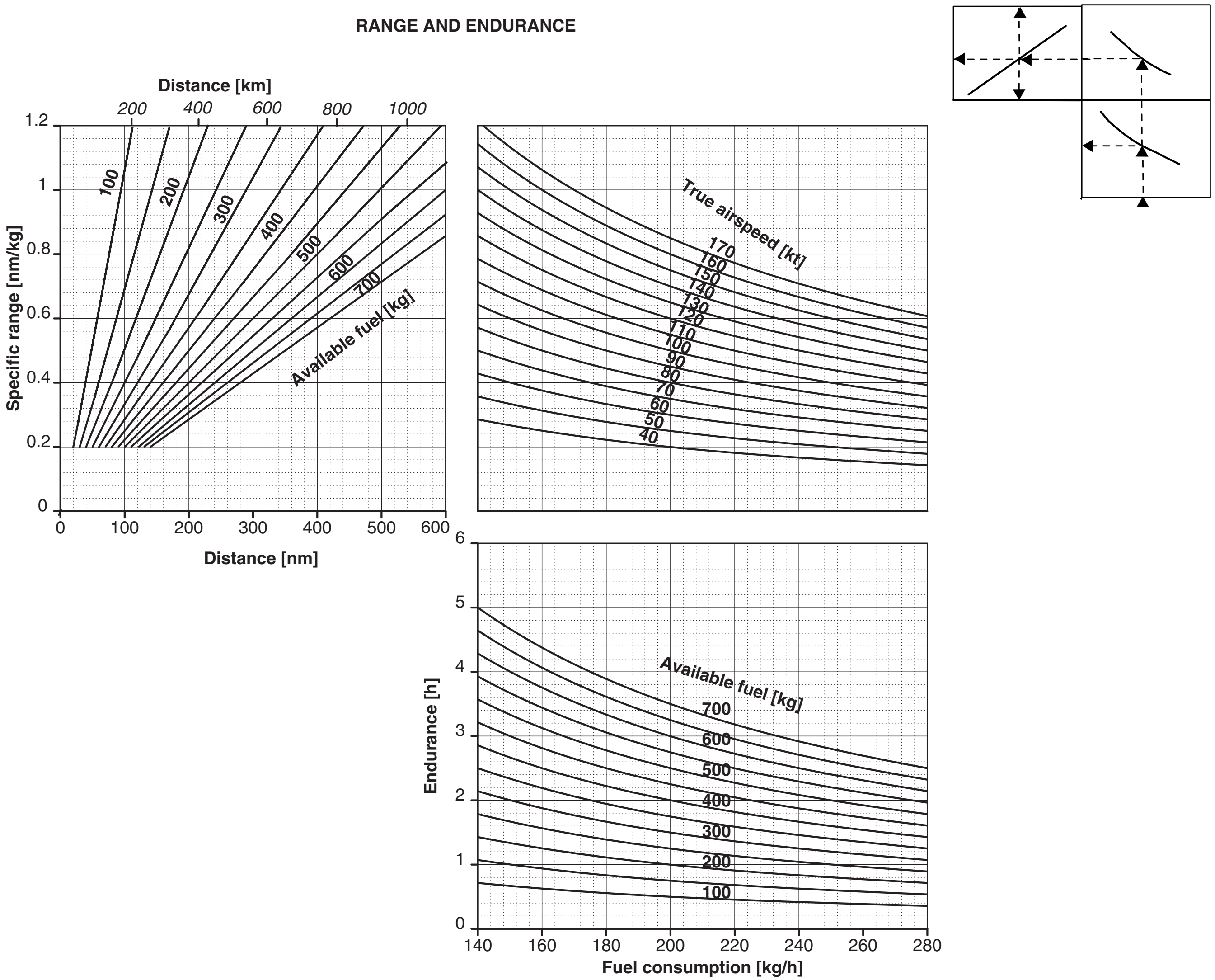

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Figure 9-55 Range and Endurance

WIND COMPONENT CHART

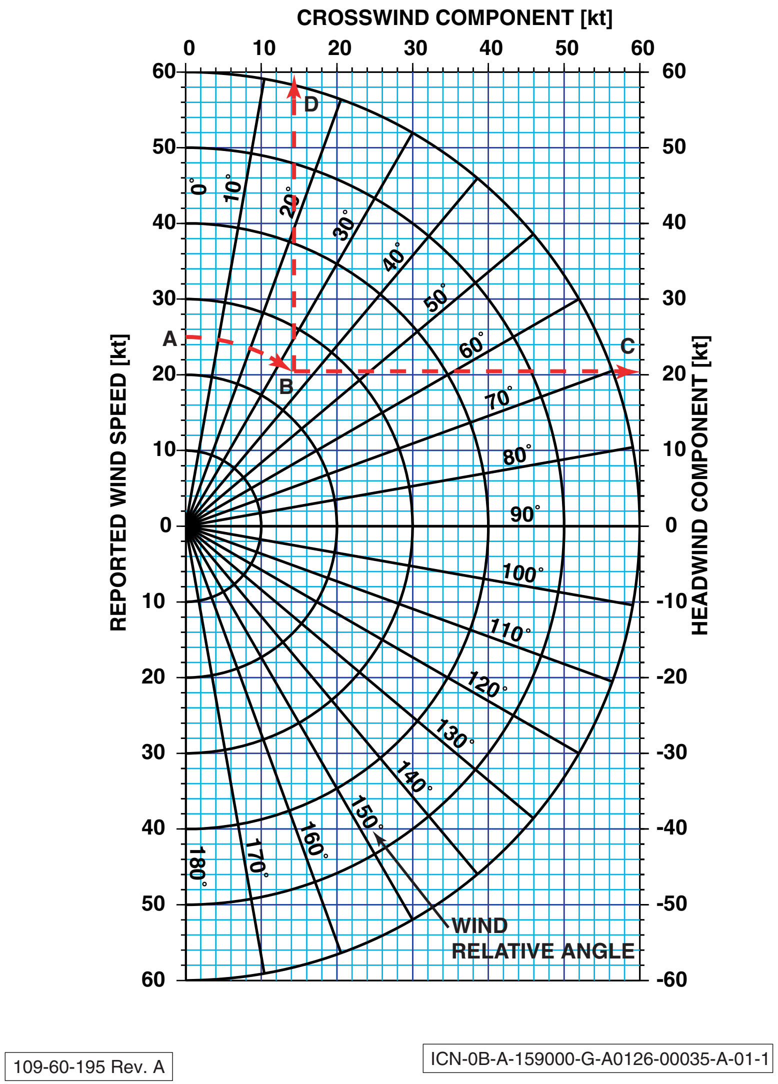

Figure 9-56 Wind Component Chart

HOVER CEILING - ONE ENGINE INOPERATIVE CHARTS

The hover ceiling charts are presented for the One Engine Inoperative (OEI), 2.5 minute power, In Ground Effect (IGE) and Out of Ground Effect (OGE) with headwind effect.

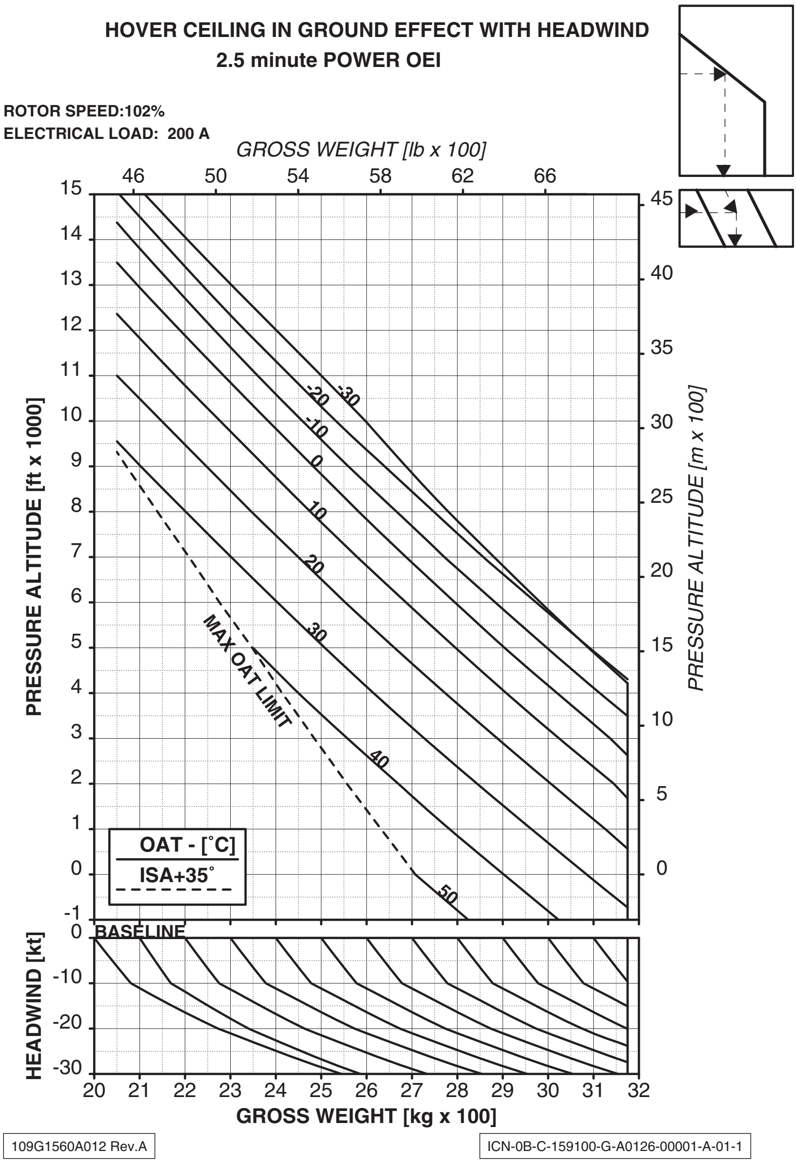

Figure 9-57 Hover Ceiling - IGE, OEI, 2.5 Minute Power with headwind effect

HOVER CEILING OUT OF GROUND EFFECT WITH HEADWIND
2.5 minute POWER OEI

ROTOR SPEED:102% **ELECTRICAL LOAD: 200 A**

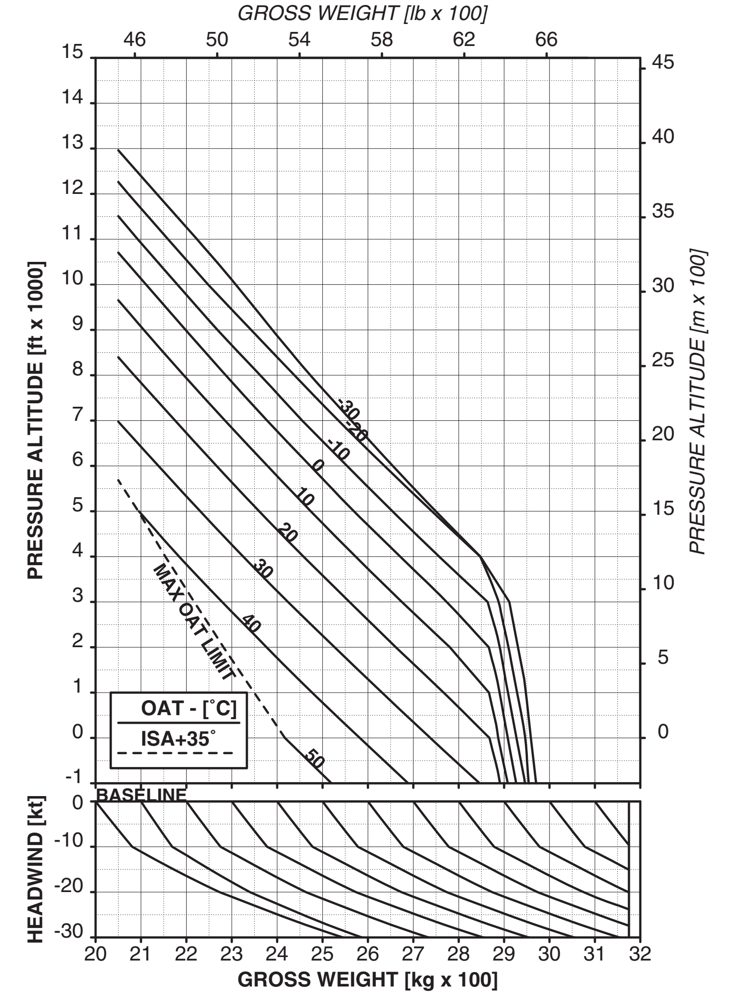

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Figure 9-58 Hover Ceiling - OGE, OEI, 2.5 Minute Power with headwind effect

TORQUE REQUIRED TO HOVER (with headwind effect)

Figure 9-61 provides Torque Required to Hover for a given condition of pressure altitude, OAT, Gross Weight, wheel height (IGE 3 ft or OGE 60 ft) at a known velocity of headwind.

Furthermore, TOP and MCP torque available (Hover Condition 102% NR) are shown in Figures 9-59 and 9-60.

EXAMPLES:

Example 1

Wanted	Torque required to Hover OGE
Known	Pressure altitude = 4000 ft OAT = 20 °C GW = 3200 kg (Cargo Hook operation) Headwind = 15 kt Wheel height = 60 ft (OGE)
Method	Enter Figure 9-61, with a pressure altitude of 4000 ft and intersect the 20 °C OAT line. Move down until the 3200 kg gross weight line (dash line) and then move right to intersect the line wheel height of 60 ft (OGE). Move up until intersecting 0 kt wind axis, then follow guidelines until intersecting the 15 kt headwind horizontal line. Then move up and read a Torque of 89%.

Example 2

Wanted	TOP and MCP Torque available in Hover OGE
Known	Pressure altitude= 4000 ft OAT= 20 °C
Method	For TOP Torque available, enter Figure 9-59, with a pressure altitude of 4000 ft and intersect the 20 °C OAT line. Move down and read 106.7% of Torque available. For MCP Torque available, enter Figure 9-60, with a pressure altitude of 4000 ft and intersect the 20 °C OAT line. Move down and read 100% of Torque available. Both the TOT and MCP Torque available (106.7% and 100% respectively) are greater than the required torque of 89%

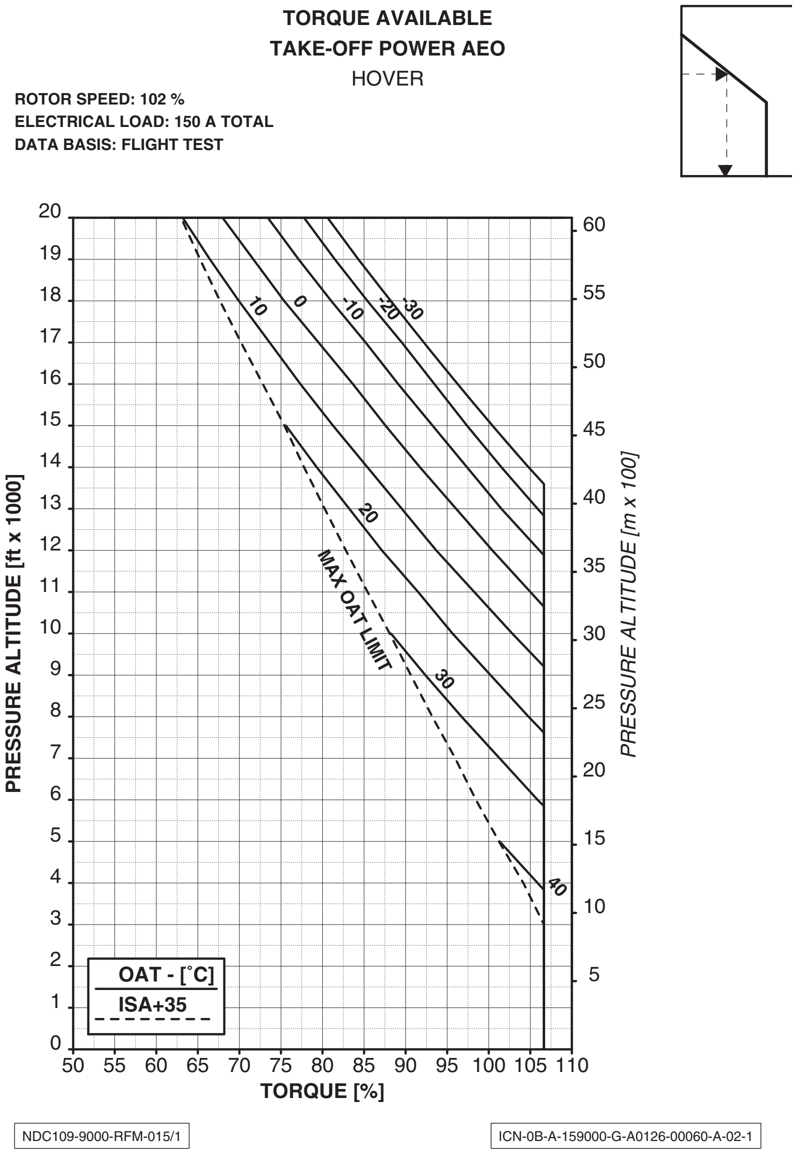

Figure 9-59 Torque Available AEO (TOP)

TORQUE AVAILABLE
MAXIMUM CONTINUOUS POWER AEO
HOVER

ROTOR SPEED: 102 %
DATA BASIS: FLIGHT TEST
ELECTRICAL LOAD: 150 A TOTAL

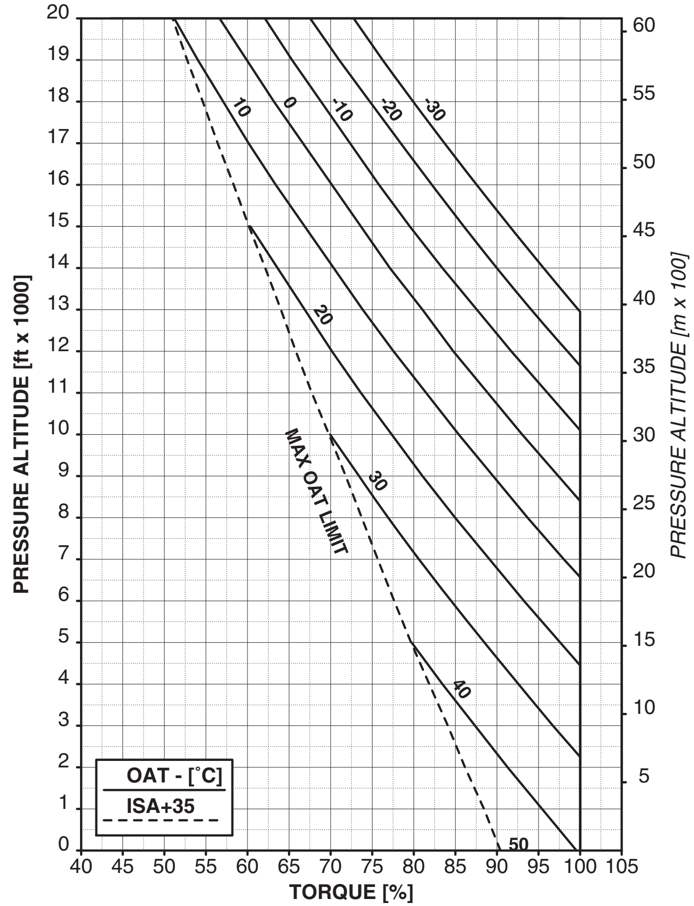

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Figure 9-60 Torque Available AEO (MCP)

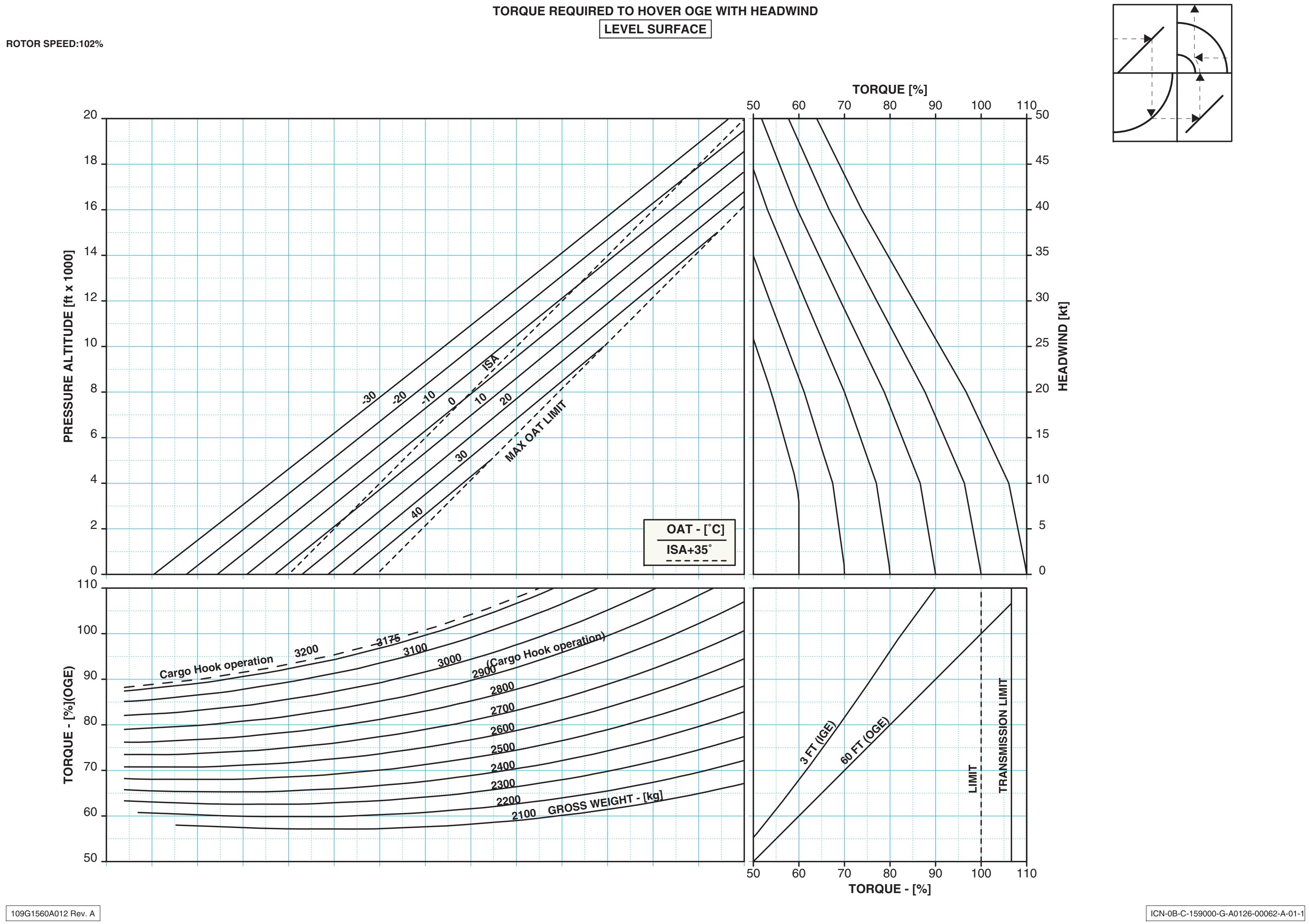

Figure 9-61 Torque Required to Hover with Wind